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AW139TYPE RATING GROUND COURSE TRAINING MANUAL

ISSUED: 18 MARCH 2009

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CHAPTER
00
AIR VEHICLE GENERAL

SECTION 00 – AIR VEHICLE DESCRIPTION

AW139 HELICOPTER

GENERAL

The AW139 is a large twin turbo-shaft engines multi-purpose helicopter designed for the following:

PRIMARY ROLES

- transport of passengers
- transport of material
- Emergency Medical Service (EMS) air ambulance

SECONDARY ROLES

- air taxi
- transport of underslung loads
- SAR – Search And Rescue (over land, sea, mountain)
- off-shore
- fire-fighting
- ecologic, radiologic surveillance
- civil protection
- law enforcement
- paramilitary

NOTE: AW139 and AB139 are two names for the same product. They identify two batches of aircraft manufactured in conformity with a unique Type Certificate Data Sheet.

AB139: up to s/n 31054
AW139: from s/n 31055 onward

JAR 29 CERTIFICATION

The AW139 is certificated by European Aviation Safety Agency (EASA) in accordance with JAR 29 for Large Rotorcraft Category A (Appendix B – IFR, and Appendix C – Icing conditions) and Category B.

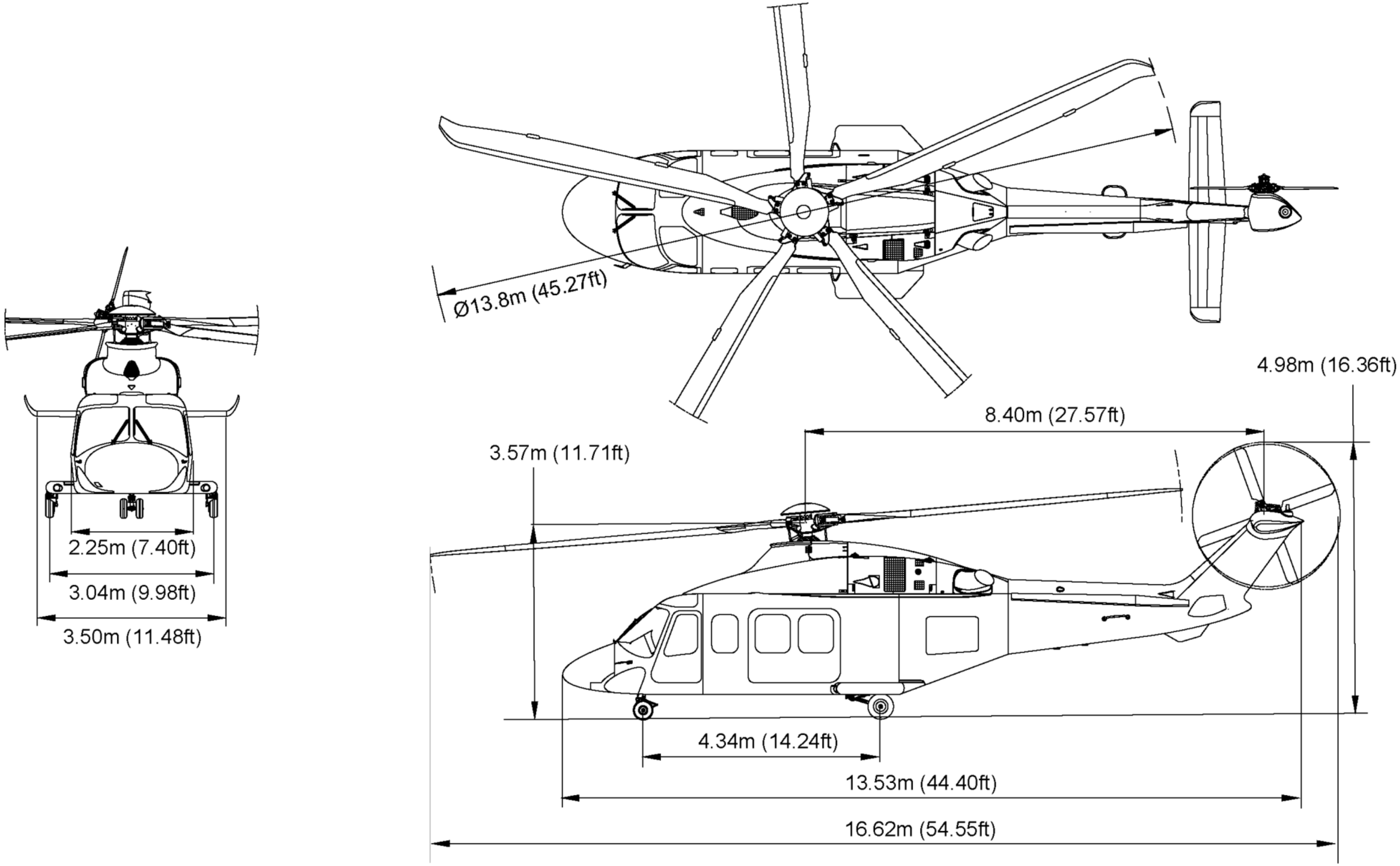
Certification includes:

- Taxiing from prepared surfaces
- Cat. 1 ILS approaches
- Overwater operations (no amphibian operations)
- Carriage of external loads
- Cold weather operations
- Hot and high operations
- IFR single pilot operations
- VFR night single pilot operations

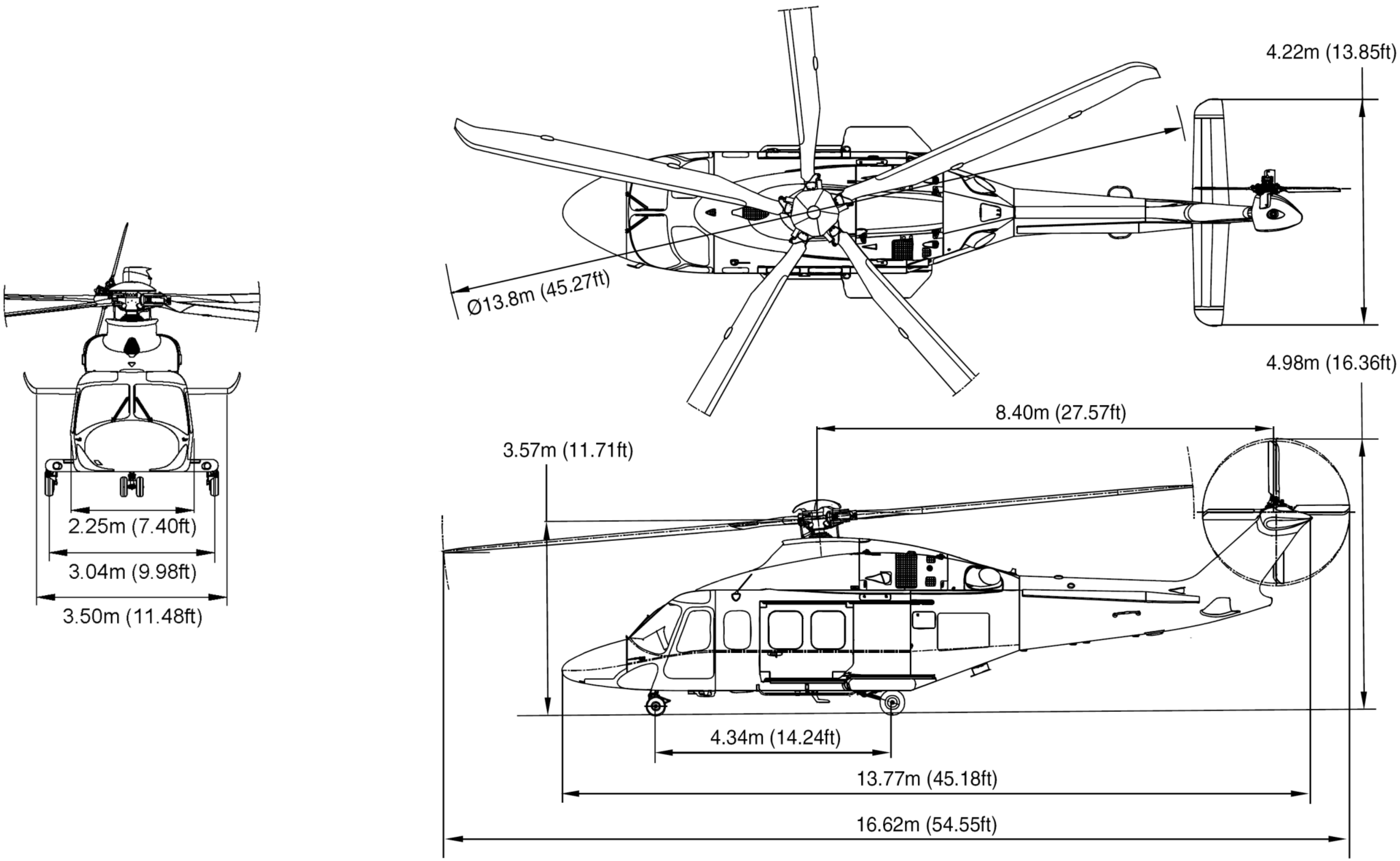
Installation of dedicated optional kits may be necessary to perform some of the certified operations.

The maximum number of occupants, including the crew, is:

- low density configuration..14
- high density configuration...17



EXTERNAL DIMENSIONS (SHORT NOSE CONFIGURATION)



EXTERNAL DIMENSIONS (LONG NOSE CONFIGURATION)

LIMITATIONS AND PERFORMANCE DATA

The limitations are listed in:

- Rotorcraft Flight Manual (RFM), Section 1
- Quick Reference Handbook (QRH), LIMITS section

The performance data are given in:

- Rotorcraft Flight Manual (RFM), Section 4
- Quick Reference Handbook (QRH), PERF section

In the RFM, Section 9, supplemental performance information are also given, such as:

- Cruise Charts, which allow determining the fuel flow, max endurance, max range and recommended cruise speeds
- Climb Gradients, which allow determining the mean height gained in 100-ft horizontal distance for different airspeeds and gross weights
- Hovering Ceiling OGE at OEI 2.5-min power with headwind benefit

ALTITUDE, TEMPERATURE AND AIRSPEED ENVELOPE

A chart, only provided in the RFM, Section 1, provides the applicable airspeed envelope of the helicopter and permits determining the V_{NE} as a function of density altitude (Hd).
The diagram also shows the altitude and temperature limits.

V_{NE} is continuously computed and presented on the airspeed indicators of pilot and copilot.

When flying in emergency conditions, a placard on the instrument panel provides V_{NE} values for several pressure altitude (Hp) and OAT values.

ICING LIMITATIONS

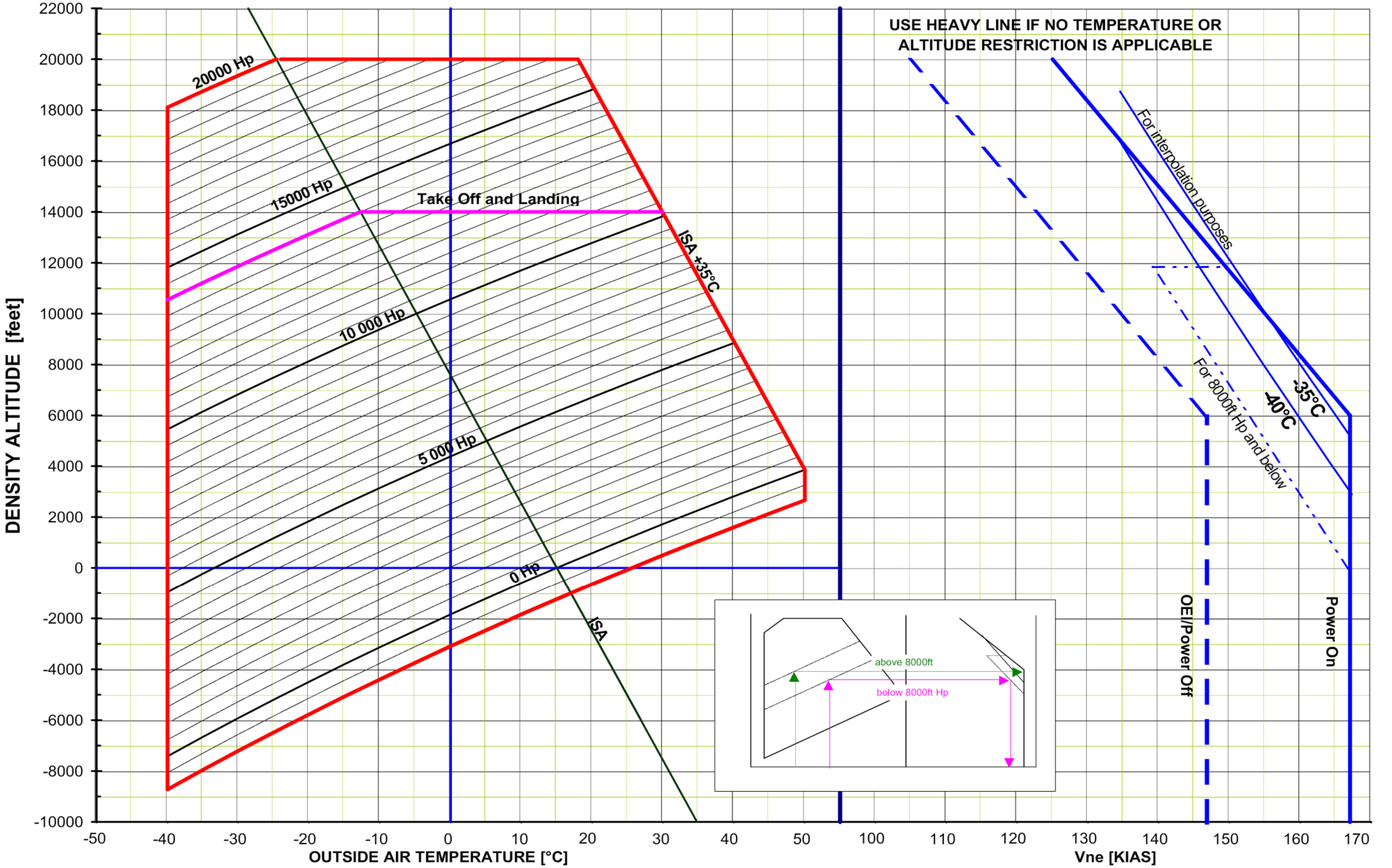
Flight into known icing conditions or freezing rain is prohibited.

A placard on the instrument panel reminds the pilot about this limitation in addition to listing the approved types of operation.

WEIGHT LIMITATIONS

The maximum gross weight for takeoff and landing, CAT A or CAT B, is 6400 kg.

It can be increased up to 6800 kg with the installation of kit P/N 4G0000F00111 (see RFM Supplement 50).



139G0290T006/2 Issue C

ICN-39-A-151000-A-A0126-00003-A-08-1

NOTE: For reference only. Refer to latest issue of RFM for applicable data.

AIRSPEED ENVELOPE (V_{NE} – Power ON, OEI/Power OFF)

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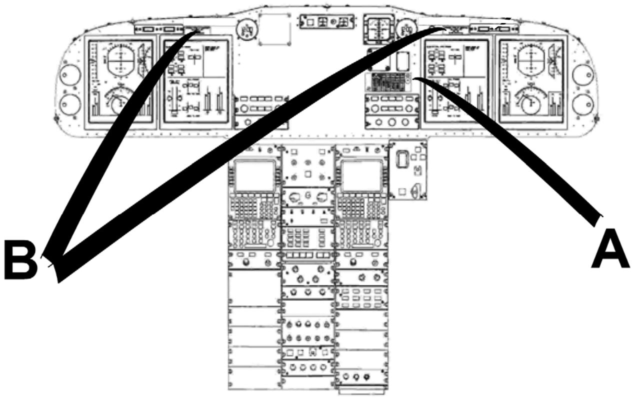


VIEW OF COCKPIT WITH PLACARD POSITIONS

A

VNE POWER ON AIRSPEED LIMITATION KIAS						
Hp ft / OAT °C	-1000 to SL	4000	8000	10000	15000	20000
50	160	-	-	-	-	-
20	167	157	145	147	128	-
10	167	159	147	151	133	-
0	167	162	150	156	136	-
-10	167	164	154	159	139	-
-25 AND BELOW	167	167	158	165	144	125
VNE OEI/PWR OFF=VNE PWR ON LESS 20KIAS					V_{LE}/V_{LO}=150KIAS	

B

APPROVED TYPES OF OPERATION

DAY/NIGHT VFR
DAY/NIGHT IFR

ICING PROHIBITED

PLACARDS – AIRSPEED LIMITATION AND TYPES OF OPERATION

MAJOR ZONES

The major zones of the helicopter are named as per the figure.

STRUCTURE

The primary structure comprises the fuselage forward section and the tail section (tail boom).

The secondary structure, include:

- cockpit doors
- cabin doors
- baggage doors
- inspection doors
- transparent panels
- cowlings
- fairings
- radomes
- strake

The forward section includes the nose compartment, the cockpit, the cabin and the aft fuselage, which includes the baggage compartment. The top side of the forward section is the upper deck floor.
The tail-boom is attached to back of the fuselage and can be removed. The tail-boom is integral with the vertical fin and supports the tail rotor, the tail rotor drive train and the horizontal stabilizer.

COCKPIT AND CABIN

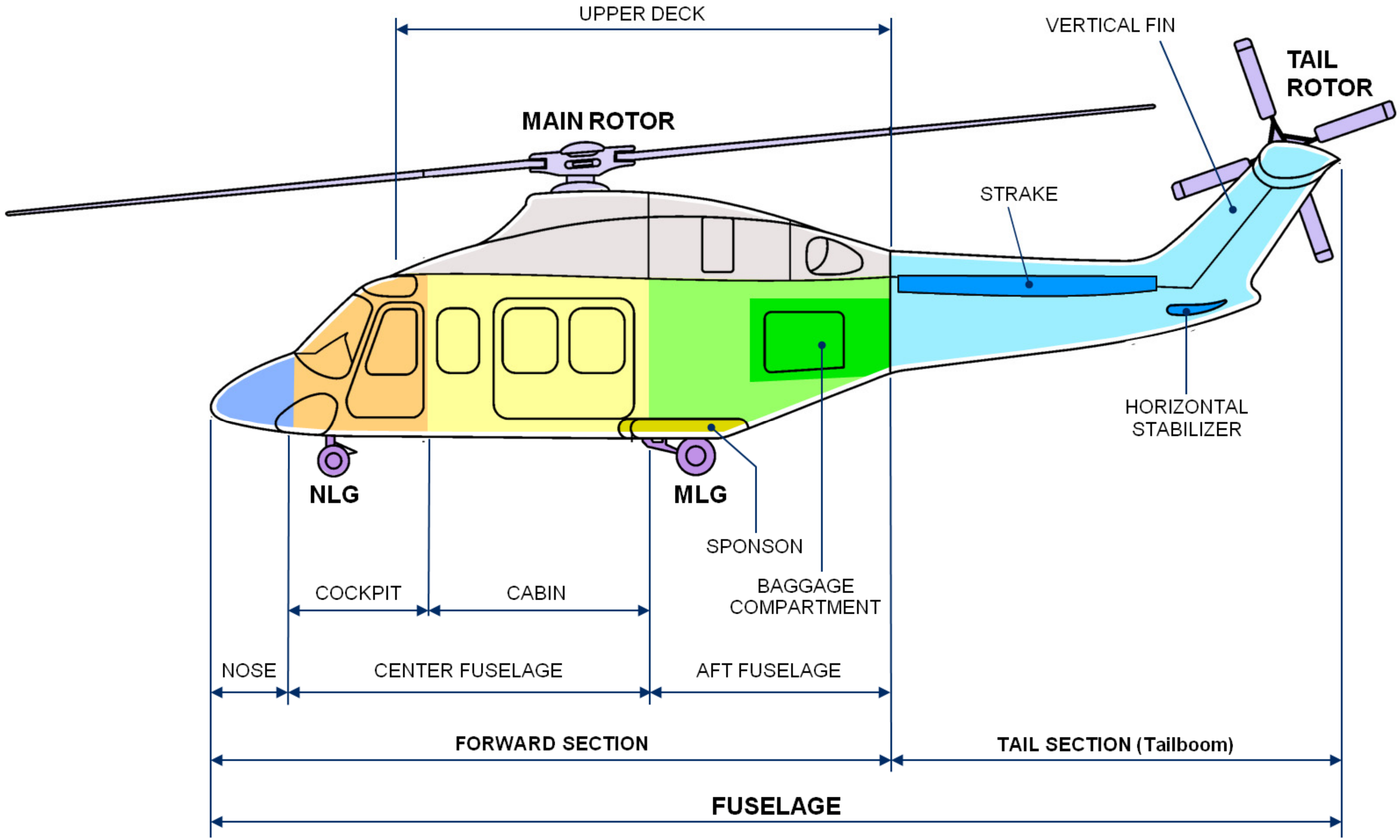
The cockpit includes the pilot (RH) and co-pilot (LH) seats. The seats, installed side by side, are crash-worthy, adjustable in position and provided with 4-point seat belts with quick-disconnect ring and mechanical reeling.

Two forward opening hinged-doors give access to the cockpit on both sides. Large glass windshields and clear acrylic windows in the crew doors, lower nose, and roof area give the crew good visibility.

The cabin includes the passenger seats and the baggage-bay area at the back of the fuselage. A full size plug-in sliding door is located on each side of the cabin area. Each sliding door has two acrylic-plastic windows and can be secured in the fully open or fully closed position.

The seating arrangements can be changed depending on the configuration:

- Standard civil transport ....12 passengers
- High density civil transport ....15 passengers
- Civil EMS with many possible arrangements of litters, medical attendant seats, hoist operator seat, medical equipment racks



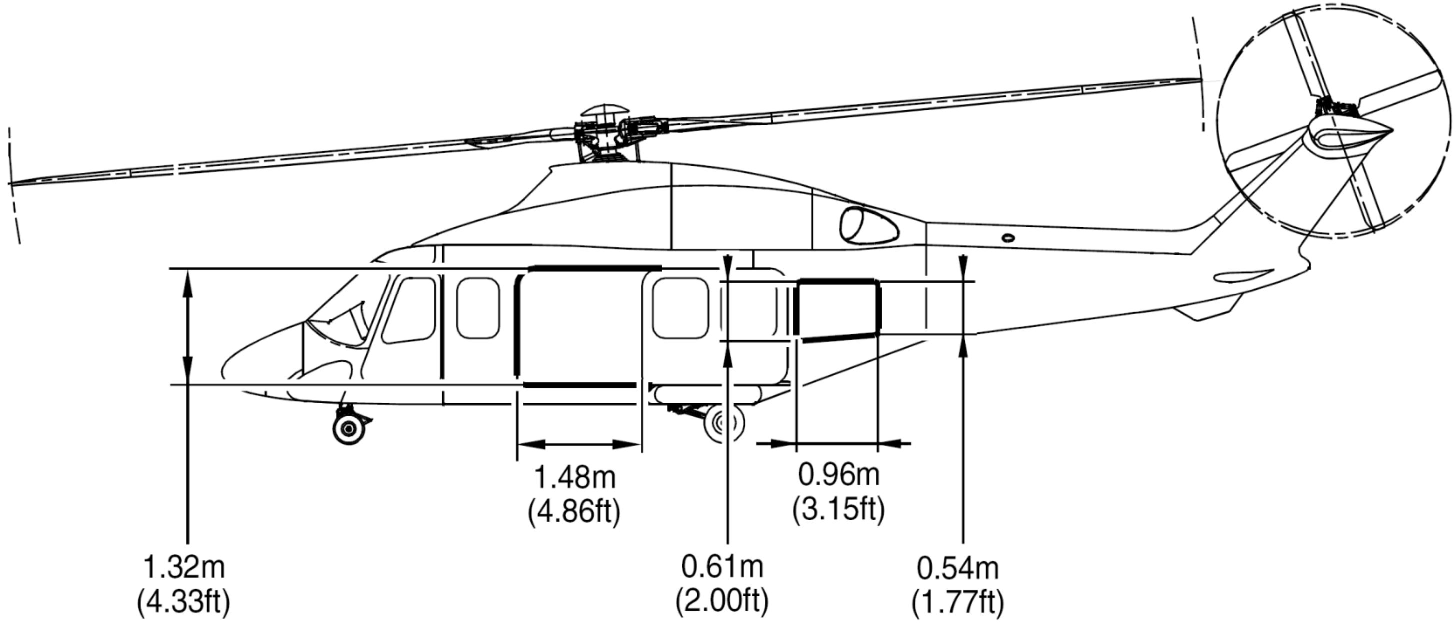
AW139 HELICOPTER — MAJOR ZONES

CABIN SLIDING DOOR

NOTE:
Same arrangement
on both sides.

PILOT DOOR

ACCESS TO COCKPIT AND CABIN



CABIN AND BAGGAGE DOOR DIMENSIONS

EMERGENCY EXITS

Four emergency exits per side are provided by jettisoning the following windows:

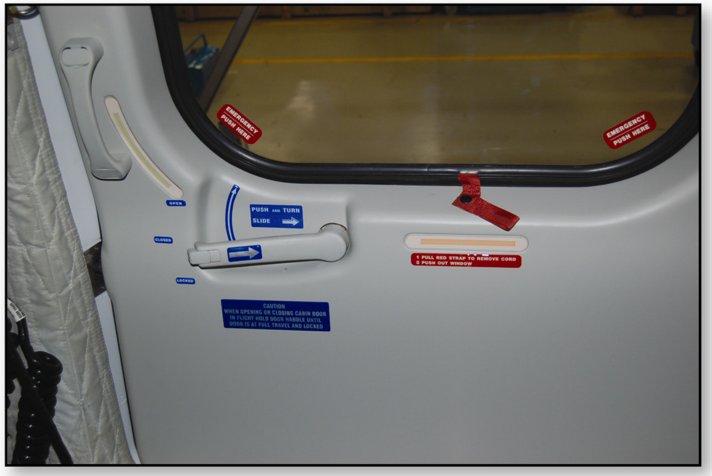
- Cockpit Door Window (1 per side)
- Cabin Window (1 per side, on the fixed frame)
- Cabin Door Windows (2 per side)

A red strap in the internal bottom side of each emergency exit window permits removal of the rubber cord that holds the window transparent panel in place. The emergency exit is made free by pushing the transparent panel out acting on its bottom left corners.

OPTIONAL EQUIPMENT

A wide selection of optional kits can be installed on the AW139 helicopter. Optional kits include:

- Rotor Brake
- Air Conditioning System
- 4-Axis Flight Director
- Engine Air Particle Separator (EAPS)
- HF Radio
- V/UHF Radio
- Weather Radar
- TCAS
- EGPWS
- Emergency Floats and Liferafts (Ditching configuration)
- Deployable ELT
- Hoist
- Cargo Hook
- Auxiliary Fuel Tank
- External Public Address System (External Loudspeakers)
- Second Radar Altimeter
- Snow Skids
- Slump Pads (soft terrain pads)
- traffic advisory system KTA 970
- hinged passenger doors
- sky force observer Mark II digital map system
- EURONAV IV digital map system
- SX-16 nightsun searchlight
- operations with cockpit doors removed
- increased baggage compartment load
- cabin bubble window
- 4-axis Flight Director



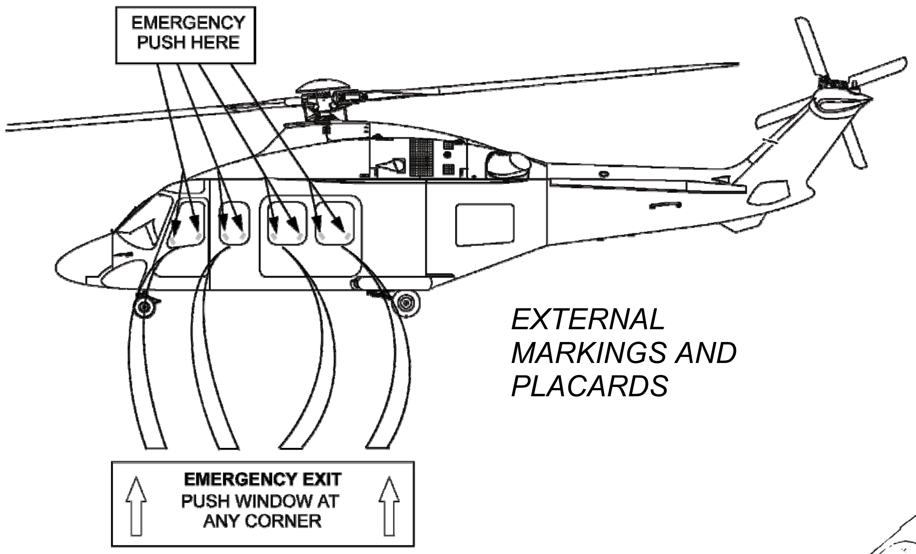
CABIN DOOR

COCKPIT DOOR

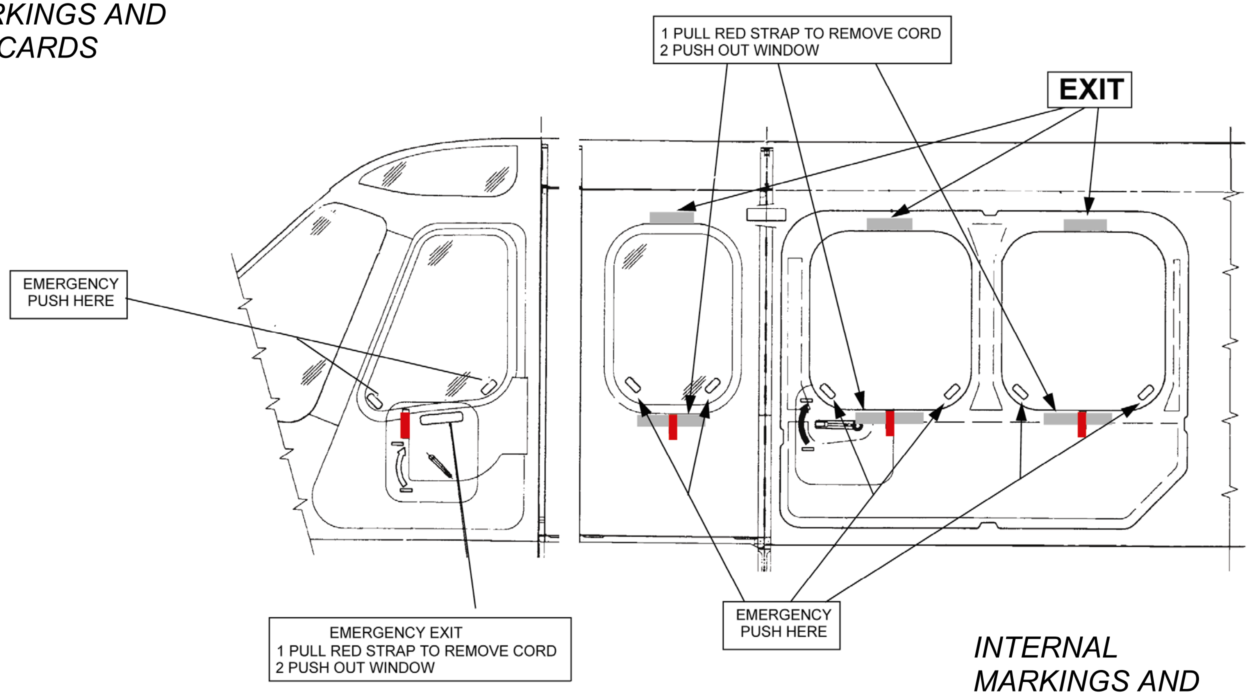
CABIN WINDOW

CABIN WINDOW

EMERGENCY EXITS

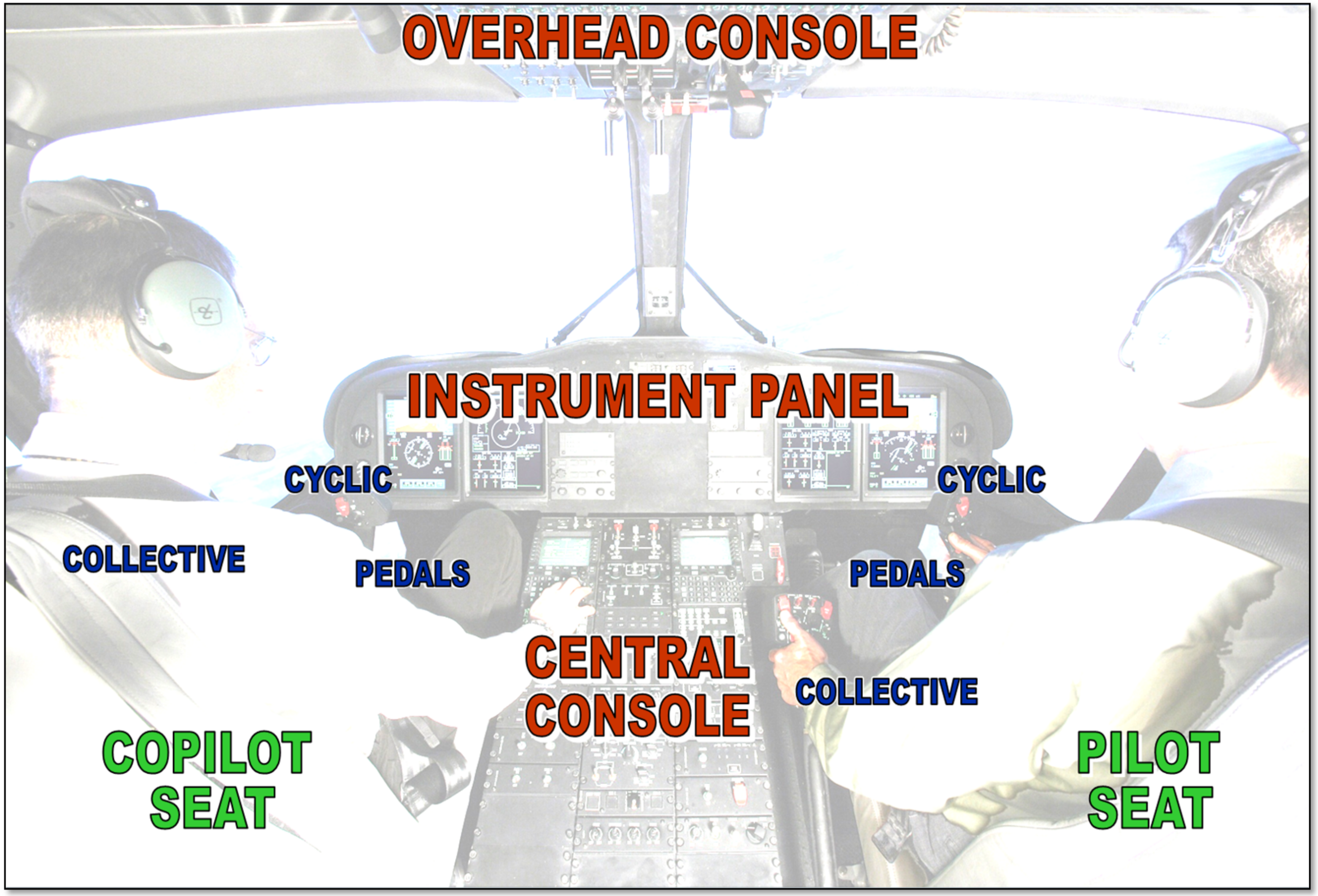


EXTERNAL MARKINGS AND PLACARDS

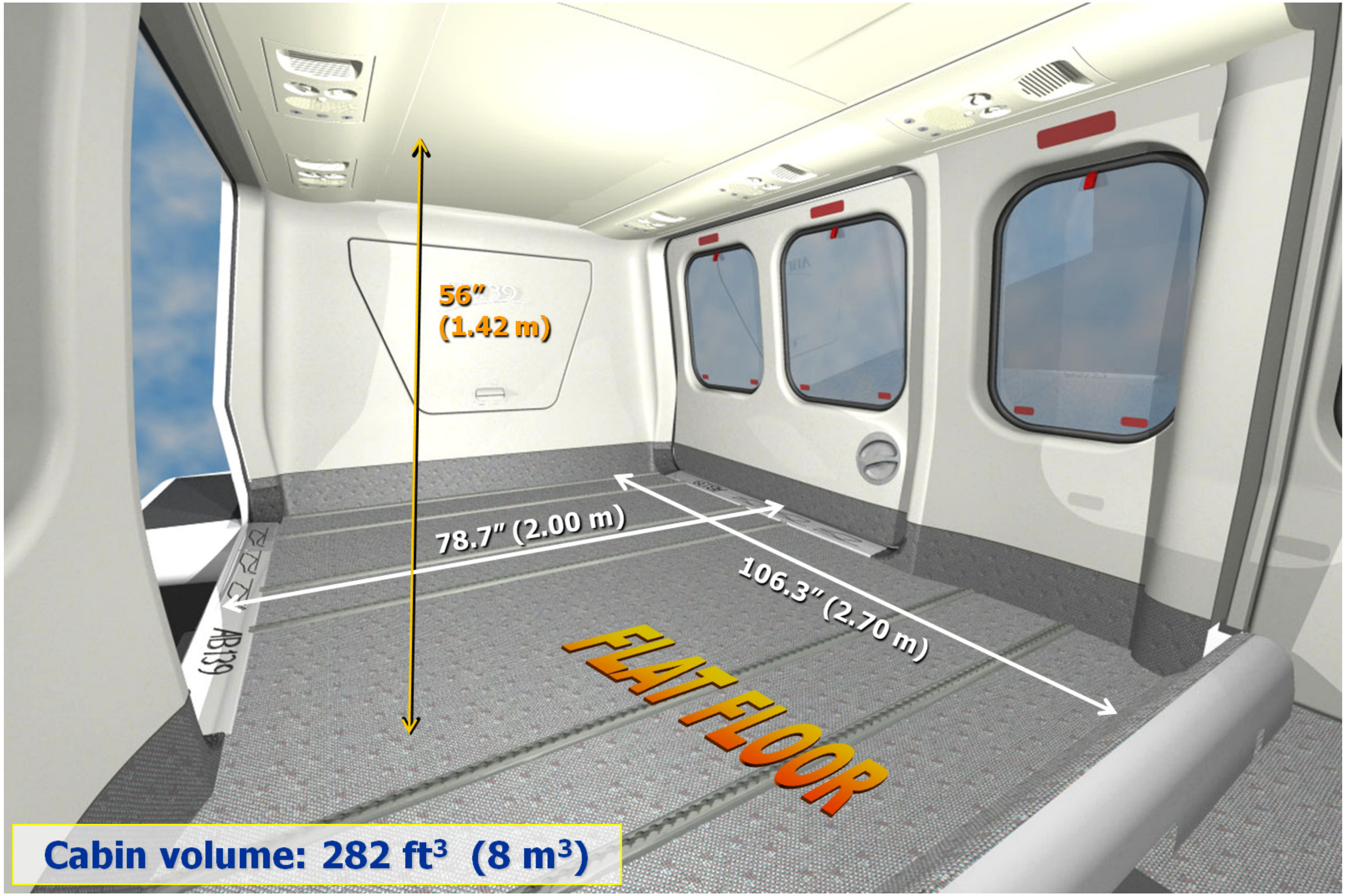


INTERNAL MARKINGS AND PLACARDS

EMERGENCY EXITS — PLACARDS



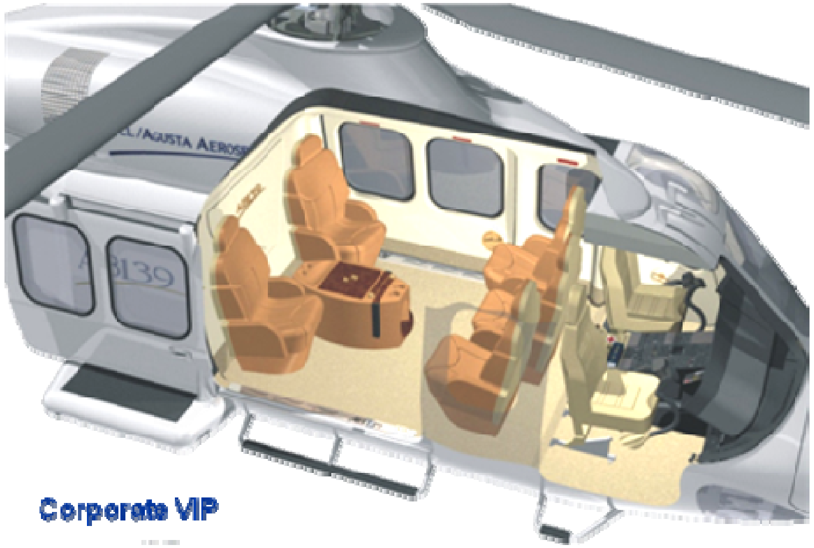
COCKPIT GENERAL ARRANGEMENT

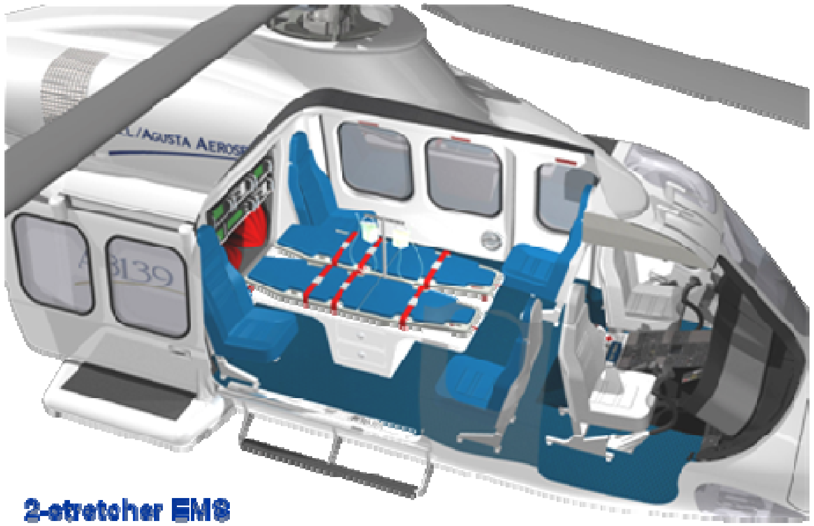


CABIN – INTERNAL DIMENSIONS


11 passegers


12 passegers


Corporate VIP


2-stretcher EMS

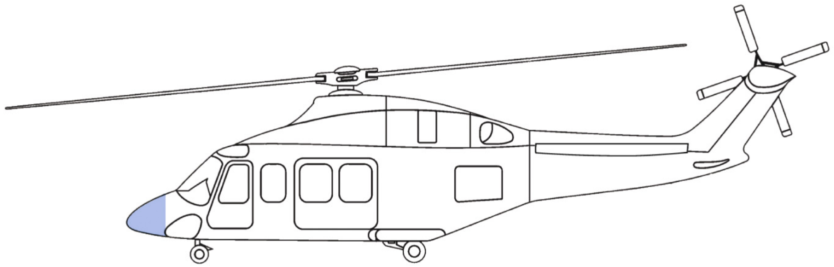
CABIN CONFIGURATION

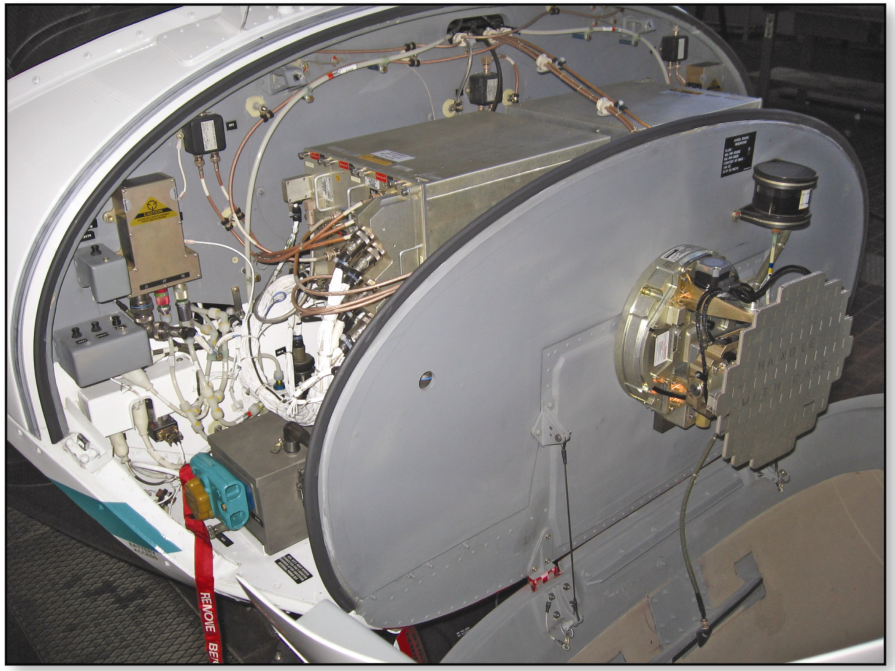
LOW DENSITY CONFIGURATION

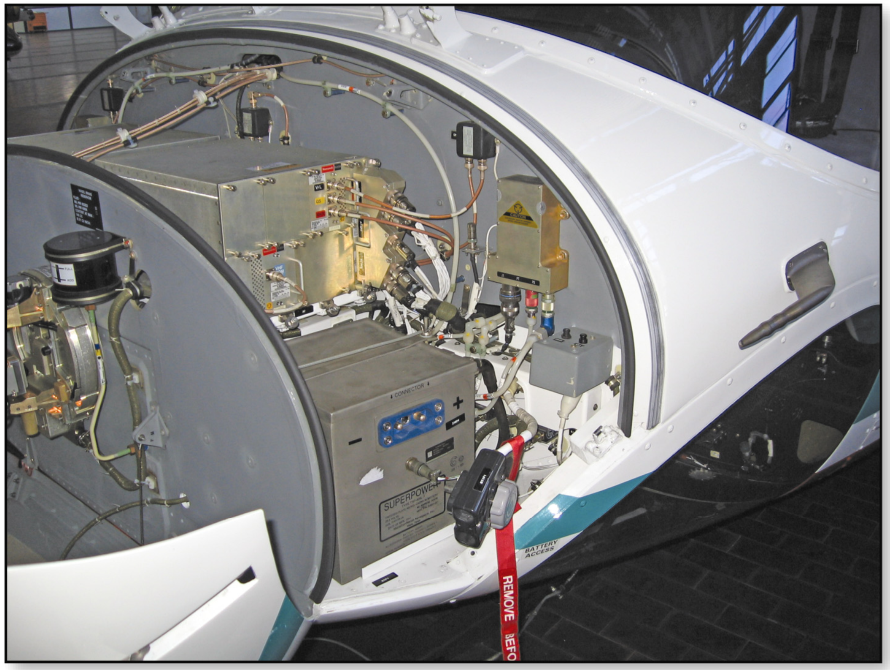
HIGH DENSITY CONFIGURATION

EMERGENCY MEDICAL SERVICE (EMS) CONFIGURATION

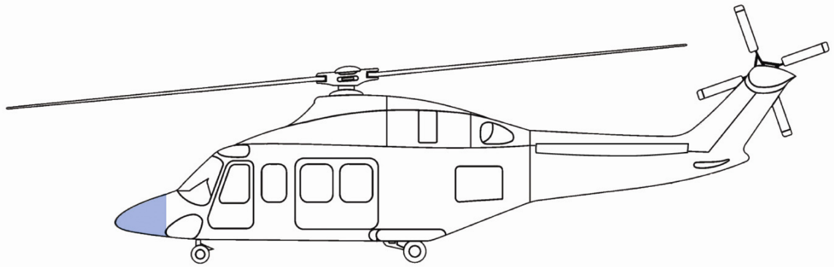
VIP/CORPORATE CONFIGURATION



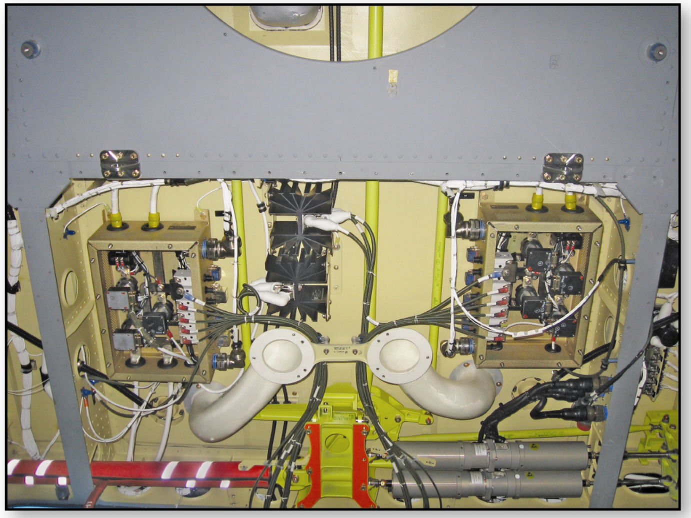




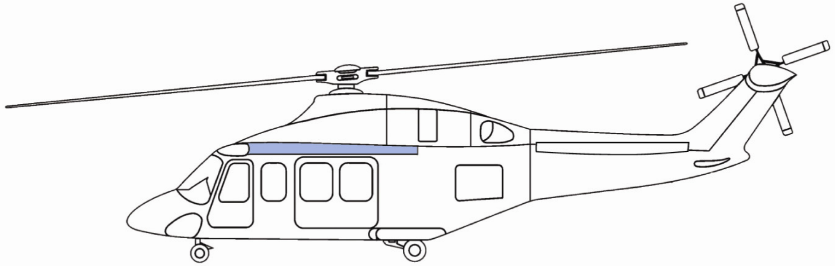
ELECTRICAL / AVIONIC BAY – SHORT NOSE CONFIGURATION



ELECTRICAL / AVIONIC BAY – LONG NOSE CONFIGURATION

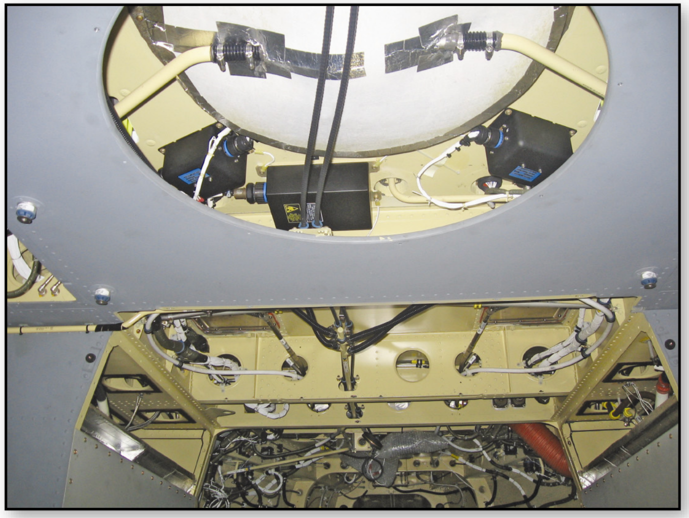


FORWARD AREA

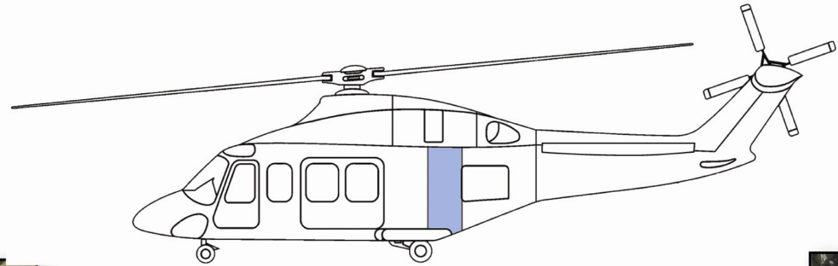


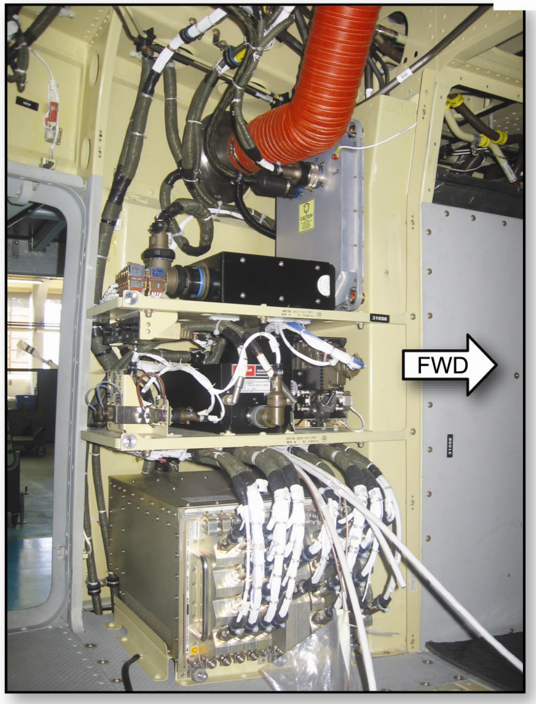
CENTRAL AREA

AFT AREA

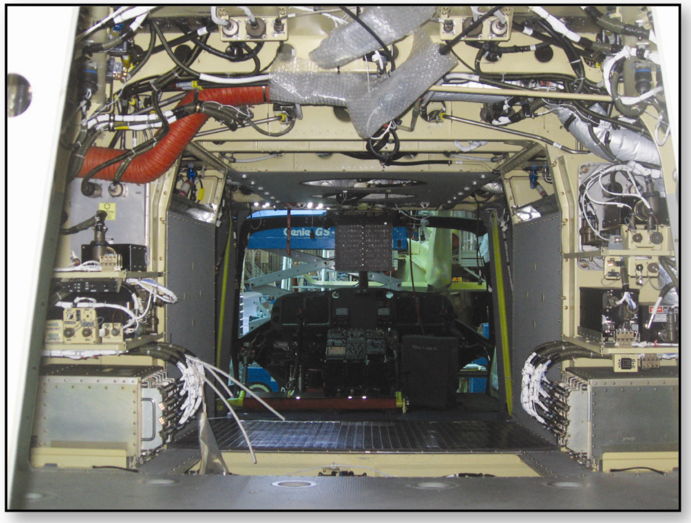


ELECTRICAL / AVIONIC BAY – CABIN CEILING CONFIGURATION

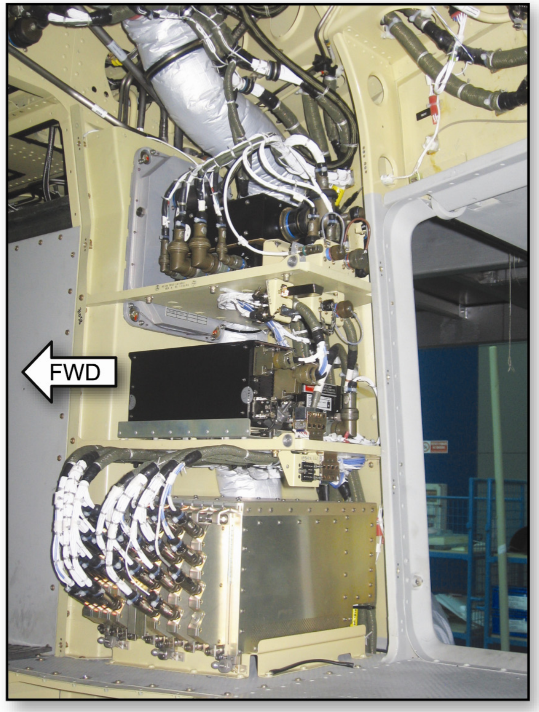




LH AVIONICS RACK

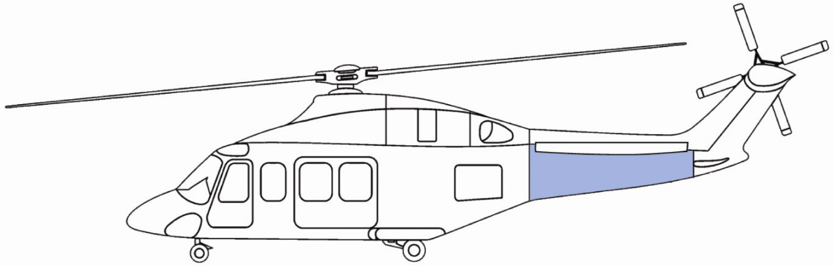


VIEW LOOKING FORWARD
FROM BAGGAGE COMPARTMENT



RH AVIONICS RACK

ELECTRICAL / AVIONIC RACKS CONFIGURATION



VIEW LOOKING AFT FROM BAGGAGE COMPARTMENT

ELECTRICAL /AVIONIC BAY TAILBOOM CONFIGURATION

SYSTEMS OVERVIEW

POWER PLANT

The power plant comprises the engines and related installation, fire detection and extinguishing system.

Engines

The helicopter is powered by two PT6C-67C turboshaft engines. Each engine is installed in a separated fireproof area above the cabin roof and supplies power to the drive system by means of a rotating shaft. The engines are connected to the airframe by means of two attachment points on the engine body and to the main gearbox by means of a tube and a gimbal joint.
Air is supplied to the engine via individual, side facing air inlets. The engines are started by a DC starter-generator.
Engine control is achieved via a control panel located in the cockpit and manual back-up of the engine control via push-pull cables.
The engines are provided with torque sensing and matching.

Fire detection and extinguishing

The fire detection system consists of a continuous wire detector installed in the powerplant fire zones, routed in a way that allows coverage of all critical areas.
The fire extinguishing system consists of directional flow valves which allow discharging the contents of one bottle while sealing the connection to the other bottle and the subsequent discharge of the second bottle in the same bay if required.

DRIVE SYSTEM

The drive system consists of the Main Rotor Drive System and the Tail Rotor Drive System.

Main Rotor Drive System

The Main Rotor Drive System mainly consists of the Main Gearbox (MGB) that is mounted on the roof of the cabin by means of four struts and an anti-torque device.
The MGB has three stages of reduction and includes a duplicated oil lubrication system. It provides the attachment points for the rotor brake coaxial with the tail rotor drive output.
The MGB drives three hydraulic pumps and other accessories.

Tail Rotor Drive System

The tail rotor drive system consists of three drive shaft driven by the MGB, the Intermediate Gearbox IGB and the Tail Gearbox TGB oil splash lubricated.

ROTORS

The rotor system consists of a Main Rotor (MR) and a tail Rotor (TR).
The main rotor is a five blades, fully articulated rotor.
The tail rotor is a four blades, fully articulated rotor.

HYDRAULIC POWER

The hydraulic power system supplies the hydraulic power necessary to operate the flight control servo-actuators and the landing gear. The hydraulic power system includes two independent circuits that supply hydraulic fluid at a nominal working pressure of 3000 psi (207 bar).
Each one of the two circuits is able to provide hydraulic power to flight controls servo-actuators.
For ground test and malfunction condition, a shut-off valve allows to shut-off the flight controls circuits.
For safety reason, another shut-off valve allows to shut the landing gear circuits off, increasing the survivability of the flight control functions. Hydraulic power to the landing gear actuators is used to extend and retract the main and the nose landing gears.
The main and nose landing gear are maintained in UP position by the hydraulic pressure (no mechanical uplocks are provided). The extended DOWN position is maintained with a mechanical locks in the main and nose gear actuators.

LANDING GEAR

The AW139 helicopter is provided with a fully retractable, nose tricycle landing gear equipped with shock absorbers.
Hydraulic actuators provide for extension and retraction of the landing gear. The operation is electrically controlled and hydraulically operated, both in normal and emergency conditions.
Two main landing gear (MLG) legs are installed in the sponsons: each MLG leg is provided with a single wheel and weight-on-wheel (WOW) microswitches.
An independent disk type braking system permits differential braking of the main wheels.
The nose landing gear (NLG) leg is free to rotate 360° when on the ground. An automatic mechanical centering device re-centers the nosewheel at lift off if it is within ±115° from forward direction.
The nosewheel can also be held centered by an electro-mechanic centering system.

FUEL

The fuel system includes crashworthy fuel tanks located in the rear area of the cabin. Each tank contains a booster pump, an engine feed line, a fuel and water drain valve. Filling with fuel is achieved by gravity.
The fuel selector manifold allows fuel to supply each engine separately or fuel both engines in a cross-feed condition.
The fuel quantity gauging system is composed by four capacity probes, a Fuel Computer Unit (FCU) and a fuel low level sensor for each tank. The fuel venting system consists of pipes for each tank designed to prevent fuel leakage in case of helicopter roll-over after crash landing.

ELECTRICAL POWER

DC power is generated by two 30V, 300A DC air cooled generators. Two batteries provides a back-up source of emergency power in the event that both generators fail and power for autonomous ground operations and engine starting.
The electrical power is delivered to aircraft systems by a dual system of distribution bus bars consisting of the Main (MAIN),

Essential (ESS) and Non-Essential (NON ESS) busses. Power from a DC external power source can also be connected to the aircraft busses.

LIGHTING

The lighting system includes interior and exterior lights.
The interior lights supply instruments lighting, panels lighting, overhead panels lighting and cockpit utility lighting.
The exterior lights include anti collision lights, position lights, navigation lights and landing lights.
NVG compatibility is provided as an option.

COCKPIT AND CABIN VENTILATION SYSTEM

The ventilation system consists of two separate sub-systems for cockpit and cabin ventilation; heating and cooling systems are provided as optional kits.

AVIONICS

The PRIMUS EPIC® system is an integrated avionics system that includes the following sub-systems necessary to operate:

- Auto-Pilot
- Flight Management System
- Communications
- Indicating and Recording Systems
- Aural Warning Generator
- Navigation
- Crew Alerting System
- Central Maintenance Systems (CMS)

The PRIMUS EPIC® system is integrated into:
- two Modular Avionics Units (MAU)
- four flat panel color LCD Display Units (DU) to show data in the cockpit
- two Modular Radio Cabinets (MRC) that include the following radios:
    - VHF-COMM
    - VOR/ILS
    - ADF
    - DME
    - Transponder (XPDR)

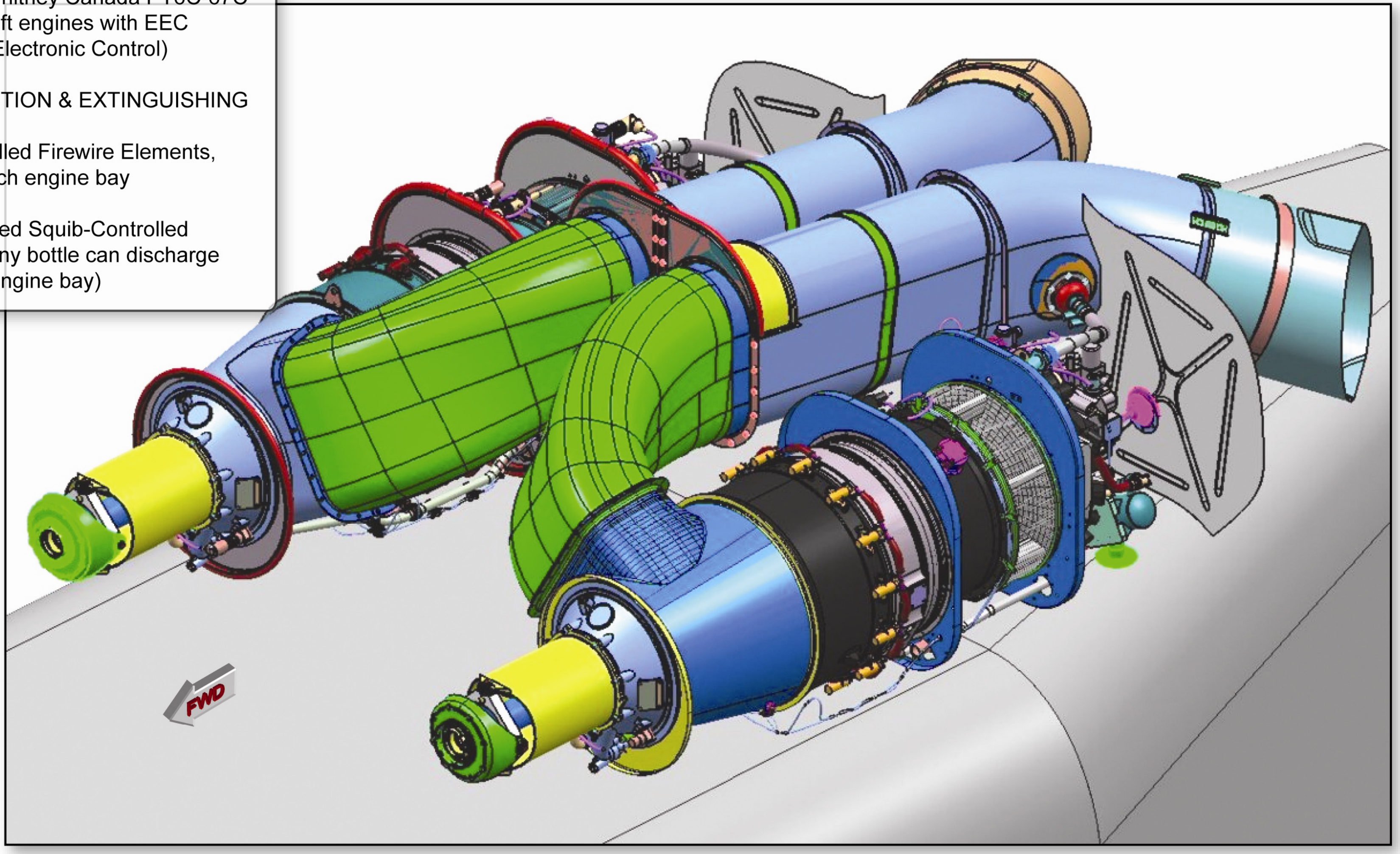
The MAUs, the DUs and the MRCs are directly connected to each other via a bi-directional digital data bus named Avionic Standard Communication Bus-D (ASCB-D).
A LAN digital bus also interconnects the same units for maintenance purposes.

2 × Pratt & Whitney Canada PT6C-67C Turboshaft engines with EEC (Engine Electronic Control)

FIRE DETECTION & EXTINGUISHING

2 x Helium Filled Firewire Elements, one in each engine bay

2 x Halon-Filled Squib-Controlled Bottles (any bottle can discharge into any engine bay)



POWER PLANT

MAIN ROTOR

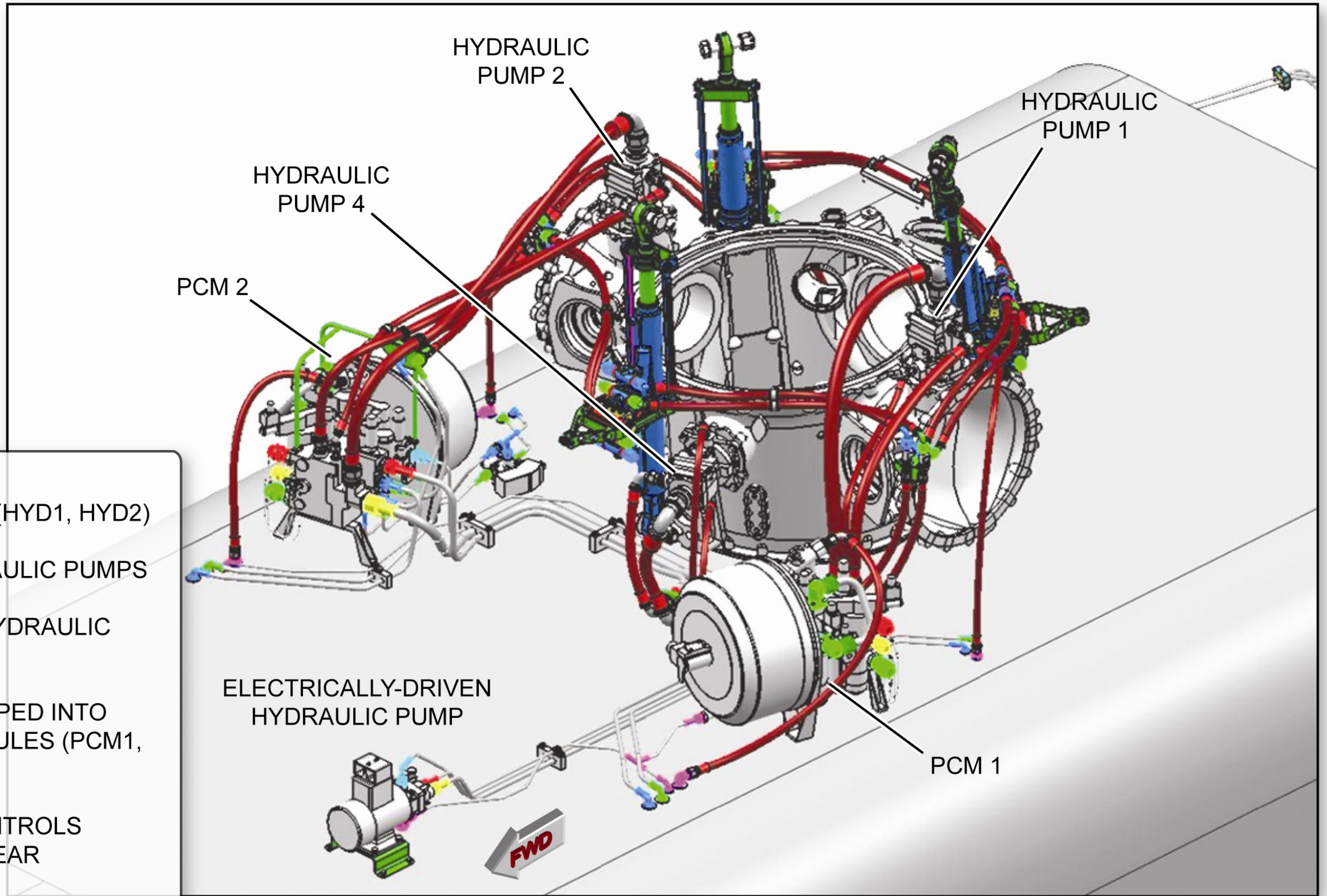
MAIN ROTOR DRIVE SYSTEM

TAIL ROTOR

TAIL ROTOR DRIVE SYSTEM

ROTORS AND ROTOR DRIVE SYSTEMS

FLIGHT CONTROLS



2 × INDEPENDENT SYSTEMS (HYD1, HYD2)

3 × GEARBOX-DRIVEN HYDRAULIC PUMPS

1 × ELECTRICALLY-DRIVEN HYDRAULIC PUMP

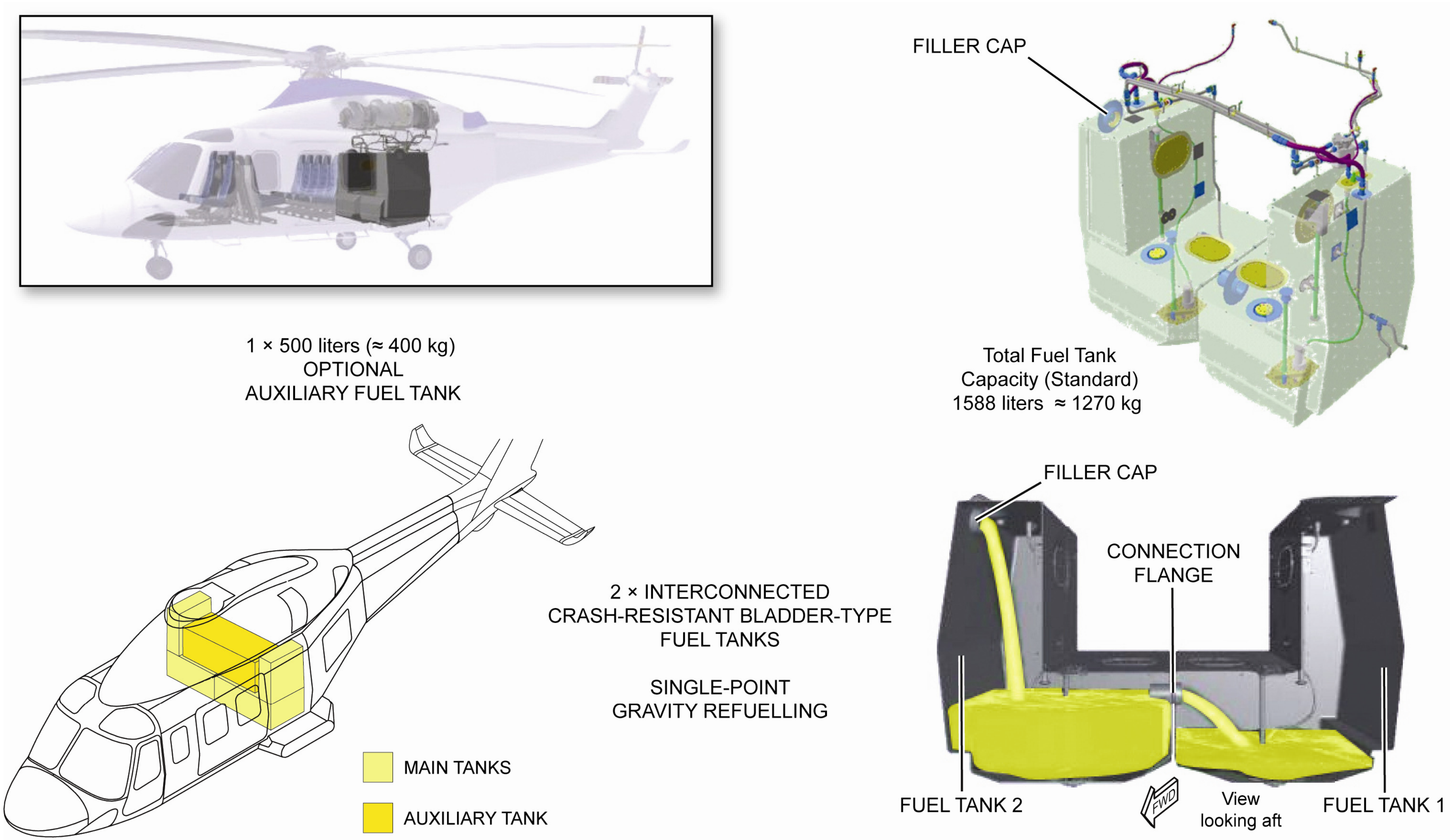
MAJOR COMPONENTS GROUPED INTO TWO POWER CONTROL MODULES (PCM1, PCM2)

Supplies power to: FLIGHT CONTROLS
LANDING GEAR

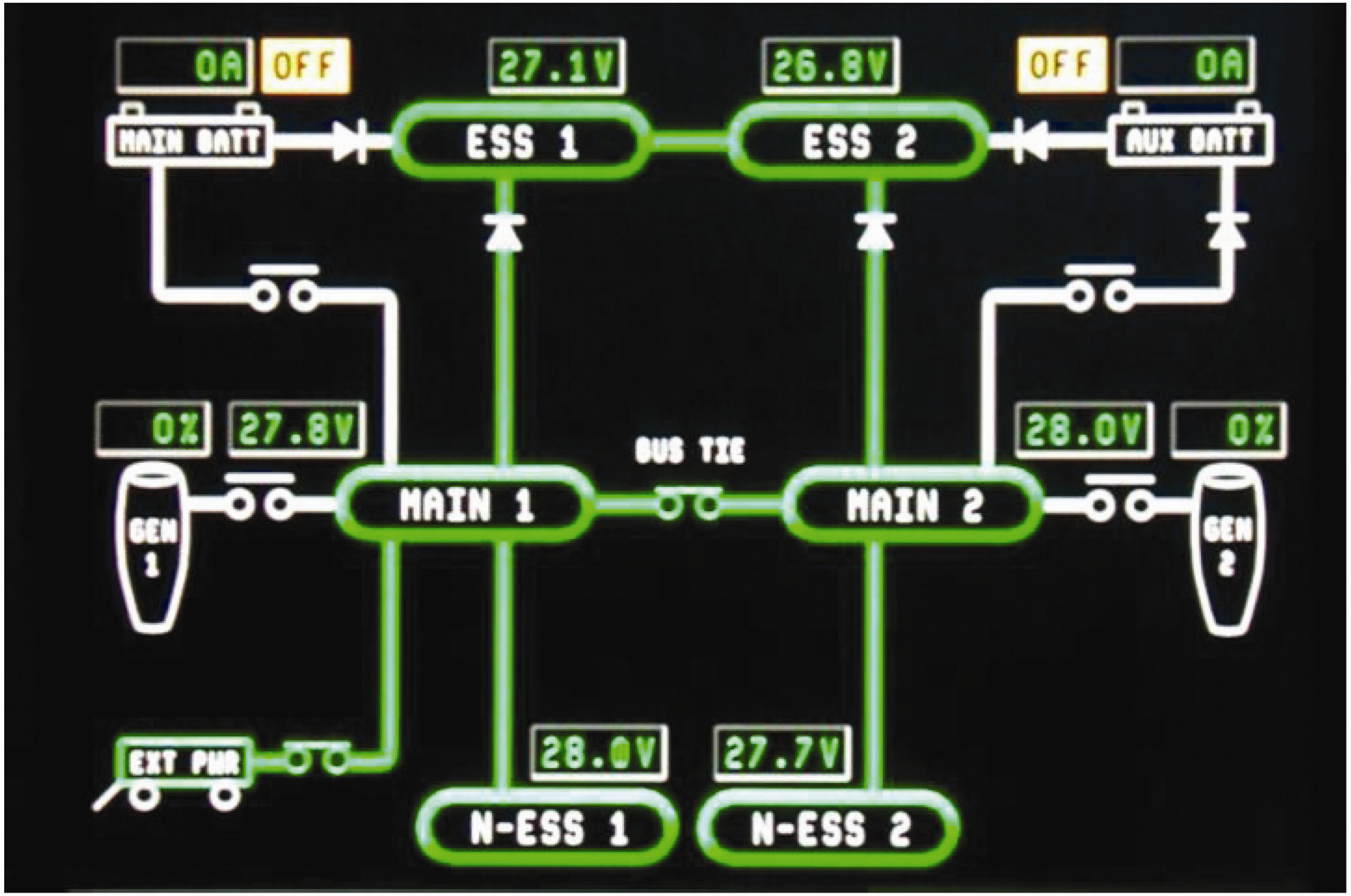
Operating Pressure: 207 bar (3000 PSI)

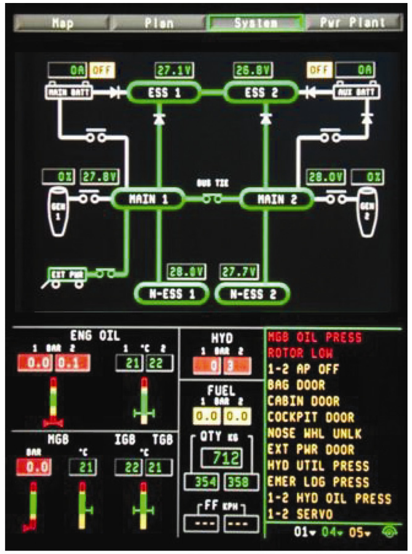
HYDRAULIC POWER

LANDING GEAR


FILLER CAP
1 × 500 liters (≈ 400 kg)
OPTIONAL
AUXILIARY FUEL TANK
Total Fuel Tank
Capacity (Standard)
1588 liters ≈ 1270 kg
FILLER CAP
CONNECTION
FLANGE
2 × INTERCONNECTED
CRASH-RESISTANT BLADDER-TYPE
FUEL TANKS
SINGLE-POINT
GRAVITY REFUELLING
MAIN TANKS
AUXILIARY TANK
FUEL TANK 2
FWD
View
looking aft
FUEL TANK 1

FUEL





ELECTRICAL POWER

INTERIOR LIGHTS

COCKPIT
INSTRUMENT, CONSOLE, OVERHEAD
2 x UTILITY
2 x ANTI STORM
1 x DOME

CABIN
6 x FLUORESCENT
12 x READING
2 x EMERGENCY DOME

BAGGAGE
3 x DOME

ANTI COLLISION
1 x TAIL TOP

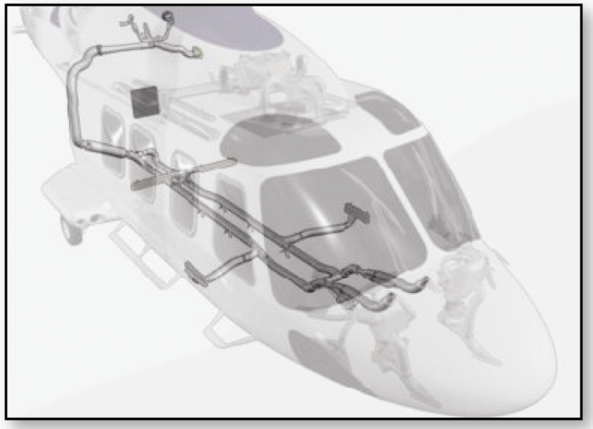
NAVIGATION
2 x SIDE
1 x TAIL

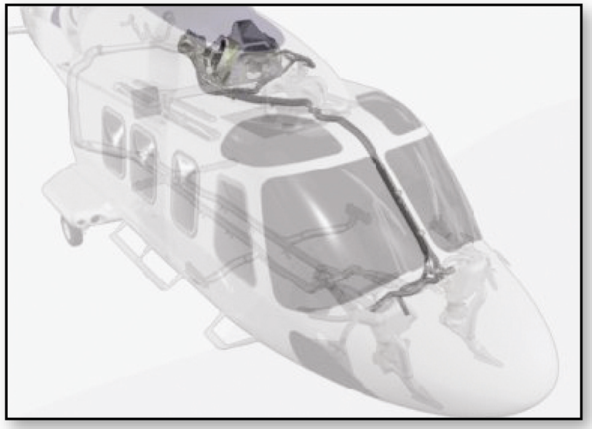
LANDING
2 x LANDING (Fixed)
1 x SECONDARY LANDING (Searchlight)

EMERGENCY EGRESS
2 x SIDE (Fixed)

EXTERIOR LIGHTS

LIGHTING





VENTILATION

THREE SEPARATE SYSTEMS:
• PILOT VENTILATION
• COPILOT VE NTILATION
• CABIN VENTILATION

DYNAMIC OR FORCED AIR VENTILATION

HEATING

SINGLE SYSTEM SUPPLIED BY BOTH ENGINES

AUTOMATIC TEMPERATURE CONTROL

AIR CONDITIONING (Optional)

TWO SEPARATE VAPOUR-CYCLE SYSTEMS:
• COCKPIT AIR CONDITIONING
• CABIN AIR CONDITIONING

MAIN GEARBOX-DRIVEN COMPRESSORS

ENVIRONMENTAL CONTROL SYSTEM

Honeywell PRIMUS EPIC®

STANDARD INTEGRATED AVIONICS CONFIGURATION

- 4 × DISPLAY UNITS
- 2 × SETS OF DISPLAY CONTROL PANELS
- 2 × MCDU
- 2 × VOR/ILS/MB
- 1 × DME
- 1 × ADF
- 2 × FMS
- 1 × GPS
- 2 × VHF-AM COMM
- 2 × COCKPIT DIGITAL AUDIO CONTROL PANELS
- 1 × CABIN ATTENDANT AUDIO CONTROL PANEL
- 1 × PASSENGER ADDRESS
- 1 × GROUND CREW JACK
- 1 × AWG
- 2 × 3-AXIS DIGITAL AUTOPILOT
- 2 × ADS
- 2 × AHRS
- 1 × RAD ALT
- 1 × ATC TRANSPONDER
- 1 × CMC

OTHER STANDARD AVIONICS SYSTEMS

- 1 × STBY INSTRUMENT
- 1 × ELT
- 1 × FDR/CVR

OPTIONAL AVIONICS SYSTEMS

- 2 × 4-AXIS DIGITAL FLT DIR
- 1 × 2ND DME
- 1 × 2ND ADF
- 1 × 2ND GPS
- 1 × 2ND RAD ALT
- 1 × V/UHF
- 1 × HF
- 1 × SATCOM
- A/R CABIN AUDIO PANELS
- 1 × EXTERNAL LOUDSPEAKERS
- 1 × WX RDR
- 1 × LSS
- 1 × TCAS
- 1 × FLIR
- 1 × EGPWS
- 1 × HUMS

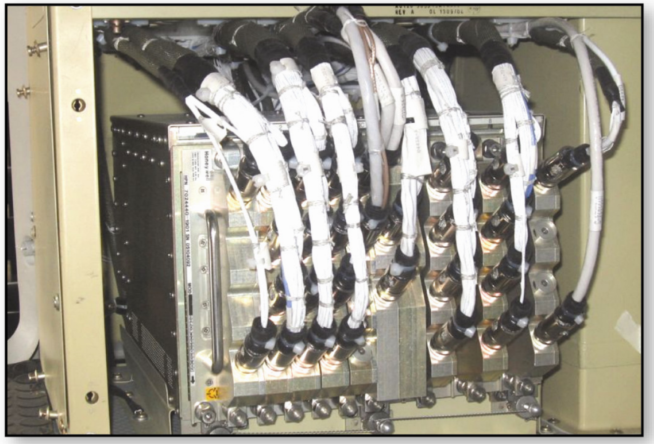
AVIONIC SYSTEM CONFIGURATION

Honeywell PRIMUS EPIC®

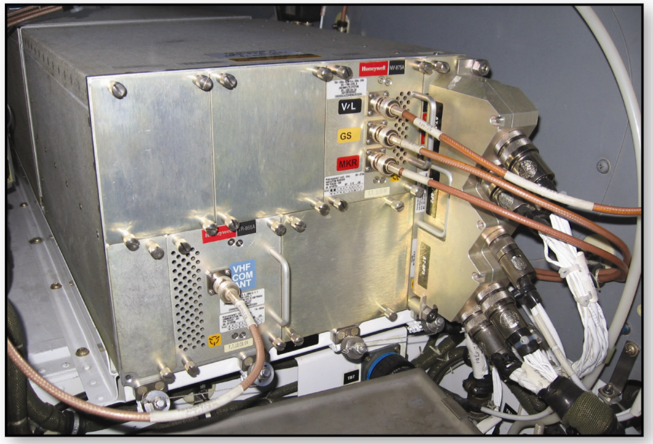
Integrates systems and subsystems for:
- NAVIGATION
- COMMUNICATIONS
- AUTOFLIGHT
- INDICATING/RECORDING
- CENTRAL MAINTENANCE

Can be used as:
- Single-Pilot VFR
- Dual-Pilot IFR

DU (Display Unit)



MAU (Modular Avionic Unit)



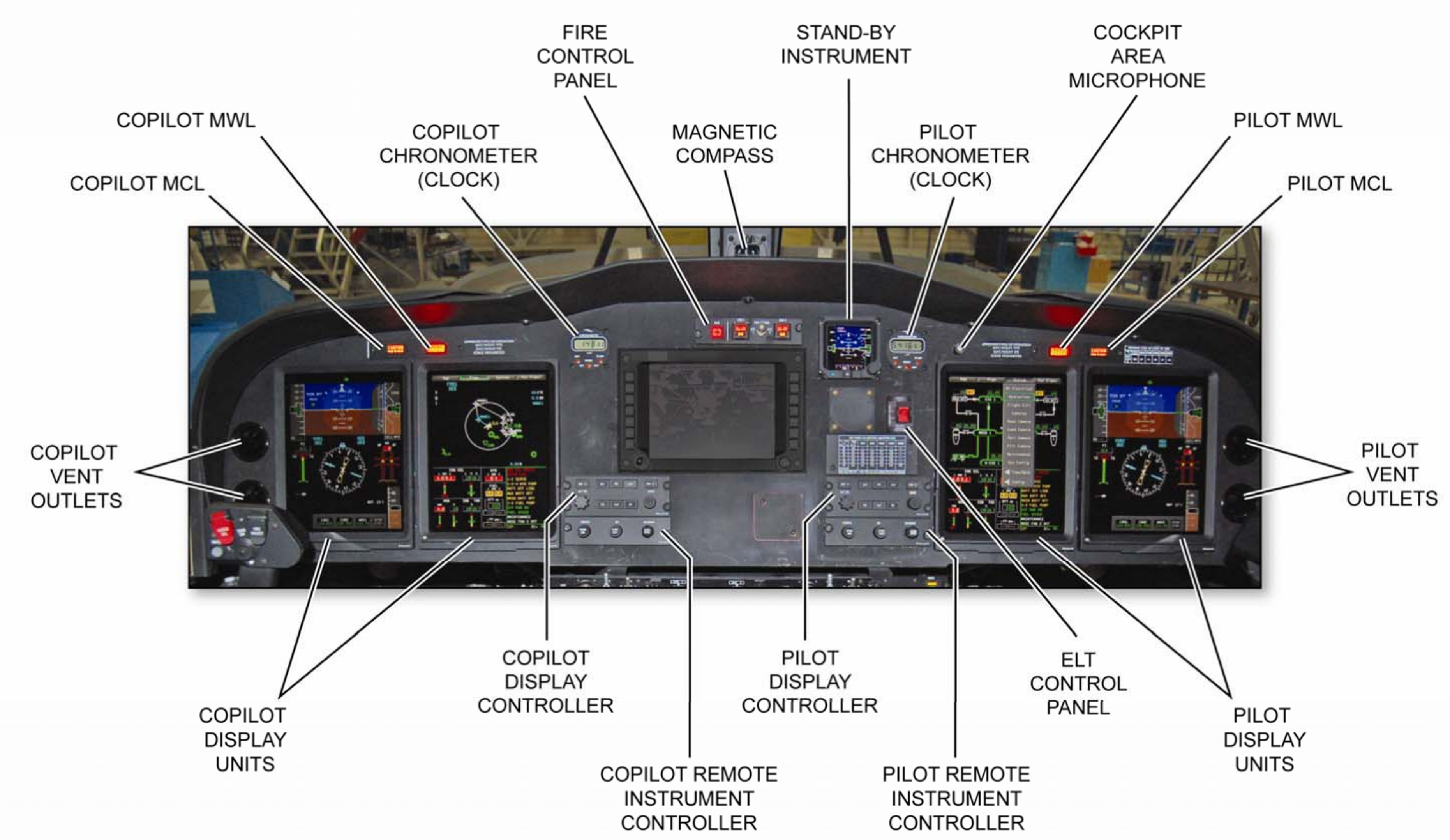
MRC (Modular Radio Cabinet)

AVIONICS SYSTEM - HIGHLIGHTS

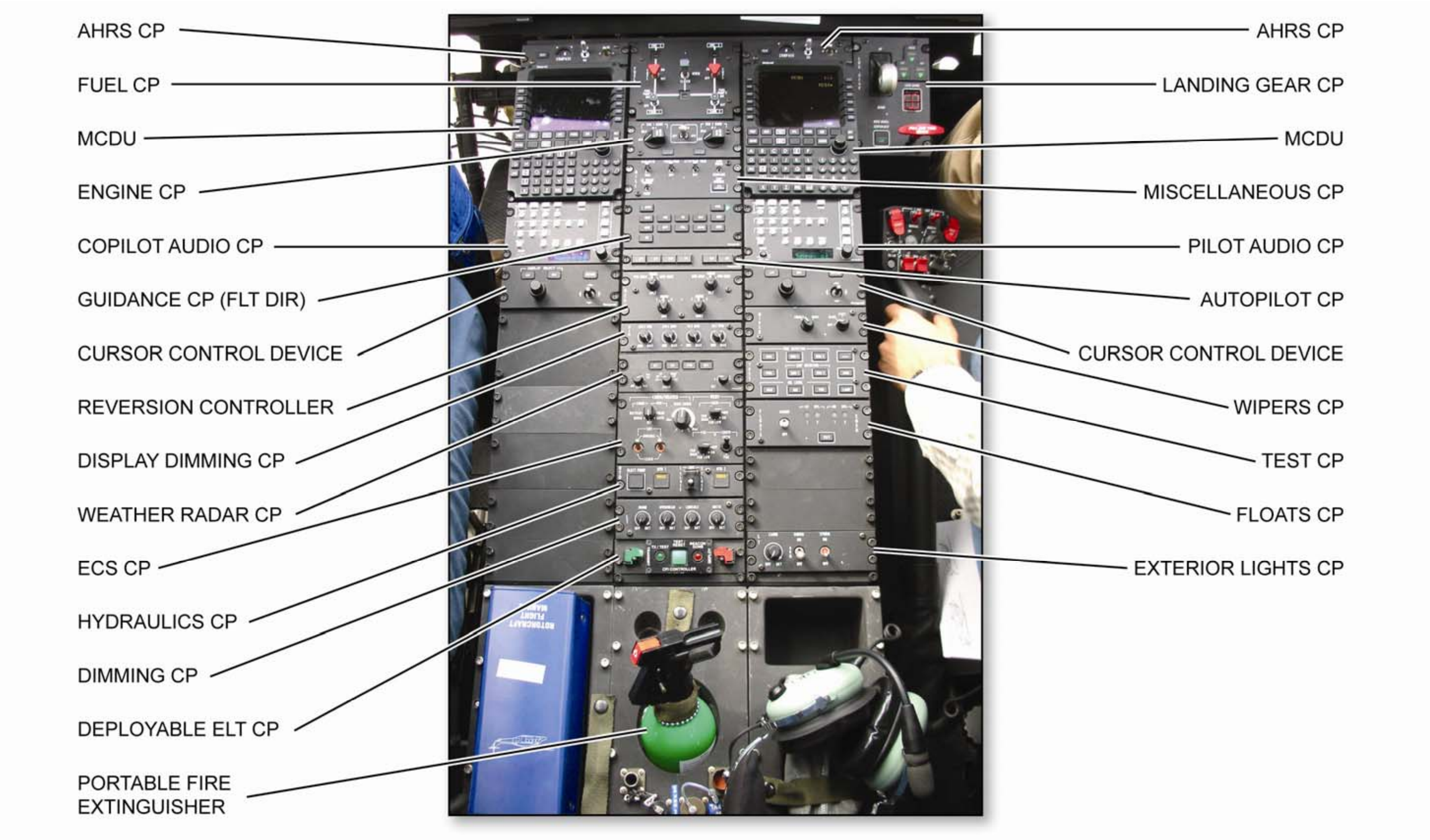
CHAPTER
00
AIR VEHICLE GENERAL

SECTION 03 – COCKPIT LAYOUT

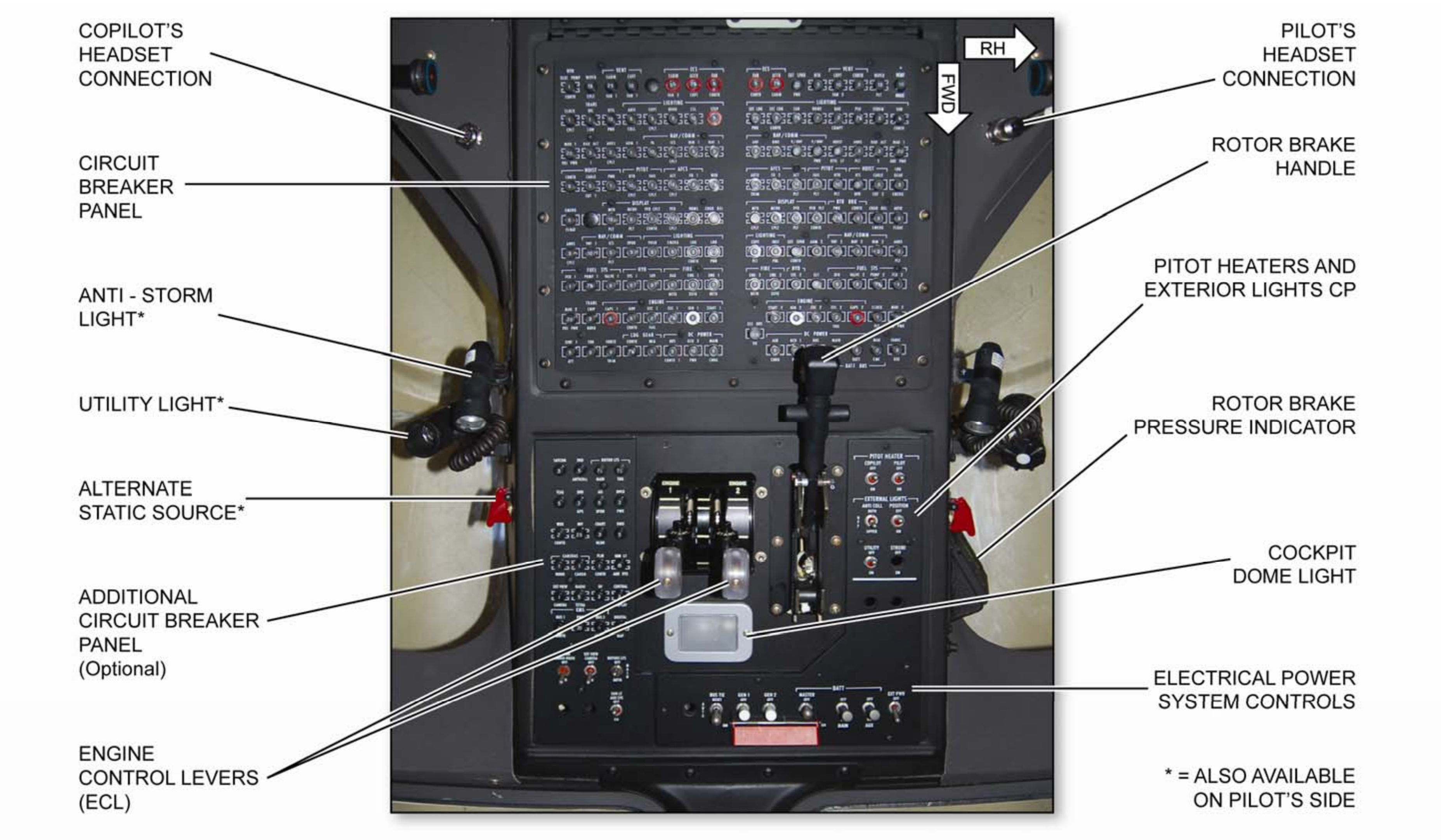
AW139 HELICOPTER – COCKPIT



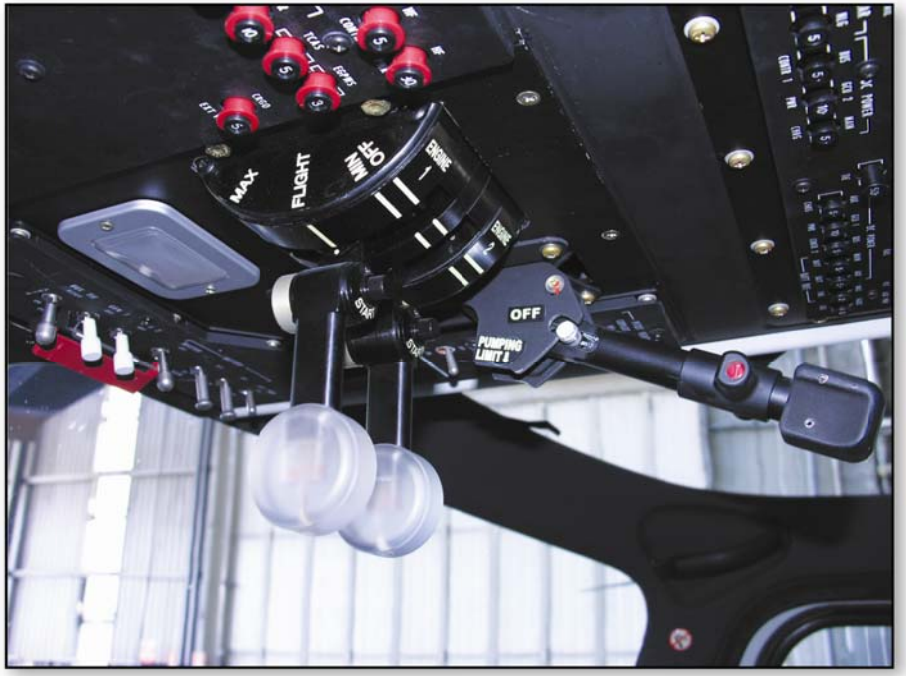
INSTRUMENT PANEL



CENTRAL CONSOLE



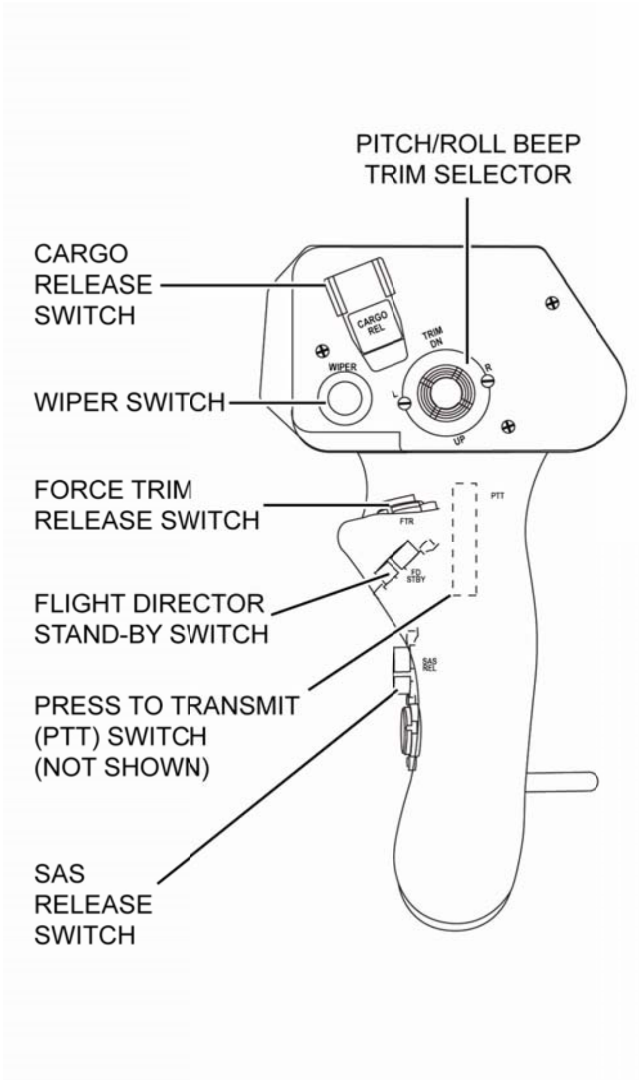
OVERHEAD CONSOLE



OVERHEAD CONSOLE – COPILOT SIDE WALL – AND ENGINE CONTROL LEVERS (ECL)

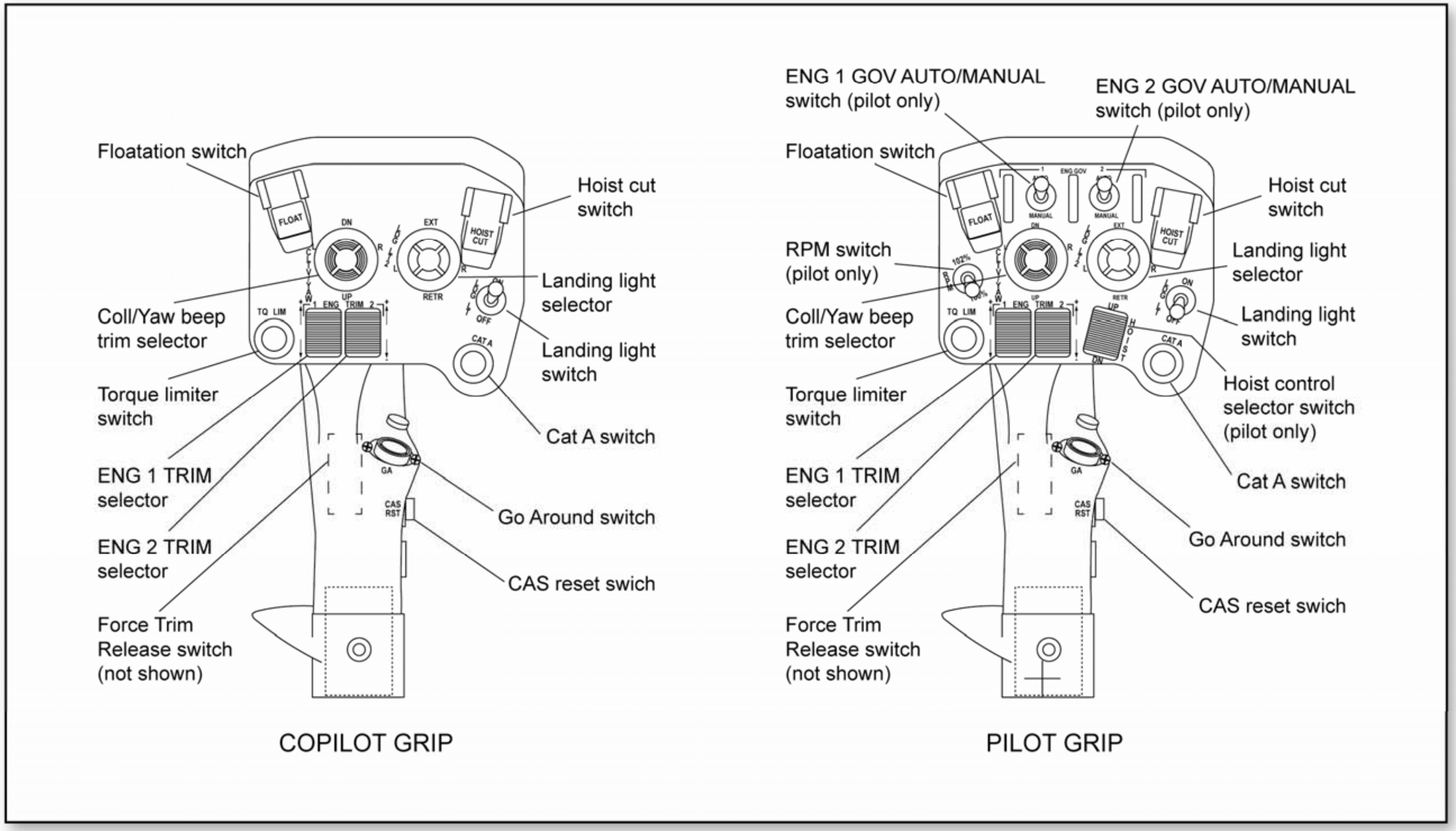
OVERHEAD CONSOLE – PILOT SIDE WALL

STANDBY OUTSIDE AIR TEMPERATURE (OAT) INDICATOR

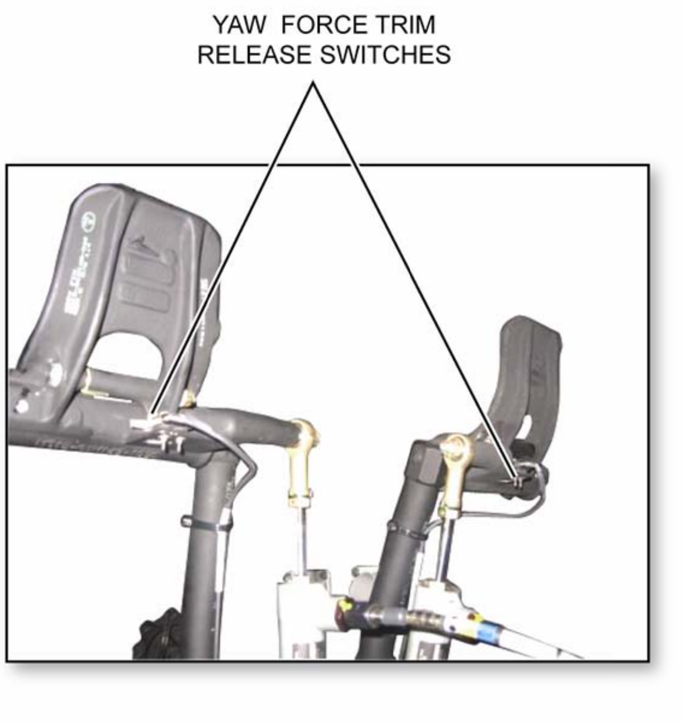


PILOT / COPILOT CYCLIC STICK

PILOT / COPILOT COLLECTIVE LEVER



PILOT / COPILOT COLLECTIVE LEVER



NOTE: Copilot station arranged symmetrically

PEDALS AND FOOT SWITCH

CHAPTER
00
AIR VEHICLE GENERAL

SECTION 11 – ACRONYMS LIST

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A

AC	Alternating Current
ACCB	Air Conditioning Control Box
ACP	Audio Control Panel
ADC	Air Data Computer
ADI	Attitude Director Indicator
ADF	Automatic Direction Finder
ADM	Air Data Module
ADS	Air Data System
AEO	All Engine Operative
A/F	Airframe
AFCS	Automatic Flight Control System
AGB	Accessory Gear Box
AGL	Above Ground Level
Ah	Ampere hour
AHRS	Attitude And Heading Reference System
AHRU	Attitude Heading Reference Unit
AIOP	Actuator Input/Output Processor (Module)
ALS	Ambient Light Sensor
ALT	Barometric Altitude
ALTA	Altitude Acquire
ALT SEL	Altitude Select
AMLCD	Active Matrix Liquid Crystal Display
AMM	Air Management Module
AMSL	Above Mean Sea Level
AOA	Angle of attack
AP	Autopilot
APP	Approach
APM	Aircraft Personality Module
ARINC	Aeronautical Radio Inc.
ASCB-D	Avionics Standard Communication Bus ver.D
ASEL	Altitude Preselect
ATC	Air Traffic Control
ATT	Attitude retention mode
AWG	Aural Warning Generator

B

BC	Back Course
BDGW	Basic Design Gross Weight
BFO	Beat Frequency Oscillator
BIT	Built In Test
BL	Buttock Line
BOD	Bottom Of Descent
BOV	Bleed Valve
BOW	Basic Operating Weight
BRG	Bearing

C

°C	Celsius degree
CAS	Crew Alerting System
CAS	Calibrated Air Speed
CB	Circuit Breaker
CCD	Cursor Control Device
CCP	Cockpit Control Panel
CCU	Cockpit Control Unit
CCW	Counter Clock-Wise
CG	Center of Gravity

CIO	Control Input/Output (MAU Module)
CMC	Central Maintenance Computer
CMC-RT	Central Maintenance Computer Remote Terminal
CMS	Central Maintenance System
COMM	Communication
CPI	Crash Position Indicator
CPLT	Copilot
CPL	Coupled/Decoupled
CSIO	Custom Input/Output (MAU Module)
CVR	Cockpit Voice Recorder
CW	Clock-Wise
CWS	Central Warning System

D

DAU	Digital Acquisition Unit
DBM	Data Base Module
DC	Direct Current
DCL	Deceleration
DCM	Detachable Configuration Module
DCU	Data Collection Unit
DF	Directional Finder
DG	Directional Gyro
DGPS	Differential Global Positioning System
DGR	Degraded
DH	Decision Height
DICP	Display Instrument Control Panel
DME	Distance Measuring Equipment
DMG	Digital Map Generator
DN	Down
DR	Dead Reckoning
DU	Display Unit
DWS	Debris Warning System

E

EAPS	Engine Air Particle Separator
EASA	European Aviation Safety Agency
ECL	Engine Control Lever
ECP	Engine Control Panel
ECS	Environmental Control System
ECU	Engine Control Unit
EDU	Electronic Display Unit
EEC	Electronic Engine Control
EFIS	Electronic Flight Instrument System
EGPWS	Enhanced Ground Proximity Warning System
EICAS	Engine Instrument and Crew Alerting System
ELT	Emergency Locator System
EMS	Emergency Medical Service
ENAC	Ente Nazionale Aviazione Civile
EPU	Estimated Position Uncertainty
ET	Elapsed Time
EX	Extension
EXT	External

F

FAA	Federal Aviation Administration
FCC	Flight Control Circuit

F/C	Flight Control
FCU	Fuel Computer Unit
FD	Flight Director
FDR	Flight Data Recorder
FDR	Flight Data Recorder
FH	Flying Hours
FLIR	Forward Looking Infra Red
FMCW	Frequency Modulated Continuous Wave
FMM	Fuel Management Module
FMS	Flight Management System
FOD	Foreign Object Damage
FOG	Fiber Optic Gyro
FOM	Figure of Merit
FTR	Force Trim Release
FWD	Forward

G

GA	Go-Around
GBS	Ground Based Software
GC	Guidance Controller
GCU	Generation Control Unit
GI	Ground Idle
GOV	Governor (Engine)
GPS	Global Positioning System
GPWS	Ground Proximity Warning System
GS	Glide Slope
GSE	Ground Support Equipment
GW	Gross Weight

H

HCB	Heating Control Box
HDG	Heading
HF	High Frequency
HOV	Hover
HP	High Pressure
HP	Horse Power
HSI	Horizontal Situation Indicator

I

IAS	Indicated Air Speed
ICS	Intercommunication System
IDS	Integrated Display System
IFR	Instrument Flight Rules
IGB	Intermediate Gearbox
IGE	In Ground Effect
ILS	Instrument Landing System
IR	Infra Red
ISA	International Standard Atmosphere

J

JAA	Joint Aviation Authority
JAR	Joint Airworthiness Regulations

L

LAN	Local Area Network
LAT	Lateral
LCD	Liquid Crystal Display
LDG	Landing
LED	Light Emitting Diode
LGCL	Landing Gear Control Lever
LGCP	Landing Gear Control Panel
LGCV	Landing Gear Control Valve
LH	Left Hand
LLS	Lightning Sensor System
LONG	Longitudinal
LOS	Line-of-Sight
LRM	Line Replaceable Module
LRU	Line Replaceable Unit
LSS	Lightning Sensor System
LT	Light
LVDT	Linear Variable Differential Transducer

M

MAU	Modular Avionics Unit
MB	Marker Beacon
MCDU	Multifunction Control Display Unit
MCL	Master Caution Light
MCP	Maximum Continuous Power
MDA	Minimum Descent Altitude
MFD	Multi-Function Display
MGB	Main Gear Box
MKR BCN	Marker Beacon
MLG	Main Landing Gear
MLS	Microwave Landing System
MPFDR	Multipurpose Flight Data Recorder
MPOG	Minimum Pitch On Ground
MR	Main Rotor
MRA	Main Rotor Actuator
MRC	Modular Radio Cabinet
MTBF	Mean Time Between Failures
MTTR	Maintenance Time to Replace
MWL	Master Warning Light

N

NAV	Lateral Navigation
Nf or NF	Engine free turbine speed
Ng	Engine gas generator turbine
NIC	Network Interface Controller
NIM	Network Interface Module
NLG	Nose Landing Gear
NM	Nautical Mile
Nr	Rotor rpm
NVG	Night Vision Goggle
NVM	Non Volatile Memory

O

OAT	Outer Air Temperature
OEI	One Engine Inoperative

OGE	Out of Ground Effect

P

PAX	Passengers
PCM	Power Control Module
PDP	Power Distribution Panel
PFD	Primary Flight Display
PI	Power Index
PLA	Power Lever Angle (throttle)
PWR	Power

Q

R

R/A	Retract Actuator
RADALT	Radio Altitude
RB	Rotor Brake
RBA	Rotor Brake Assembly
RBAA	Rotor Brake Actuation Assembly
RBCL	Rotor Brake Control Lever
RBCM	Rotor Brake Control Module
RBPI	Rotor Brake Pressure Indicator
RBRB	Rotor Brake Relays Box
RCP	Reversion Control Panel
RFM	Rotorcraft Flight Manual
RH	Right Hand
RHT	Radar Altitude Hold
RIC	Remote Instrument Controller
RICP	Remote Instrument Control Panel
RNAV	Area Navigation
RNP	Required Navigation Performance
ROC	Rate of Climb
RPM	Revolution Per Minute
RSB	Radio System Bus
RTD	Resistance Temperature Device

S

SA	Shortening Actuator
SAR	Search and Rescue
SAS	Stability Augmentation System
SHP	Shaft Horse Power
S/N	Serial Number
SOV	Shut Off Valve
STA	Station (line)
STAR	Standard Terminal Arrival Route
STBY	Stand-By

T

TAWS	Terrain Awareness and Warning System
TACAN	Tactical Air Navigation
TAS	True Air Speed
TBD	To Be Defined
TBO	Time Between Overhaul

TCAS	Traffic Alert and Collision Avoidance System
TCPS	Temperature Compensated Pressure Switch
TCV	Temperature Control Valve
TGB	Tail Gear Box
TO	Take Off
TOC	Top Of Climb
TOD	Top Of Descent
TOP	Take Off Power
TQ	Engine Torque
TR	Tail Rotor
TRA	Tail Rotor Actuator
TRSOV	Tail Rotor Shut-Off Valve

U

UHF	Ultra High Frequency
UP	Up
UTIL SOV	Utility Shut-Off Valve

V

V_{LO}	Maximum landing gear operating speed
V_{LE}	Maximum landing gear extended speed
V_{NE}	Never Exceed speed
VDR	VHF Data Radio
VFR	Visual Flight Rules
VGP	Vertical Glide Path
VHF	Very High Frequency
VIDL	VOR/ILS Data Link
VIP	Very Important Person
VMS	Vehicle Monitoring System
VOR	VHF Omnidirectional Range
VREF	Reference Speed
VROC	Vertical Rate Of Climb
VS	Vertical Speed
VSI	Vertical Speed Indicator

W

WGT	Weight
WL	Water Line
WOW	Weight-On-Wheels
WXR	Weather Radar

X

XFEED	Crossfeed
XPDR	Transponder

Y

Z

CHAPTER
00
AIR VEHICLE GENERAL

SECTION 40 – TECHNICAL PUBLICATION

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GENERAL

The flight and maintenance operations must be carried out according to the officially issued documents which are composed of

- Rotorcraft Flight Manual (RFM)
- Interactive Electronic Technical Publication (IETP)
  - Maintenance Publication
  - Component Repair and Overhaul Manual
  - Structural Repair Manual
  - Illustrated Parts Catalogue
  - Fault Isolation Manual
  - Wiring Diagram Manual
  - Illustrated Tools and Equipment Manual
  - Master Minimum Equipment List

In addition a Quick Reference Handbook (QRH) is also available as a checklist that is mandatory for single-pilot operations.

ROTORCRAFT FLIGHT MANUAL (RFM)

It provides all the information required to operate the helicopter in normal and emergency conditions. It is divided into

RFM TABLE OF CONTENTS

PART I – E.A.S.A. APPROVED

- 1 - Limitations
- 2 - Normal Procedures
- 3 - Emergency and Malfunctions Procedures
- 4 - Performances Data
- 5 - Optional Equipment Supplement

PART II – MANUFACTURER'S DATA

- 6 - Weight and Balance
- 7 - Systems Description
- 8 - Handling, Servicing and Maintenance
- 9 - Supplemental Performance Information

The Limitations section contains limitations required by regulation or to safely operate rotorcraft, powerplant, systems, and equipment. It includes operating limitations, instrument markings, colour coding, and basic placards.

The Normal Procedures section contains the checklist for the normal procedures ordered by phase of flight. Normal procedures are the result of extensive flight tests and experience with the AW139 aircraft. They are intended to ensure that the level of safety required by the design and certification process is achieved.

The Emergency and Malfunctions Procedures section contains the procedures that must be performed in the event of an emergency or malfunction. These procedures are based on experience acquired in the operation of helicopters, in general, and on flight tests conducted on AW139 helicopter. The Emergency and Malfunction procedures are presented in the form of logic trees (flow charts). These flow charts have been formulated based on analysis and test of the cockpit indications that would be available to the flight crew following the failures/malfunctions that are included in this section. The section includes three sets of procedures:

- Emergency procedures for CAS messages
- Malfunction procedures for CAS messages
- Emergency and malfunction procedures for PDF indications

Emergency procedures are related to warning (red) messages/indications.
Malfunction procedures are related to caution (amber) messages/indications.

The Performance Data section includes charts with standard performance data based on flight test results and engineering analysis.

The Optional Equipment Supplement section contains all the information necessary to operate optional equipment. Each supplement is arranged in 4 sections:

- Section 1 – Limitations
- Section 2 – Normal procedures
- Section 3 – Emergency and malfunction Procedures
- Section 4 – Performance Data

The weight and balance data contain the charts that permit to determine the aircraft weight and the position of the center of gravity.

INTERACTIVE ELECTRONIC TECHNICAL PUBLICATION (IETP)

The IETP is distributed on CD-ROM and includes all the technical publications used to properly perform all maintenance tasks to permit the Release To Service of the AW139 helicopter, including the Master Minimum Equipment List (MMEL).

MAINTENANCE PUBLICATION

It provides all the information required to perform all the procedures used to preserve the airworthiness and flight characteristic of the helicopter. It contains the following information

- inspection requirements
- maintenance procedures
- removal and installation procedures
- test and inspection

COMPONENT REPAIR AND OVERHAUL MANUAL

It provides all the information required to the disassembly, inspection, repair and reassembly of the major helicopter components when applicable.

STRUCTURAL REPAIR MANUAL

It provides all the information required for the identification of structure damages and the repair associated.

ILLUSTRATED PARTS CATALOGUE

It provides all the illustration and identification data about the replaceable parts of the air vehicle for which the maintenance procedures has been provided.

FAULT ISOLATION MANUAL

It provides all the information and procedures required by the user to isolate faults not identified by built-in test equipment.

WIRING DIAGRAM MANUAL

It provides all the electrical/electronic wiring diagrams required for maintenance tasks.

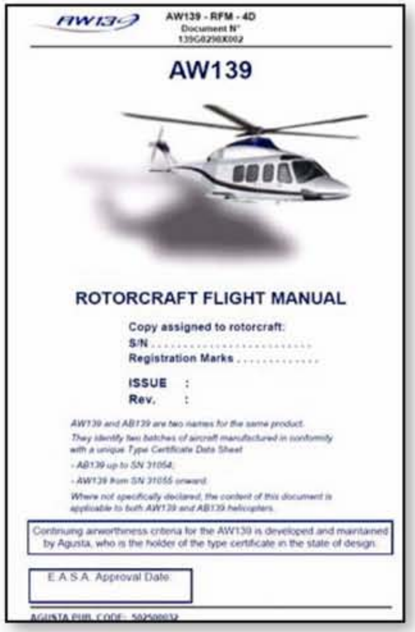
ILLUSTRATED TOOLS AND EQUIPMENT MANUAL

It provides all the characteristics and the illustrations of all the special tools and equipment, including test equipment recommended for the maintenance of the air vehicle.

MASTER MINIMUM EQUIPMENT LIST

It provides the list of all the airborne equipment which is mandatory to achieve a safe flight condition.

INTERACTIVE ELECTRONIC TECHNICAL PUBLICATION (IETP)



AW139 - RFM - 4D
Document N°
139G0290X002

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PART I - E.A.S.A. APPROVED

E.A.S.A. Approved TOC-1

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PART II - MANUFACTURER'S DATA

TOC-2 E.A.S.A. Approved

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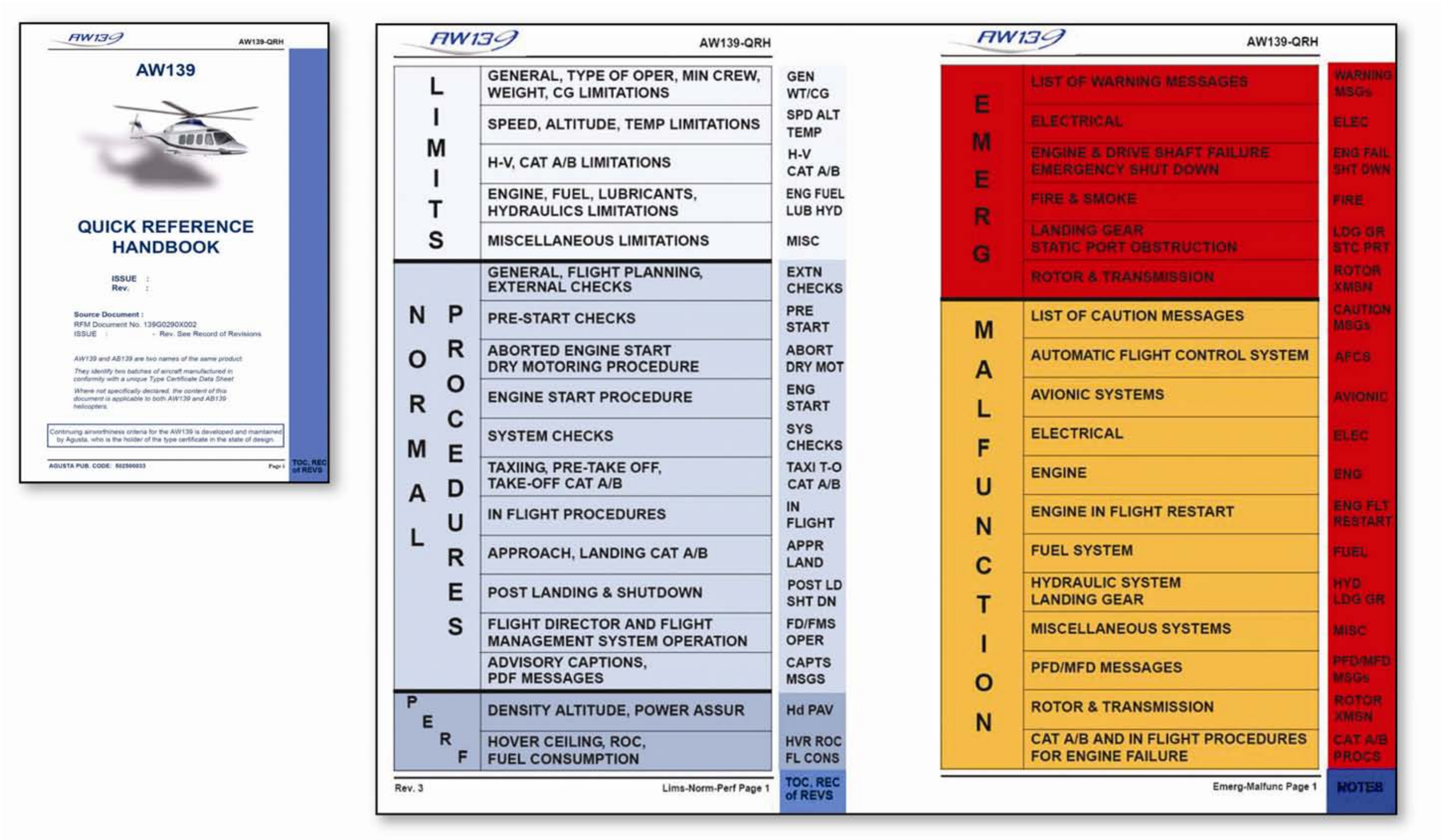
ROTORCRAFT FLIGHT MANUAL



QUICK REFERENCE HANDBOOK

CHAPTER
21
ENVIRONMENTAL CONTROL

SECTION 00 – GENERAL

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ENVIRONMENTAL CONTROL – GENERAL

The environmental control system consists of the following sub-systems:

- ventilation
- heating
- air conditioning

VENTILATION SYSTEM – GENERAL

The purpose of the ventilation system is to supply fresh air to the cockpit and cabin.

The ventilation system is composed of three independent sub-systems as follows:

- Pilot ventilation sub-system
- Copilot ventilation sub-system
- Cabin ventilation sub-system

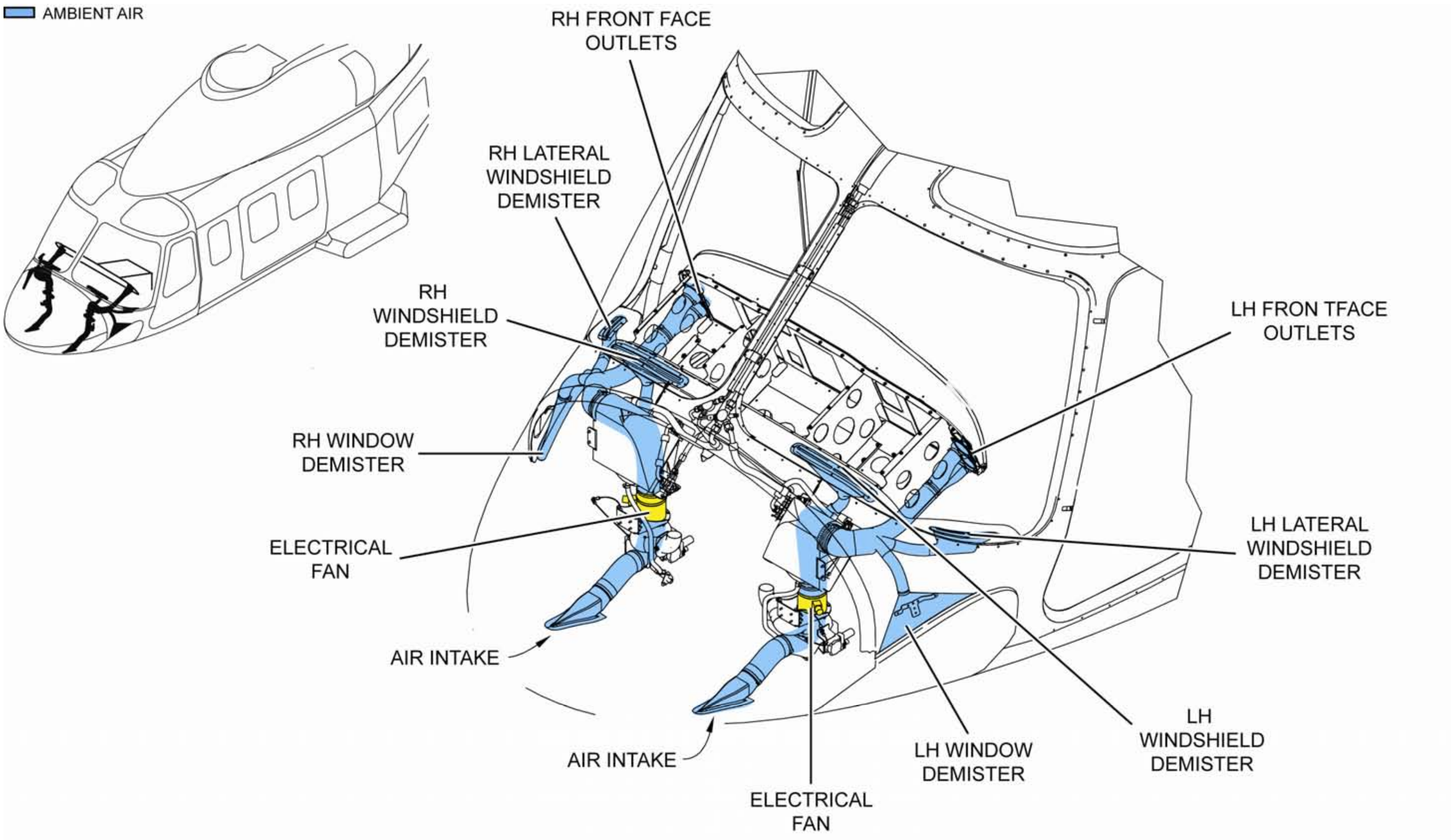
Pilot and copilot ventilation systems make up the cockpit ventilation system controlled by a single CREW selector on the Environmental Control System (ECS) control panel.

The cabin ventilation system is controlled by the PAX selector on the ECS control panel.

COCKPIT VENTILATION - GENERAL

Pilot and copilot ventilation systems are independent and arranged symmetrically but controlled by a single rotary knob only (VENT CREW). Each of them is composed of:

- a ram air intake located under the lower part of the nose compartment
- a flapper valve, electrically controlled to be either fully open (ventilation on) or fully closed (ventilation off)
- an electrical fan for forced air operation
- five outlets:
  - two adjustable face outlets on the instruments panel
  - two free outlets for the windshield (main and side)
  - one free outlet for the lower window.

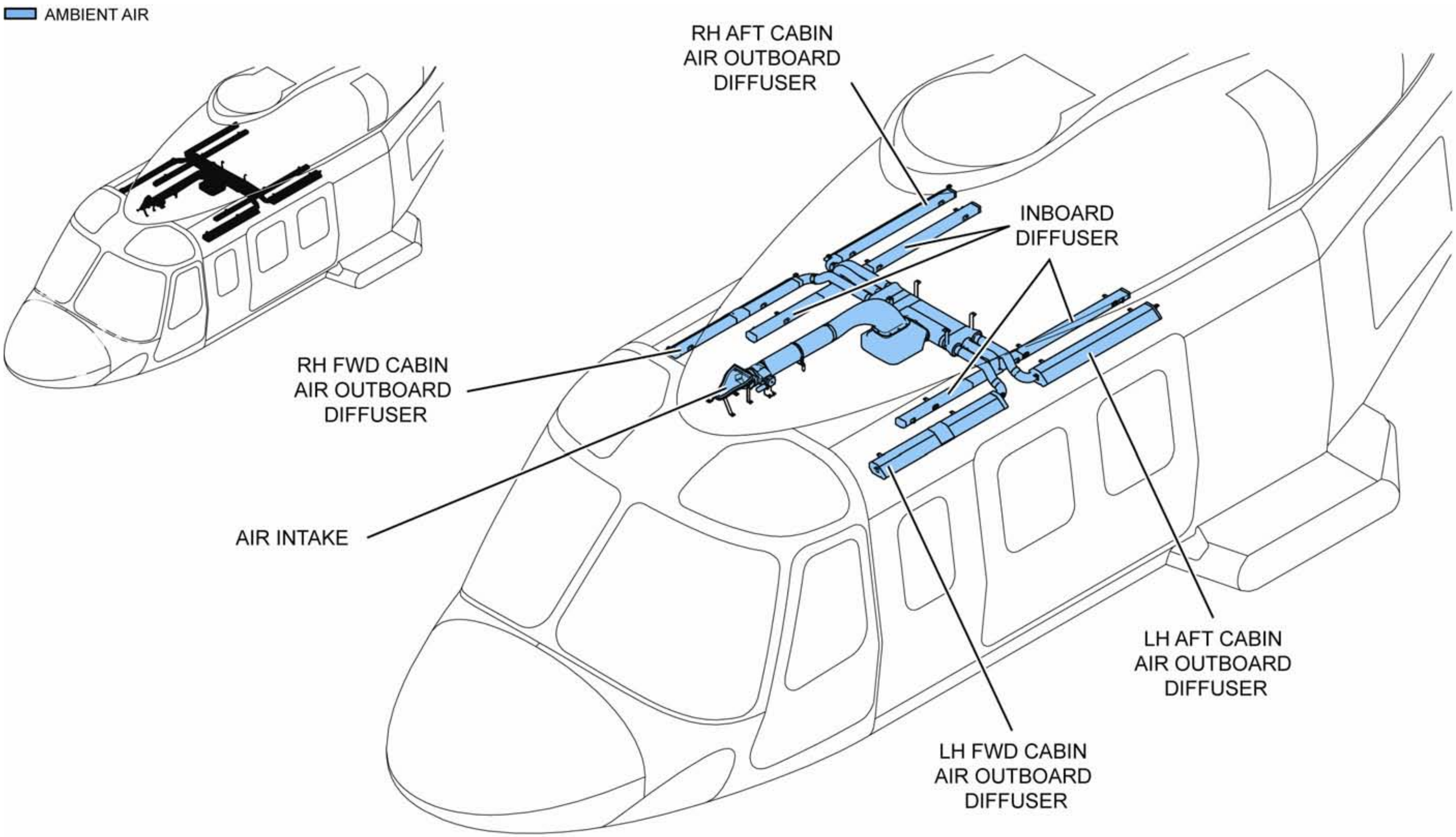


COCKPIT VENTILATION

CABIN VENTILATION - GENERAL

The cabin ventilation system is a single independent system which provides fresh air to passengers and is composed of:

- a ram air intake located on the upper deck fairing
- a flapper valve electrically controlled to be either fully open (ventilation on) or fully closed (ventilation off)
- two electrical fans for forced air operation
- twelve adjustable outlets located in the PSUs (Passenger Service Units)



CABIN VENTILATION

VENTILATION SYSTEM – CONTROLS AND INDICATORS

1. VENT CREW rotary knob

   OFF ………............…….. pilot and copilot flapper valves are closed (no airflow)

   ON .....……………………... pilot and copilot flapper valves are open (ram airflow)

   FAN LOW ……................. pilot and copilot flapper valves are open and electrical fan operates at low speed (forced airflow)

   FAN HIGH ……………..…. pilot and copilot flapper valves are open and electrical fan operates at high speed (forced airflow)

2. VENT PAX rotary knob

   OFF ……...………….……. cabin flapper valves are closed (no airflow)

   ON .....…………………….. cabin flapper valves are open (ram airflow)

   FAN LOW ……................. cabin flapper valves are open and electrical fan operates at low speed (forced airflow)

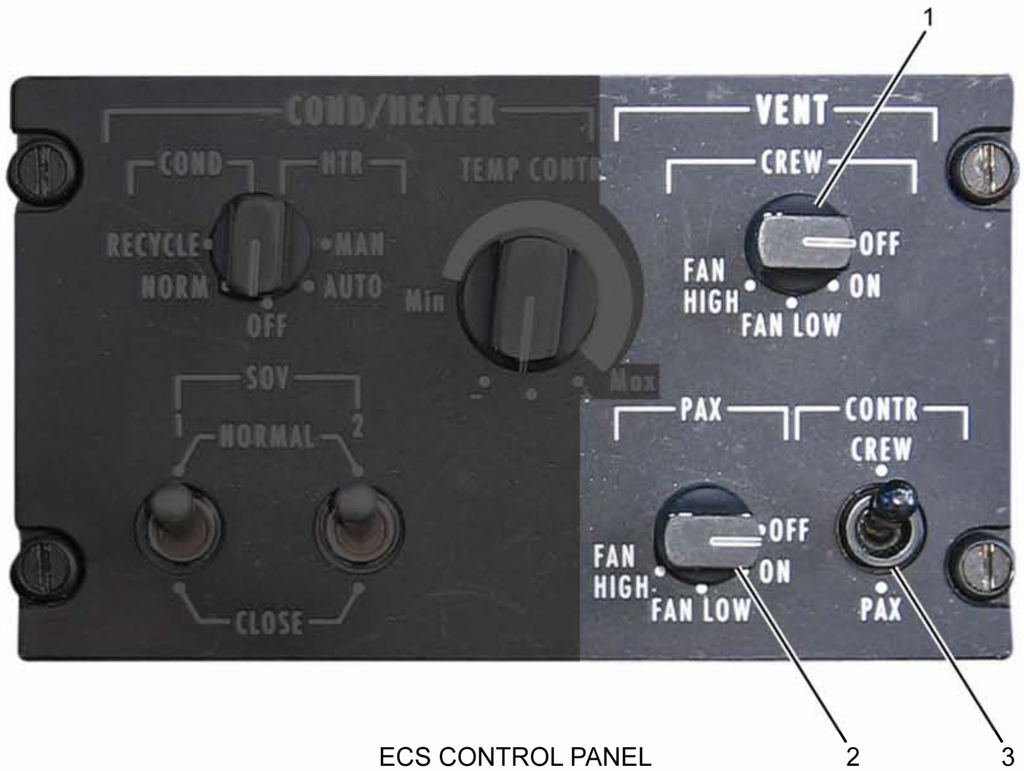
   FAN HIGH ……………..…. cabin flapper valves are open and electrical fan operates at high speed (forced airflow)

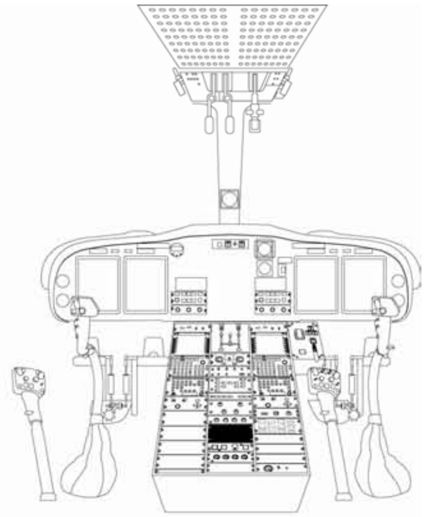
3. VENT CONTR switch

   CREW ...…....................…. enables the VENT PAX rotary switch (2)

   PAX ...……………………... enables the VENT rotary switch in the cabin (optional)

   NOTE. If the cabin controller is not installed, selecting the VENT CONTR switch to PAX causes the cabin flapper valves to open (ram airflow) and disables the VENT PAX rotary switch.





ECS CONTROL PANEL

VENTILATION CONTROLS AND INDICATORS

VENTILATION – CAS CAUTION MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
VENT FAIL	Failure of the crew ventilation fan	VENT FAN FAILURE	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES ENVIRONMENTAL CONTROL SYSTEM
VENT FAIL **NOSE FAN 1 OFF** **NOSE FAN 2 OFF**	Failure of both nose avionic bay fans (long nose configuration only) Caution triggered on ground only	NOSE AVIONIC FANS FAILURE	

VENTILATION – CAS ADVISORY MESSAGES

CAS CAPTION	MESSAGE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
FWD VENT	Forward (crew) ventilation fan ON		Supplement 2 VENTILATION, HEATING AND AIR CONDITIONING SYSTEMS Section 2 NORMAL PROCEDURES
AFT VENT ON	Cabin fan switched ON		
FWD-AFT VENT ON	Both forward (crew) and cabin fans ON		

VENTILATION – LIMITATIONS

Refer to AW139-RFM-4D Section 1.

HEATING SYSTEM – GENERAL

The purpose of the heating system is to supply warm air to the cockpit and cabin to maintain a comfortable environment and to defrost windshields and lower windows.
The heating system supplies cockpit and cabin with a mix of hot pressurized air bled from the compressor discharge port (P3) of both engines and external air sucked in through an air inlet on the LH aft fuselage.
Two solenoid controlled bleed air shut-off valves (SOV) control the relevant engine hot pressurized air to supply the heating system when selected on by the pilot and the engine operates normally.
The bleed air SOV are automatically closed if any of the following occurs:

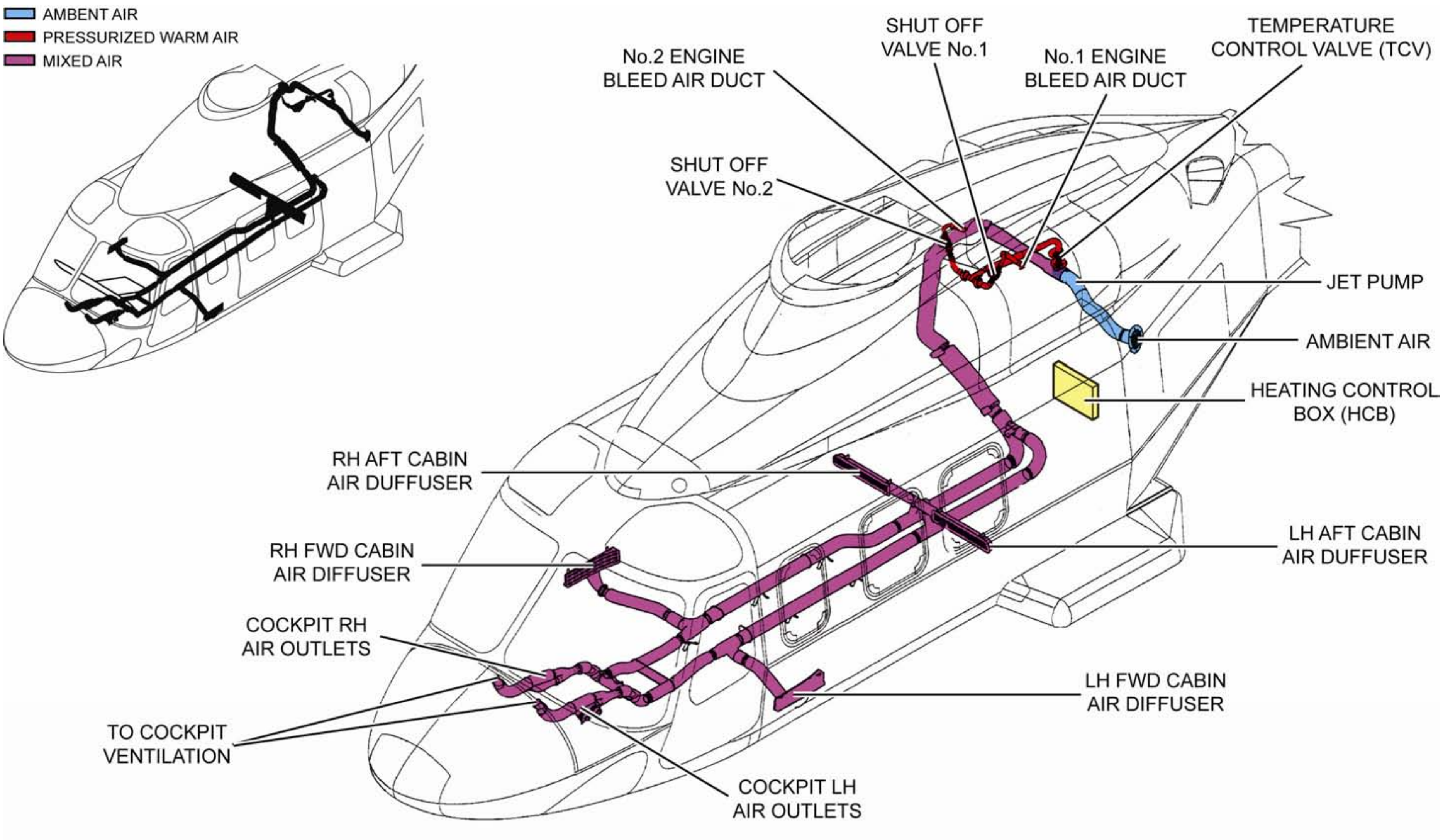
- a failure in the heating system is detected
- engine is not running
- fire extinguishing system is armed (see Ch.26-00-00)
- loss of electrical control signal

The engine hot pressurized air is routed to the Temperature Control Valve (TCV) which controls the quantity of hot air to be mixed with outside fresh air sucked in by a jet pump. The mixing occurs in the jet pump.

The mixed air enters the cabin and the cockpit via the air distribution ducts. The right and left diffusers are located on the floor area. They provide the distribution of the heated air through the cabin (passenger area). The distribution ducts are connected to the cockpit ventilation system and they are also dedicated to the pilots.

The airflow temperature is automatically controlled by the Heating Control Box (HCB) through the TEMP CONTR knob on the COND/HEATER control panel when the COND/HTR selector is set at AUTO.

In case of failure of the automatic temperature control, the pilot can manually control the position of the TCV by setting the COND/HTR selector to MAN and using the TEMP CONTR knob as a trim switch in the positions.



HEATING SYSTEM - GENERAL

HEATING – CONTROLS AND INDICATORS

4. COND/HTR selector

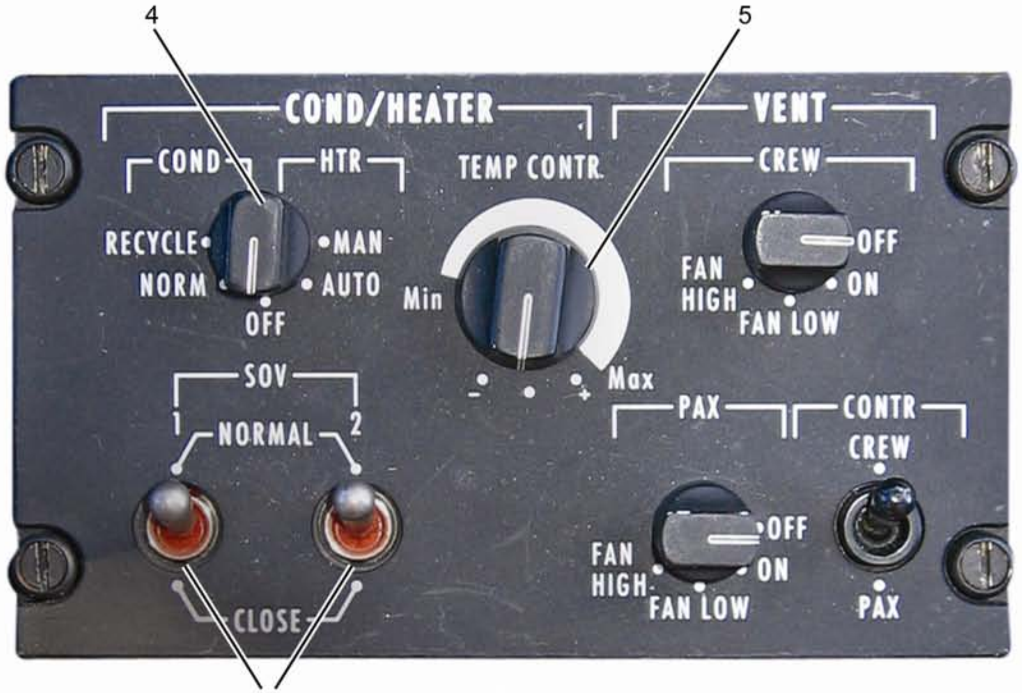
OFF, NORM, RECYCLE ….	the heating is off
AUTO ..…………..………....	the heating system keeps the cockpit and the cabin air at the selected temperature automatically
MAN ….…….………..…….…...	the heating is operated in manual mode

5. TEMP CONTR knob (with COND/HTR selector in HTR area)

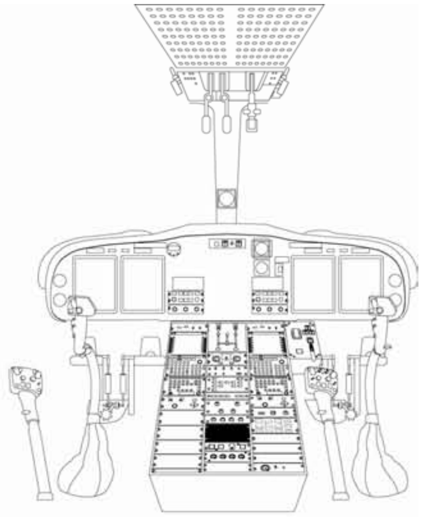
Min to Max range ……...	(potentiometer) selects the cockpit / cabin air temperature for automatic temperature control
● ……………………..……	Neutral position for manual temperature control. No input is given to the TCV which stay still.
+ ……………………..……	(momentary position) manually controls the TCV to open (increases temperature)
– ……………………..………	(momentary position) manually controls the TCV to close (decreases temperature)

6. SOV 1 (2) switch

NORMAL ..…..………...	the no.1 (no.2) shut-off valve is automatically opened or closed
CLOSE …....…………..	the no.1 (no.2) shut-off valve is forced to close



ECS CONTROL PANEL



HEATING CONTROLS AND INDICATORS

HEATING – CAS CAUTION MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
HEATER FAIL	Heater system failure	HEATER FAILURE	Supplement 2 VENTILATION, HEATING AND AIR CONDITIONING SYSTEMS

HEATING – CAS ADVISORY MESSAGES

CAS CAPTION	MESSAGE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
HEATER ON	Heater switched ON		Supplement 2 VENTILATION, HEATING AND AIR CONDITIONING SYSTEMS

HEATING – LIMITATIONS

Refer to AW139-RFM-4D Supplement 2.

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AIR CONDITIONING - GENERAL

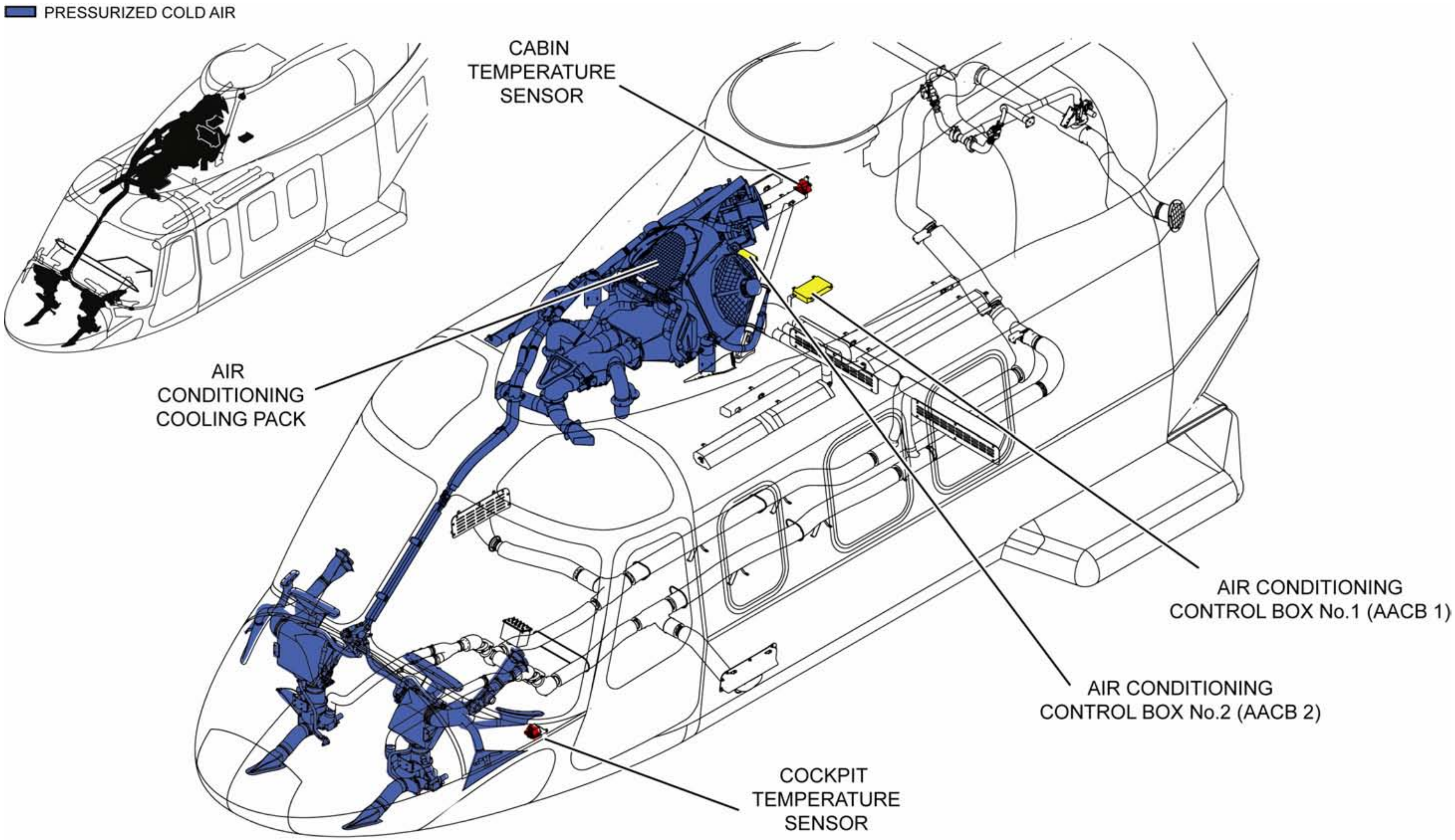
The purpose of the air conditioning system is to supply cool air to the cockpit and the cabin to maintain a comfortable environment.

The air conditioning system comprises two vapour cycle systems, one for the cockpit and one for the cabin, which use tetrafluoroethane (Freon) as refrigerant. Freon is non-toxic and non-flammable gas. Each system includes a compressor, a condenser, a heat exchanger and an evaporator. The compressors are mechanically driven by the Main Gear Box (MGB) through electromagnetic clutches.
The clutches open – thus mechanically disconnecting the compressors from the MGB – when the air conditioning is off.

Two Air Conditioning Control Boxes (ACCBs) compare the air temperature measured by the cockpit and cabin sensors with the temperature set on the ECS control panel and operate the related compressor on/off cycle as necessary.

Freon cools the ventilation air through the heat exchangers located in the cockpit and cabin ventilation ducts. In NORM mode the ventilation flapper valves are open and ram air is cooled. In RECYCLE mode the ventilation system flapper valves are closed and the recycle flapper valves are open: cockpit and cabin air is recirculated through the heat exchangers by the electrical fans and cooled.

The air conditioning system is supplied by circuit breakers grouped as ECS and connected to NON-ESS 1 (cockpit) and NON-ESS 2 (cabin).



AIR CONDITIONING – GENERAL

AIR CONDITIONING – CONTROLS AND INDICATORS

ECS control panel

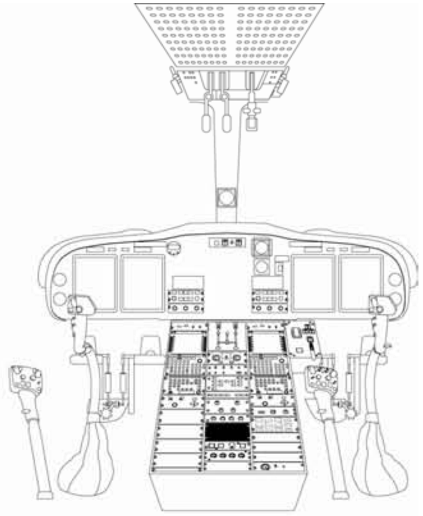
7. COND/HTR selector

OFF, AUTO, MAN ……….	the air conditioning is off
NORM ..………..………....	the air conditioning system keeps cockpit and cabin air at the selected temperature by cooling the ventilation air that enters the ram air intakes
RECYCLE …………...........	the air conditioning system keeps cockpit and cabin air at the selected temperature by cooling the ventilation air that is recirculated

8. TEMP CONTR knob (with the COND/HTR selector in COND area)

Min to Max range ……..….	(potentiometer) selects the cockpit/cabin air temperature for the air conditioning system to keep
……………………..	inoperative

ECS CONTROL PANEL



AIR CONDITIONING – CONTROLS AND INDICATORS

AIR CONDITIONING – CAS CAUTION MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
FWD COND FAIL	Crew conditioner failure	COND FAILURE	Supplement 2 VENTILATION, HEATING AND AIR CONDITIONING SYSTEMS
AFT COND FAIL	PAX conditioner failure	COND FAILURE	

AIR CONDITIONING – CAS ADVISORY MESSAGES

CAS CAPTION	MESSAGE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
AIR COND ON	Air conditioning system switched ON		Supplement 2 VENTILATION, HEATING AND AIR CONDITIONING SYSTEMS

AIR CONDITIONING – LIMITATIONS

Refer to AW139-RFM-4D Supplement 2

CHAPTER
22
AUTOPILOT

SECTION 00 – GENERAL

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AUTOMATIC FLIGHT CONTROL SYSTEM (AFCS) – GENERAL

The AW139 Automatic Flight Control System (AFCS) is part of the PRIMUS EPIC® Integrated Avionics System (see chapter 46-00).
The AFCS is available in four configurations with increasing capabilities:

- BASIC 3-AXIS AUTOPILOT SYSTEM. The configuration provides dual pitch, roll, and yaw stabilization and attitude retention (Autopilot – AP) with Force Trim on pitch, roll, yaw and collective.
  No Flight Director (FD) is included (VFR only).
- 4-AXIS SYSTEM WITH BASIC 3-CUE FLIGHT DIRECTOR. It consists of a 4-axis Autopilot (Basic 3-axis Autopilot with collective Trim) and a 3-cue (pitch, roll and collective) Flight Director (no Auto Hover mode). Ref.: RFM Supplement 40
- 4-AXIS SYSTEM WITH ENHANCED 3-CUE FLIGHT DIRECTOR. It consists of a 4-axis Autopilot (Basic 3-axis Autopilot with collective Trim) and a 3-cue (pitch, roll and collective) Flight Director including Auto Hover/ Velocity Hold mode. Ref.: RFM Supplement 34.
- 4-AXIS SYSTEM WITH ENHANCED 3-CUE FLIGHT DIRECTOR & SAR MODES. Same as the above configuration plus the following SAR modes: Transition Down (TD), Transition Up (TU), Winch-man Trim (WTR), Mark-On-Target (MOT).
  Ref.: RFM Supplement TBD.

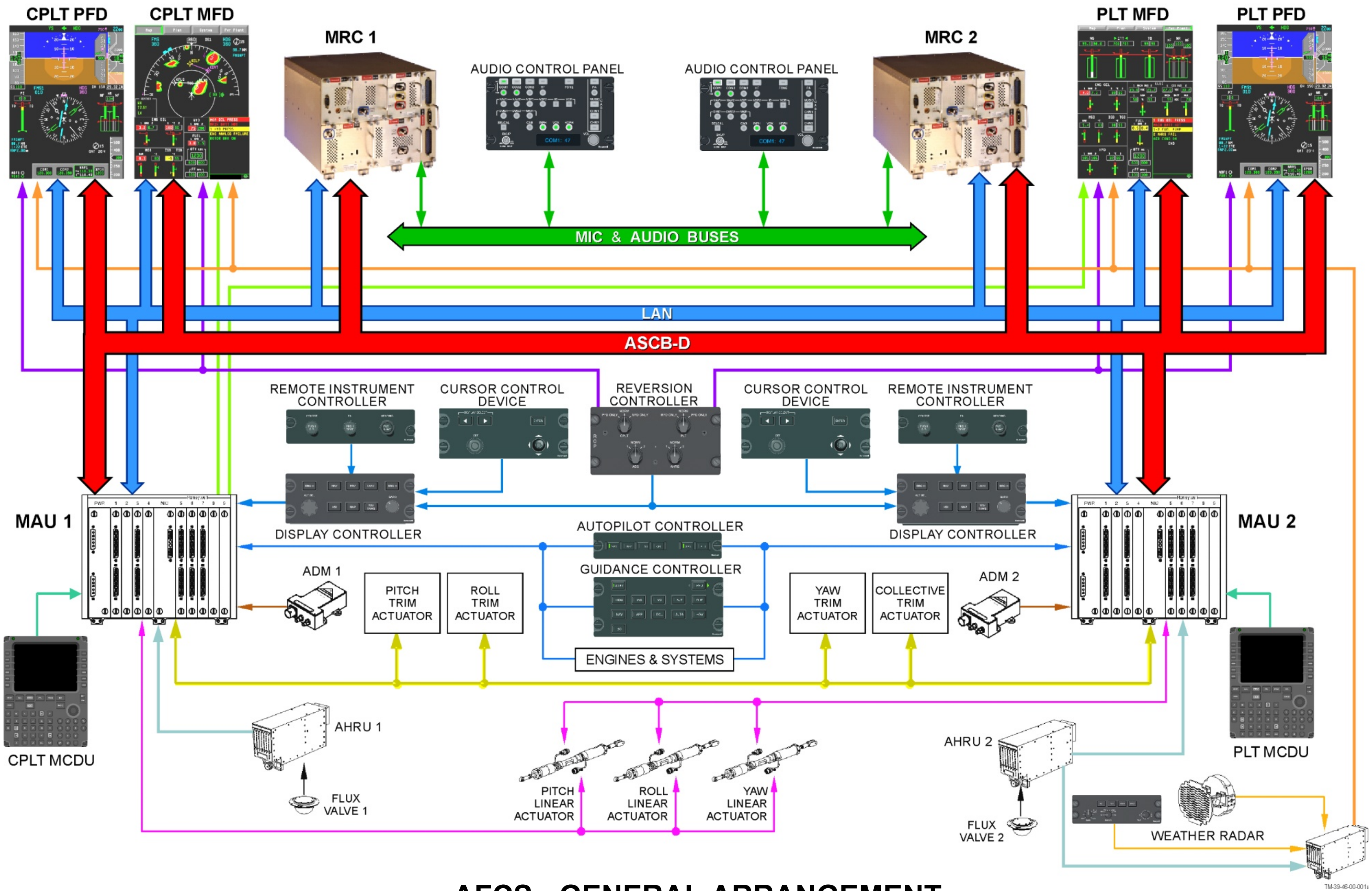
The AFCS is a system fully duplicated for redundancy and consists of:

- Autopilot No.1 (AP1)
- Autopilot No.2 (AP2)
- Flight Director No.1 (FD1)
- Flight Director No.2 (FD2)

The AFCS is an active-active system because both Autopilots operate on the flight controls at the same time.
The No.1 and No.2 AP and FD systems normally operate together. Each AP and FD is capable of providing full functionality also in case of failure of the paired system.
The AFCS is designed to be fail-safe because autopilot malfunctions are demonstrated to be safely recoverable by the pilot flying the helicopter manually.
The pilot can override the AFCS at any time by manually operating the flight controls. The system gives the pilot full authority over flight controls regardless of whether the autopilot is engaged or disengaged.

The dual AFCS uses:

- two AFCS computers (part of MAU1 and MAU2)
- two independent sets of sensors (AHRS, ADS, RAD ALT, VOR/ILS, FMS)
- two sets of Linear (series) Actuators (Pitch, Roll, Yaw)
- one set of single rotary Trim (parallel) Actuators (Pitch, Roll, Yaw and Collective)



AFCS - GENERAL ARRANGEMENT

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AFCS – MAIN COMPONENTS

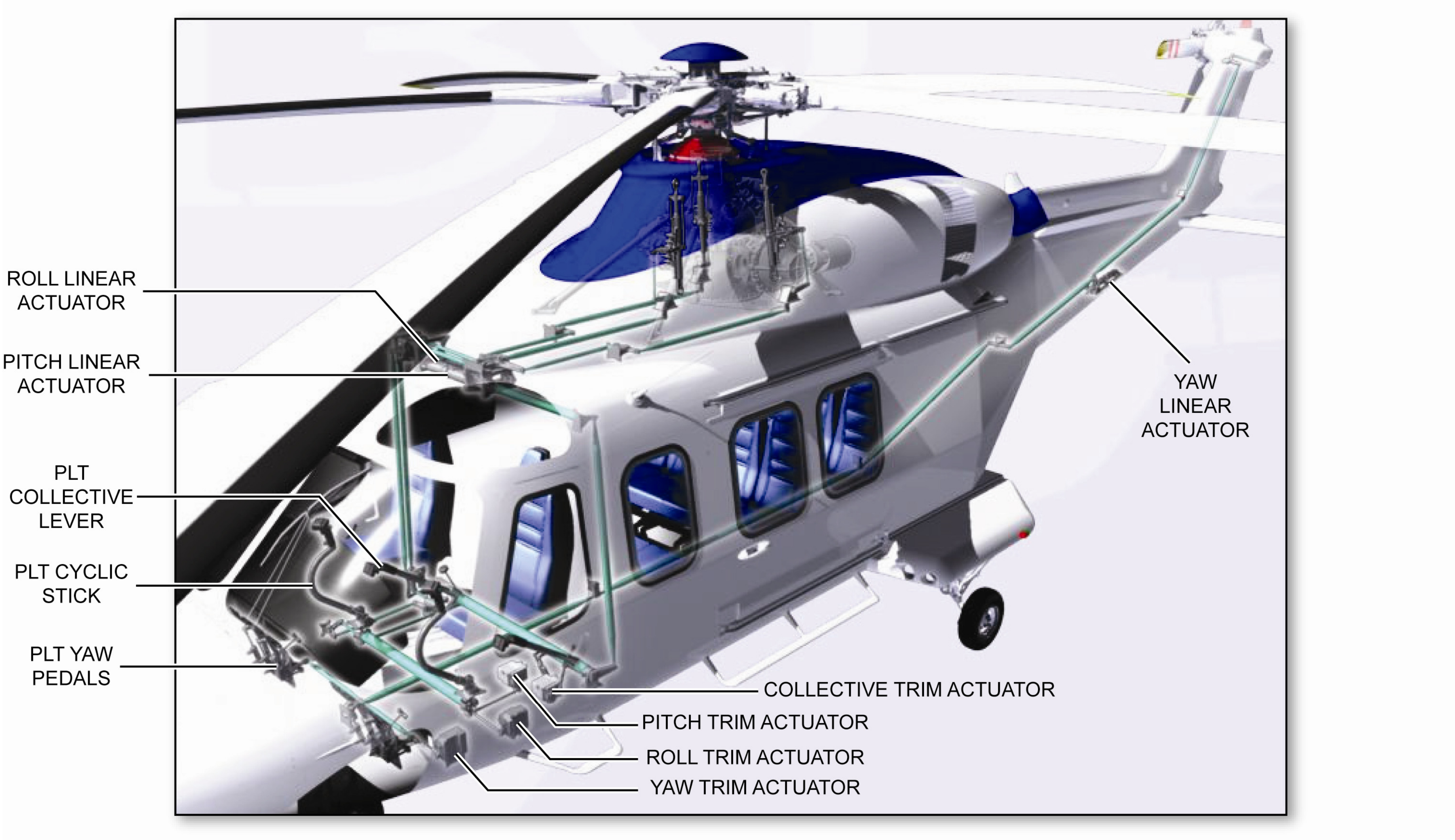
The AFCS consists of the following main components:

- four AFCS modules located inside the MAUs (two modules in each MAU)
- one Autopilot Controller
- one Guidance Controller
- three sets of dual Linear Actuators
- four Trim Actuators
- control switches on both cyclic sticks, both collective levers, both yaw pedals and on the central console

AFCS indications are provided on the Display Units.

The following systems provide the data necessary for AFCS operation:

- both ADS1 and ADS2
- both AHRS1 and AHRS2
- the Standby Instrument
- both Radar Altimeter 1 and Radar Altimeter 2
- both VHF NAV1 and VHF NAV2
- both FMS1 and FMS2



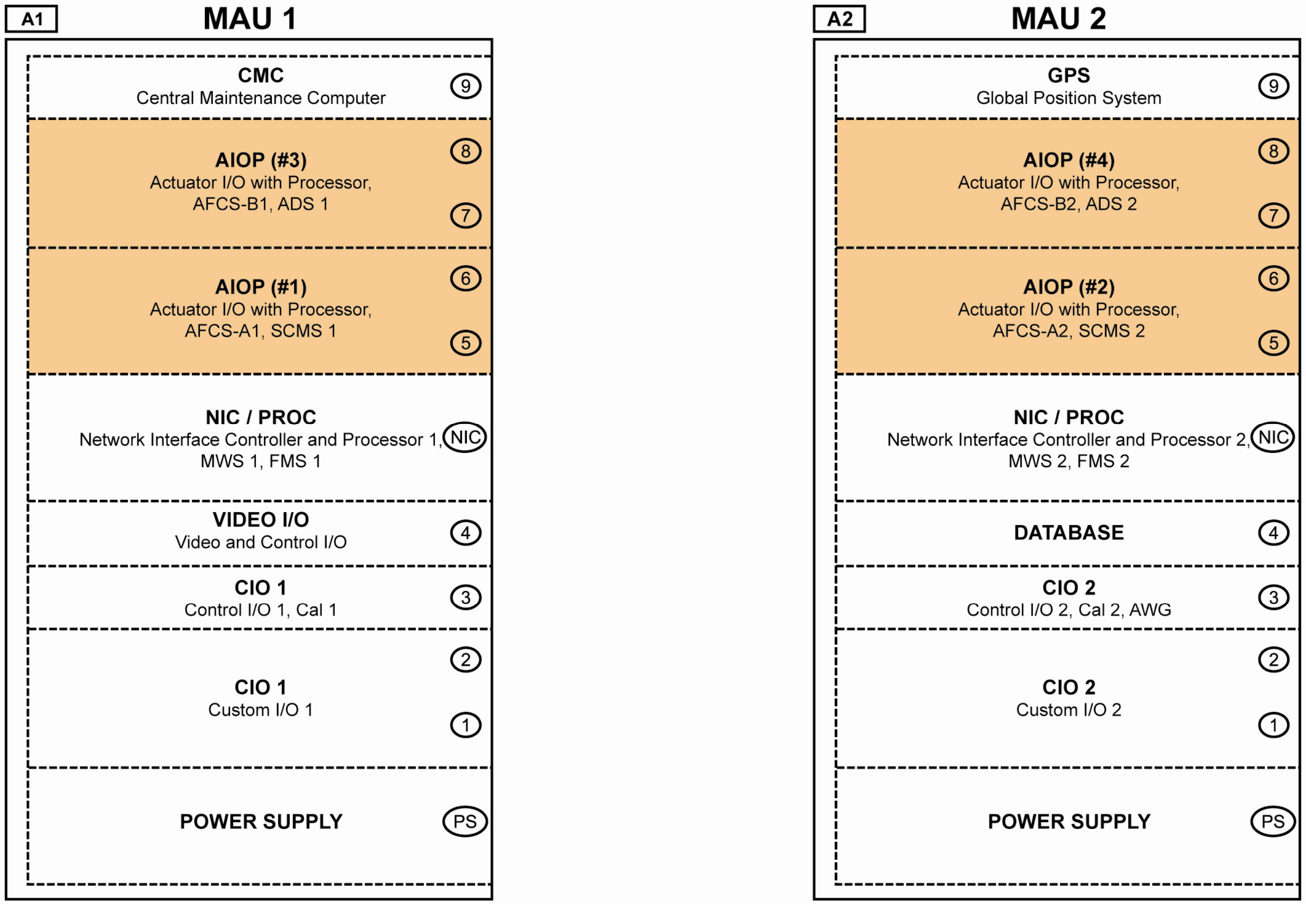
AFCS - MAIN COMPONENTS

MODULAR AVIONIC UNIT (MAU) – GENERAL

Two AFCS modules are installed in each Modular Avionic Unit (MAU) to perform AFCS computations, output commands and indication data and perform system monitoring.

The two AFCS modules —named Actuator Input/Output with Processor (AIOP) modules— in MAU1 are part of AFCS1 (AP1, FD1); the two AIOP modules in MAU2 are part of AFCS2 (AP2, FD2).
This makes the AFCS a dual-redundant computer system (AFCS1 and AFCS2) and each AFCS dual-redundant inside (channel A and channel B).

The two modules of an AFCS share the tasks and continuously monitor each other performances to positively identify any internal failure and, if the case, disengage automatically.



ADS = Air Data System

AFCS = Automatic Flight Control System

SCMS = Software Configuration Management System

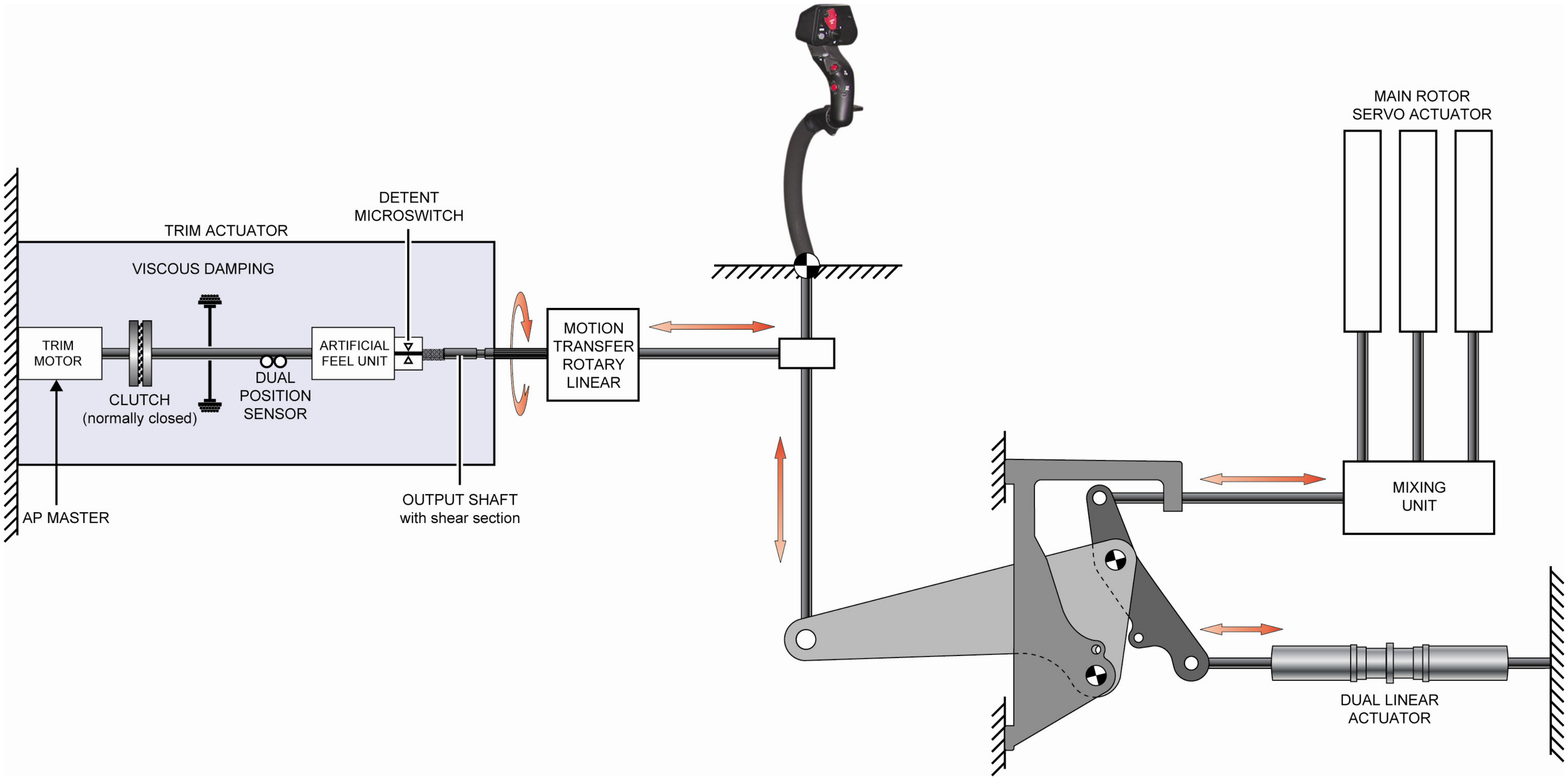
AWG = Aural Warning Generator

FMS = Flight Management System

MWS = Monitor Warning System

NIC = Network Interface Controller

MAU — AFCS MODULES



TYPICAL FLIGHT CONTROL AXIS SCHEMATIC (PITCH OR ROLL OR YAW)

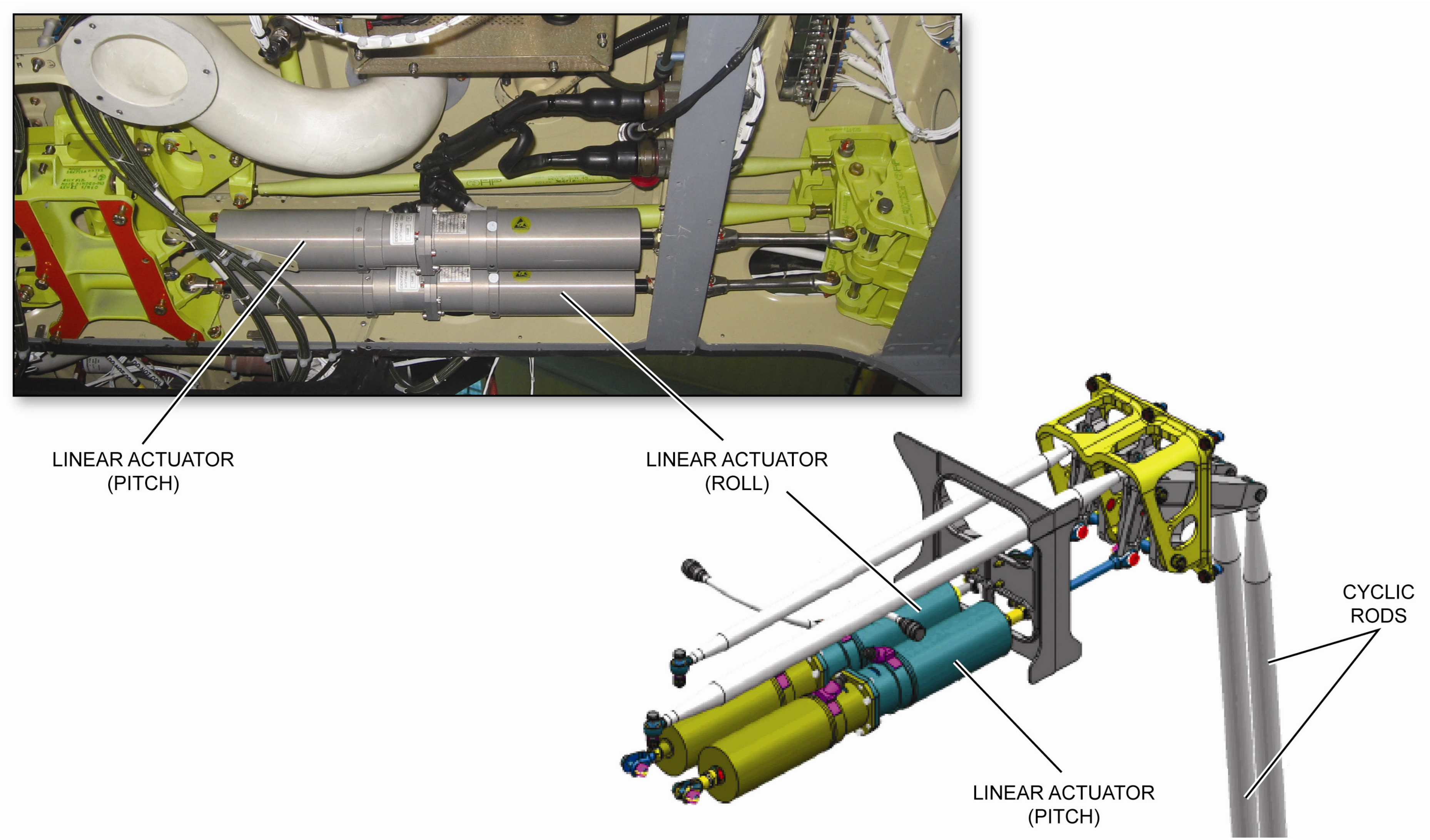
LINEAR ACTUATOR

Three sets of dual Linear Actuators provide limited control inputs to pitch, roll and yaw axis flight control lines in series with pilot input (see chapter 67-00).
Each set is connected to the relevant axis flight control line through a dual-action bellcrank that permits summing of actuator inputs to pilot inputs as well as preventing a Linear Actuator mechanical failure from losing pilot manual control on that axis.

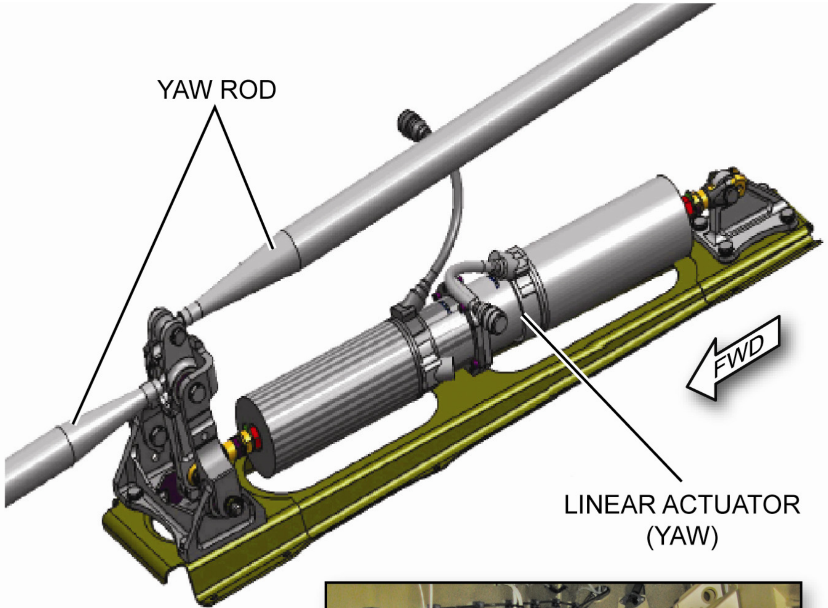
Each dual Linear Actuator set incorporates two identical and independent electrical motors, one controlled by AP 1 and one controlled by AP 2 via dedicated digital buses (CAN Bus).
They are also called “smart” linear actuators since they include a microprocessor to internally close the servo loop on the position command from the on-side MAU.
Each brushless motor drives a ball-screw which displaces the flight control line.

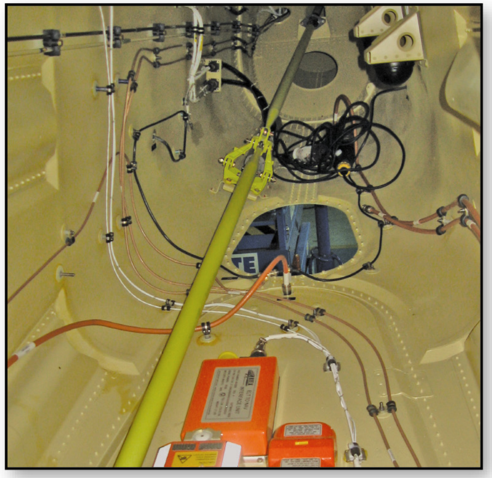
Each linear actuator includes an integrity centering function to protect against runaway failure modes.

During normal operation with both Autopilots engaged, each AP outputs 50% of the computed input for an axis.
In case of single AP operation, 100% of the computed input is provided to the on-side Linear Actuator: in this case the total authority of the control is reduced to a half, resulting in a degradation of the system performance.



PITCH AND ROLL LINEAR ACTUATORS





View inside the tailboom looking aft

YAW LINEAR ACTUATOR

TRIM ACTUATOR

Four rotary Trim Actuators provide full control of pitch, roll, yaw and collective axis for automatic flight.
The Trim Actuators are connected in parallel with pilot flight control input (cyclic stick, collective lever and pedals) and have the full authority on the flight control lines (see chapter 67-00).
Each Trim Actuator incorporates:

- A motor with non-reversible gears (Trim Motor)
- A magnetic clutch (Force Trim)
- An artificial feel device with detent microswitch
- A dual position sensor

TRIM MOTOR
The four Trim Motors are controlled by one Autopilot system at a time: the elected AP is the first that is engaged and is named "TRIM MASTER".
The "Autotrim" function of the Trim Master AP directly controls and monitors the Trim Motors using a position servo loop that is closed within the MAU. Position feedback is given to the MAU by a position sensor that is integrated with the rotary trim actuator.

FORCE TRIM
The Force Trim clutch permits locking of the associated flight control line for hands-off flying.
The Force Trim is normally engaged, ie. with no electrical power the clutch is closed.

The Force Trim can be released (open clutch for hands-on flying) by either:

- Setting the FORCE TRIM switch (pitch and roll) or the CLTV/YAW TRIM switch (collective and yaw) on the MISC panel to OFF, or
- Pressing the FTR switch on the cyclic stick (pitch and roll), on the collective lever (collective) or on the pedals (yaw)

ARTIFICIAL FEEL DEVICE
A dual-action spring permits force-feel hands-on flying while the Force Trim is engaged: pilot moving the flight controls without releasing the Force Trim results in full hands-on controllability with control force feedback to the pilot.
As pilot releases the control to resume hands-off flying, the spring returns the relevant flight control line to its neutral position against the Force Trim.
Whenever the spring is out of its neutral position, a detent microswitch forces both Autopilots to suspend ATT mode operations: SAS function remains active.

DUAL POSITION SENSOR
The dual position sensor provides both AP computers with a feedback for closed-loop control.

FORCE TRIM – OPERATION

When the cyclic FORCE TRIM switch on the MISC panel is set to OFF, the ATT mode reverts automatically to SAS.
Returning the FORCE TRIM switch to ON, the ATT mode is automatically re-engaged.

In a hands-off flight, as long as the pilot holds an FTR switch pressed, on the relevant axis:

- The Force Trim clutch is disengaged
- Pilot is temporarily flying hands-on SAS
- The Linear Actuators are centered
- If a FD mode is engaged, the AP ignores the FD commands

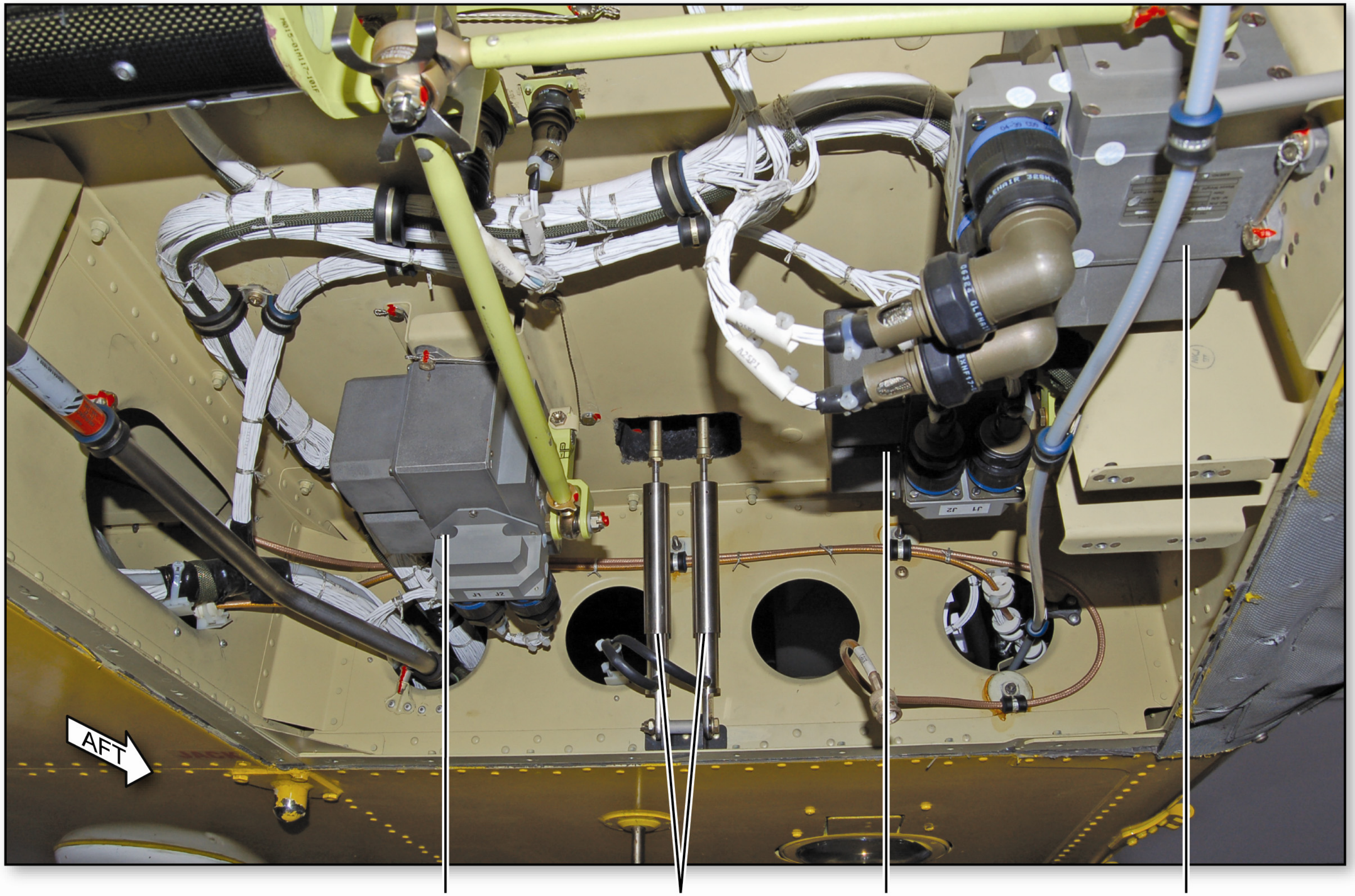
When pilot releases the FTR switch, on the relevant axis:

- The Force Trim clutch is re-engaged
- The ATT mode reference value is reset to the present attitude
- If a FD mode is engaged, the reference target parameter is reset to the present value (eg. IAS, VS, Radio Height, etc)
- Pilot returns to hands-off flight

When the Force Trim clutch is disengaged then:

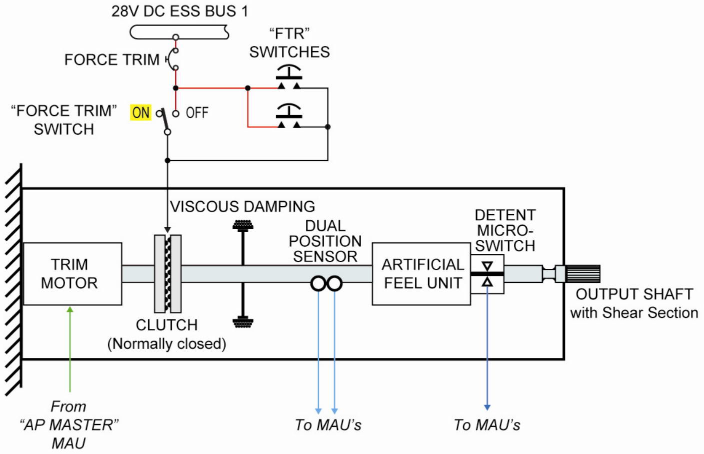
- the force feel system is disengaged
- the Trim Actuator drive is disengaged
- the Autotrim is disabled

Upon detection of the detent switch activation (Pilot moving controls without disengaging the Force Trim), the AFCS disables commands to the respective trim motors.

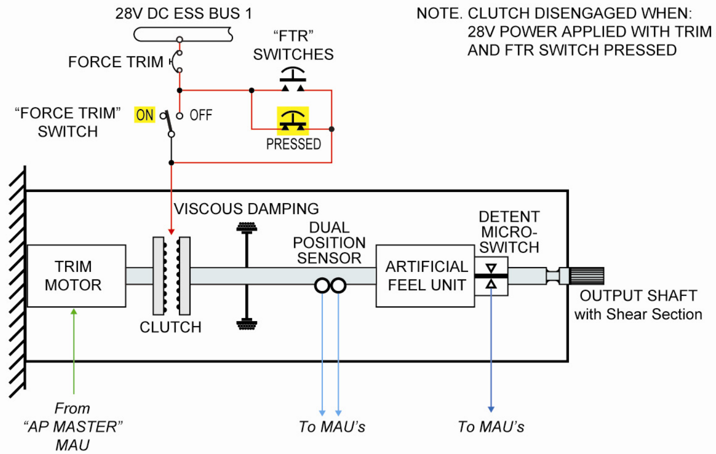


View looking up and aft of the bottom fuselage area under copilot seat

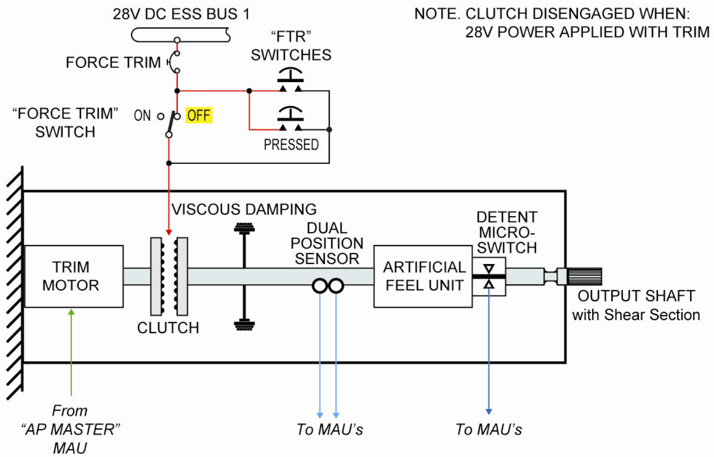
TRIM ACTUATORS



NOTE. CLUTCH DISENGAGED WHEN:
28V POWER APPLIED WITH TRIM ACTUATOR SWITCH = ON
AND FTR SWITCH PRESSED



NOTE. CLUTCH DISENGAGED WHEN:
28V POWER APPLIED WITH TRIM ACTUATOR SWITCH = OFF



NOTE. Pitch or Roll Trim Actuator is shown.
Collective and Yaw Trim Actuator circuits are similar.

TRIM ACTUATOR SCHEMATIC AND FORCE TRIM OPERATION

AFCS – CONTROLS AND INDICATORS

The following flight deck components constitute pilot interfaces for input to the AFCS:

- Autopilot Controller
- Guidance Controller (Flight Director control panel)
- Cyclic, pedal and collective Beep switches
- Cyclic, pedal and collective Force Trim Release (FTR) switches
- Cyclic, pedal and collective Force Trim enable switches on MISC panel
- AP1 & AP2 disconnect switch (SAS REL) on cyclic grips
- Remote Flight Director standby (FD STBY) button
- Go-Around (GA) button on Collective levers

Guidance reference inputs are also provided via:

- Display Controller (DC)
- Remote Instrument Controller (RIC)
- Cursor Control Device (CCD)
- Multifunction Display Control Unit (MCDU)

AFCS indications are provided on

- Autopilot Controller (status lights)
- Guidance Controller (status lights)
- PFD (annunciators, guidance cues)
- MFD (CAS messages and Synoptic)

If an AFCS problem is obvious from CAS cautions, failure indications or aircraft response, the autopilot controller should be used to deselect the faulty channel and the individual behaviour of AP 1 and AP 2 observed.

Illumination of the relevant Autopilot AP channel lights and the display of CAS captions should be used to make a positive diagnosis before, for example, disengaging an AFCS channel.

In case of an un-commanded aircraft disturbance or oscillation, occurring without an AFCS caution, the pilot should selectively disengage and re-engage individual channel in order to determine and isolate a potential non-annunciated AFCS fault. This can be achieved through the use of the autopilot channel pushbuttons (AP 1, AP 2) and monitoring of the trim display (select SYSTEM, FLIGHT CTRL on MFD to display the AFCS synoptic page) and aircraft response.

AUTOPILOT CONTROLS

AUTOPILOT CONTROLLER

1. AP 1 push-button

   PRESSED.......... Illuminates the green annunciator light and engages the AP no.1.

   Pushing the button again extinguishes the green annunciator light and disengages the AP no.1.

   NOTE. When the pilot engages AP 1 then:

   1) ATTITUDE mode is set as default mode

   2) the yaw control functions are activated

   3) the SAS mode is forced if the cyclic FORCE TRIM switch is set to OFF.

   The AP can operate both with and without FD guidance. In normal operation, both AP 1 and AP 2 are engaged in order to supply full dual system performance while coupled to FD.

2. AP 2 push-button

   PRESSED.......... Illuminates the green annunciator light and engages the AP no.2.

   Pushing the button again extinguishes the green annunciator light and disengages the AP no.2.

   NOTE. See NOTE for AP 1.

3. TEST push-button

   PRESSED.......... Illuminates the green annunciator light and starts the Built-In Test (BIT).

   Pushing the button again extinguishes the green annunciator light and exits the BIT.

4. CPL push-button

   NOTE. When a FD mode is engaged, FD automatically couples to AP and the CPL green annunciator illuminates.

   PRESSED.......... When green annunciator is illuminated, uncouples the FD from AP and the annunciator extinguishes.

Pushing the CPL button again illuminates the green annunciator light and manually re-couples the FD to the AP.

5. SAS push-button

PRESSED.......... Illuminates the green annunciator light and engages the SAS mode.

NOTE. The SAS and ATT buttons are mutually exclusive and are used to select the SAS or ATTITUDE mode of operation of the AFCS.

When the cyclic FORCE TRIM switch is set to OFF, the AP engages with the SAS mode active.

6. ATT push-button

PRESSED.......... Illuminates the green annunciator light and engages the ATTITUDE mode.

NOTE. The SAS and ATT buttons are mutually exclusive and are used to select the SAS or ATTITUDE mode of operation of the AFCS.

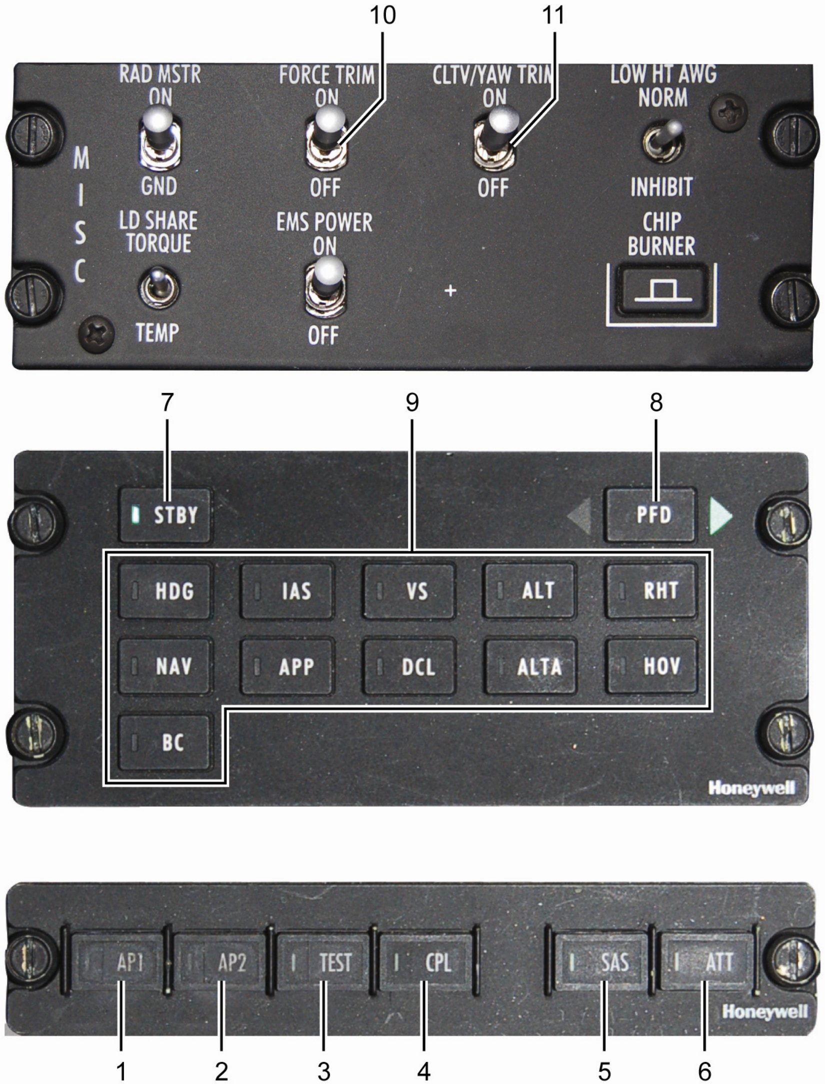
The ATT mode is automatically engaged if at least one AP is engaged and the cyclic FORCE TRIM switch is set to ON.

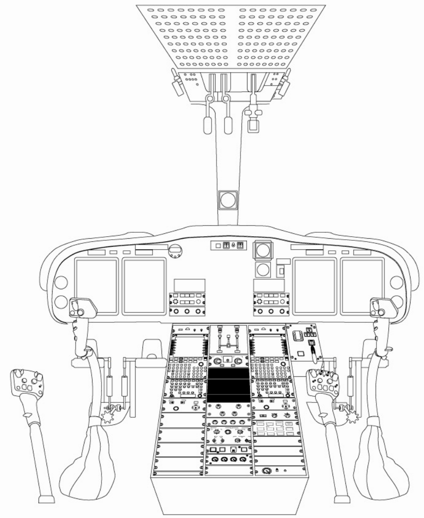
The ATT mode disengages if the pilot:

1) engages the SAS mode or

2) sets the cyclic FORCE TRIM switch to OFF or

3) disengages both AP 1 and AP 2.





AFCS CONTROLS – AUTOPILOT CONTROLLER

GUIDANCE CONTROLLER

7. STBY pushbutton

   PRESSED.......... Illuminates the green annunciator light and cancels any selected active flight director modes.

8. PFD pushbutton

   PRESSED.......... Selects which PFD (left or right) supplies source data that is used by both flight directors and toggles the associated green arrow annunciator located on each side of the PFD button. The illuminated arrow indicates the selected PFD.

9. FD Mode pushbuttons

   PRESSED.......... Illuminate the relevant green annunciator light and engage or arm the relevant Flight Director mode.

   Pushing any button again extinguishes the relevant green annunciator light and disengages the relevant Flight Director mode.

MISC CONTROL PANEL

10. FORCE TRIM switch

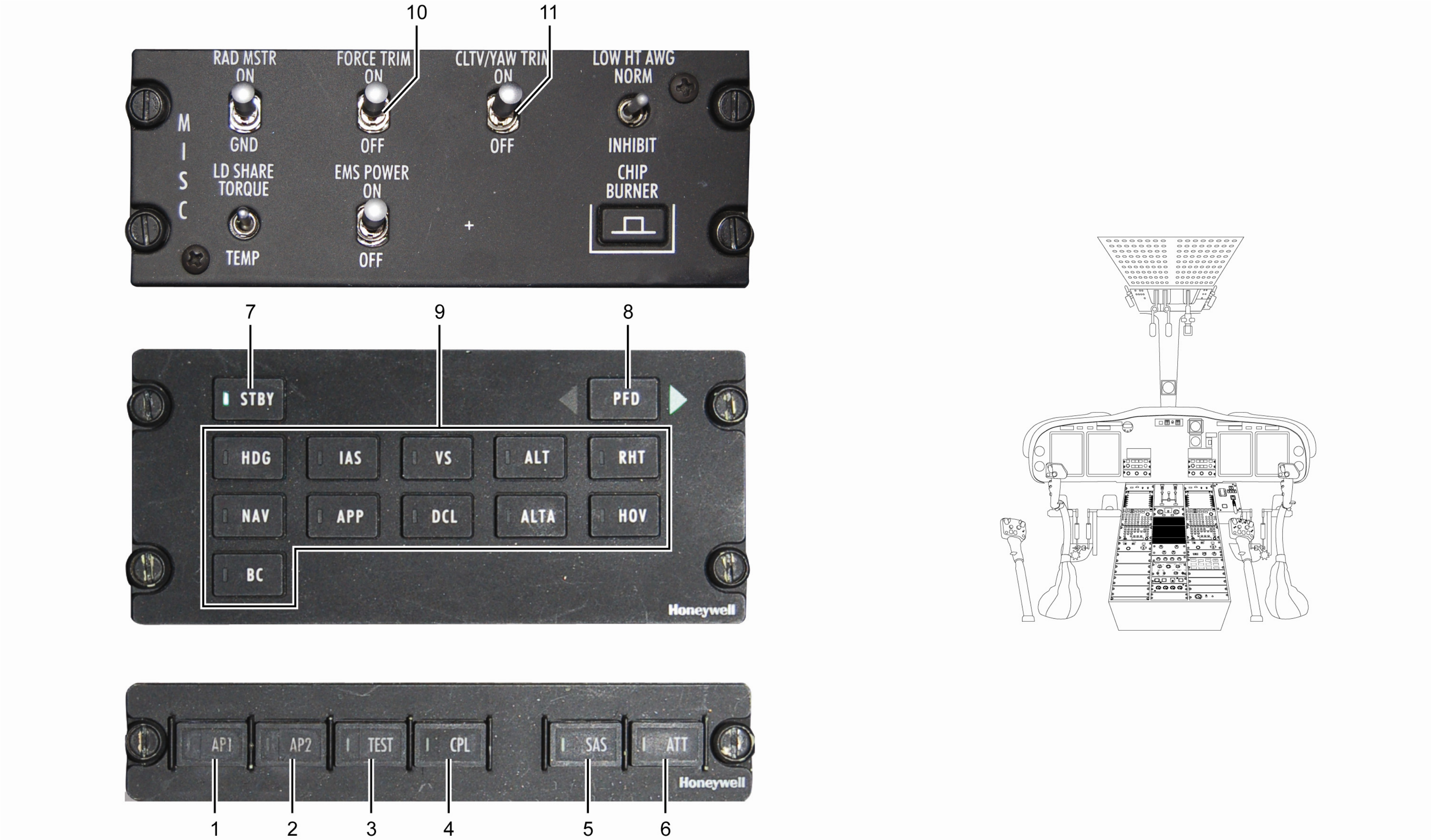
    OFF ……..……… Disengages the cyclic Force Trims (Pitch and Roll) and disables ATT mode of the AP

    ON ……………… Engages the cyclic Force Trims (Pitch and Roll) and enables ATT mode of the AP

11. CLTV / YAW TRIM switch

    OFF ……..……… Disengages the collective and pedal Force Trims

    ON ……………… Engages the collective and pedal Force Trims



AFCS CONTROLS – GUIDANCE CONTROLLER / MISC CONTROL PANEL

COLLECTIVE LEVER

12. BEEP CLTV / YAW TRIM switch (on PLT and CPLT collective grip)

DN / UP ………… Allows trimming the collective axis if the CLTV / YAW TRIM switch on Miscellaneous control panel is ON. Changes the reference target parameter value if a FD collective mode is engaged.

L / R ……………. Allows trimming the yaw axis if the CLTV / YAW TRIM switch on Miscellaneous control panel is ON

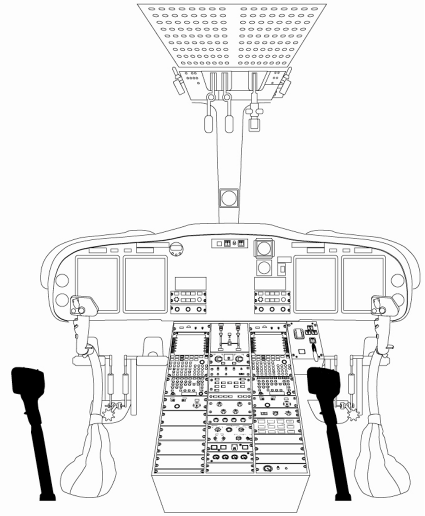
13. FTR (Force Trim Release) push-button switch (on PLT and CPLT collective grip)

PRESSED …….. Disengages the collective Force Trim suspending collective trimming and force feel

RELEASED …… Re-engages the collective Force Trim restoring collective trimming and force feel. If a FD collective mode is engaged, the reference target parameter is reset to the current value

14. GA (GO AROUND) mode push-button switch (on PLT and CPLT collective grip)

PRESSED …….. Engages the GA mode



AFCS CONTROLS – COLLECTIVE LEVER

CYCLIC STICK

15. BEEP TRIM switch (on PLT and CPLT cyclic grip)

DN / UP …………	Allows trimming the pitch axis if the FORCE TRIM switch on Miscellaneous control panel is ON and ATT mode is selected on AP
L / R …………….	Allows trimming the roll axis if the FORCE TRIM switch on Miscellaneous control panel is ON and ATT mode is selected on AP
PRESSED ……...	Engages the HOV mode on FD (4-Axis Enhanced FD only)

16. FTR push-button switch (on PLT and CPLT cyclic grip)

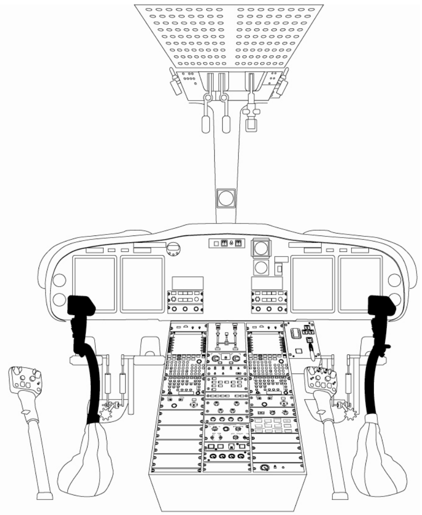
PRESSED ………	Disengages the cyclic Force Trims (Pitch and Roll) suspending collective trimming and force feel. Re-centers the Pitch and Roll Linear Actuators.
RELEASED …….	Re-engages the cyclic Force Trims (Pitch and Roll) restoring cyclic trimming and force feel. If a FD pitch or roll mode is engaged, the relevant reference target parameter is reset to the current value

17. FD STBY (Flight Director Standby) switch (on PLT and CPLT cyclic grip)

PRESSED ………	Illuminates the green annunciator light and cancels any selected active flight director modes

18. SAS REL push-button switch (on PLT and CPLT cyclic grip)

PRESSED ………	Disengages both Autopilots (AP 1 and AP 2)



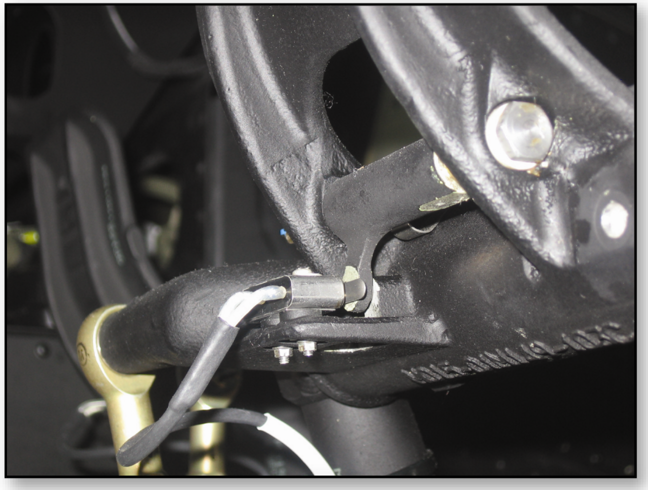
AFCS CONTROLS – CYCLIC STICK

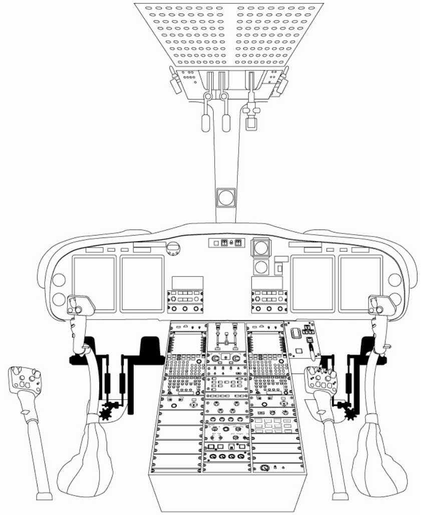
PEDALS

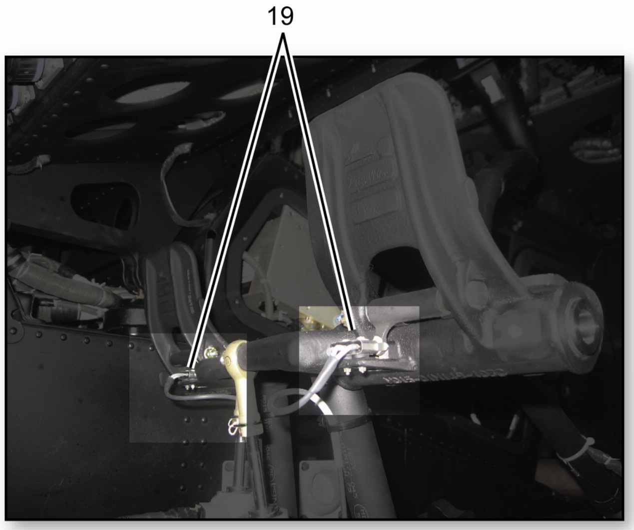
19. YAW force trim release switch (on PLT and CPLT pedals)

PRESSED ……… Disengages the pedal Force Trim (Yaw) suspending yaw trimming and force feel.
Re-centers the Yaw Linear Actuator

RELEASED ……. Re-engages the pedal Force Trims (Yaw) restoring yaw trimming and force feel







AFCS CONTROLS – PEDALS

AUTOPILOT INDICATORS

1. SAS annunciator

   In view.......... Indicates that AP is in SAS mode

2. UCPL annunciator

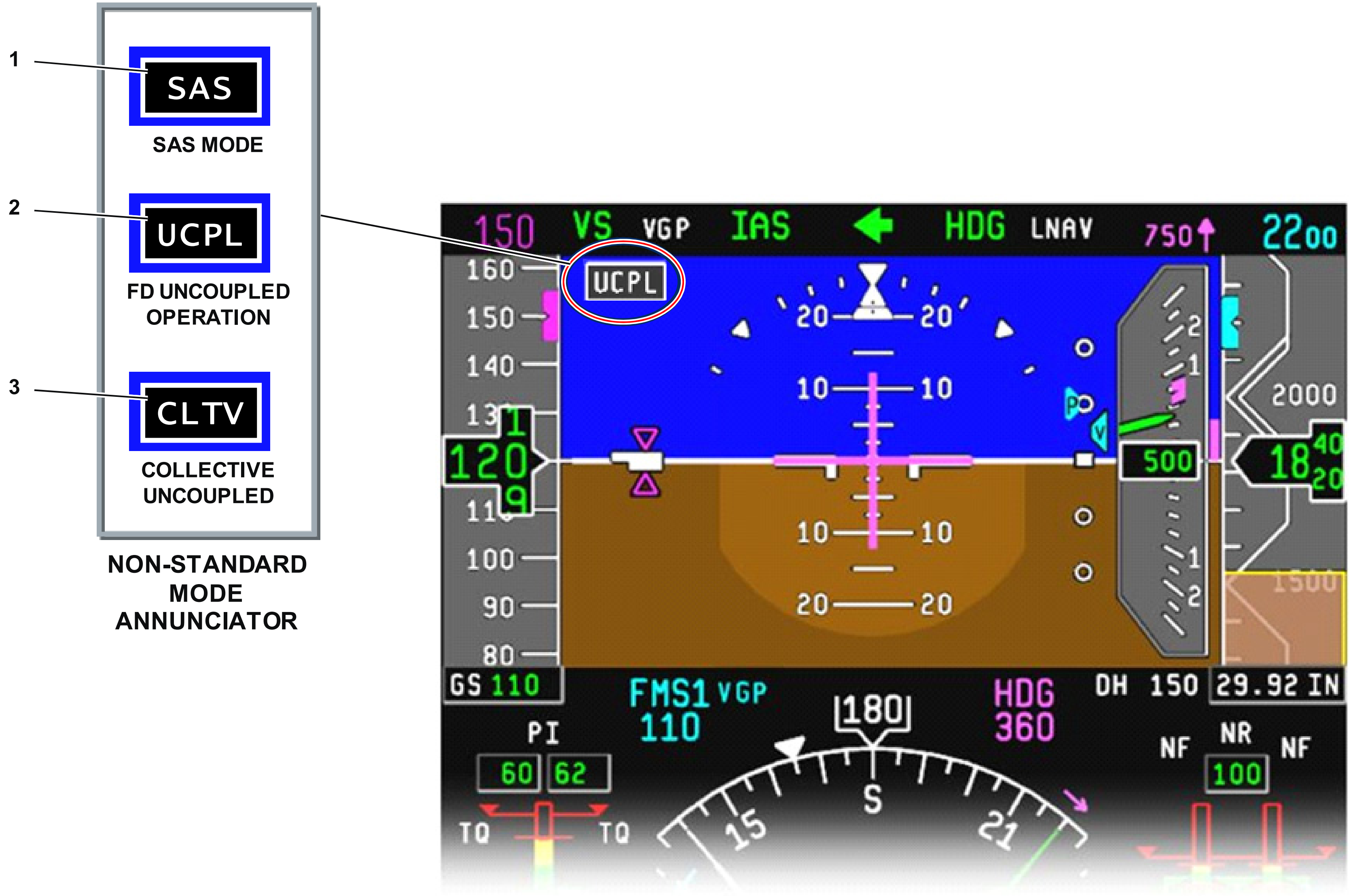
   In view.......... Indicates that Flight Director is uncoupled (all axes) from the AP

   NOTE. The UCPL annunciator is not in view when the AP is in SAS mode

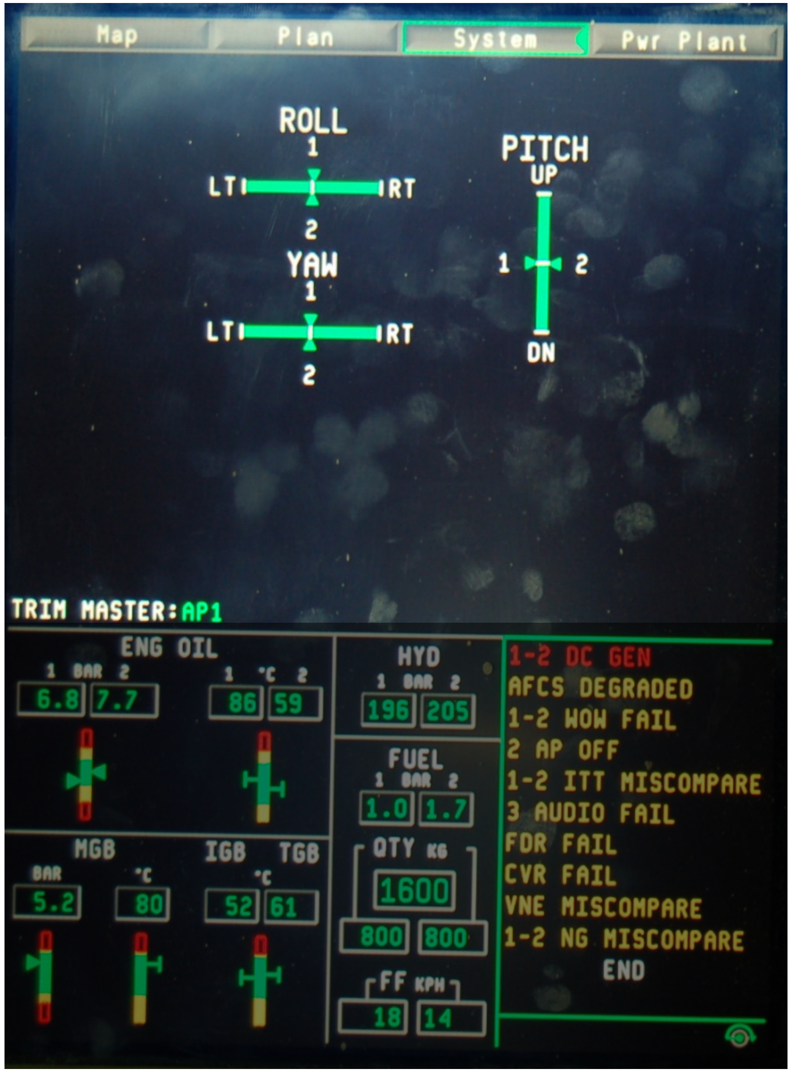
3. CLTV annunciator

   In view.......... Indicates that Flight Director is uncoupled from the AP on the Collective axis only

   NOTE. The CLTV annunciator is not in view when the AP is in SAS mode or FD is uncoupled on all axes



AUTOPILOT INDICATORS



AFCS SYNOPTIC PAGE

AFCS FUNCTIONS

The AFCS provides the following functions that are available also in case of single AP or single FD operation:

- Pre-flight test (TEST)
- Hands-off Attitude Hold (ATT)
- Hands-on Stability Augmentation System (SAS)
- Yaw Control
- Collective Control
- Autotrim
- Trim Priority Function (Trim Master)
- Flight Director (FD) commands and coupling for automatic flight (including Torque Limiting)
- System and sensors monitoring

ATTITUDE HOLD (ATT) MODE

The ATT mode is automatically selected when an Autopilot is engaged (default at engagement).
It provides long-term pitch and roll attitude retention for hands-off flying and for Flight Director coupling in addition to providing stabilization (SAS function).

In the ATT mode the AP pitch and roll Force Trims must be engaged (FORCE TRIM switch on MISC panel at ON).
The pitch and roll attitude to be retained is memorized as that at the time of engagement.

Changes to the selected attitude can be made by:

- pressing the BEEP TRIM switch on the cyclic stick (typically for small changes), or
- pressing the Force Trim Release (FTR) switch on the cyclic stick and flying manually the helicopter to attain the desired attitude, then releasing the FTR to return to hands-off.

Pitch attitude beep rate is ±2°/s for airspeeds lower than 120 KIAS and ±1°/s for airspeeds higher than 140 KIAS.
With the airspeed between 120 and 140 KIAS, the pitch changes linearly between ±2°/s and ±1°/s.
Roll attitude beep rate is ±3°/s regardless of airspeed.

The pilot can override the AFCS ATT mode at any time by taking over the controls manually: if the FTR switch is not pressed, the detent switch inside the Trim Actuator permits “flying through the AP” without changing the selected attitude. When the helicopter is returned to hands-off the ATT mode brings the helicopter back to the memorized attitude.

In ATT mode, the AP pitch and roll channels can be coupled to the Flight Director (FD): in that case the selected attitude is changed by the FD commands.

When the aircraft is flying at high speed (IAS > 45 KIAS) with no FD roll mode engaged, the ATT mode holds roll attitude only if it is selected to be greater than 3°.
If the selected roll attitude is less than 3°, AP forces the roll attitude to 0° (wings level) and holds current aircraft heading

through the yaw control channel (High Speed Wings-Level Yaw Heading Hold function).

STABILITY AUGMENTATION SYSTEM (SAS) MODE

The SAS improves the handling characteristics of the helicopter by damping the effects of the short-term external aircraft disturbances on pitch, roll and yaw axes and improves the controllability during low-speed manoeuvring or hovering flight.
The SAS function is active whenever the AP is engaged, either in ATT or in SAS mode.
SAS mode is intended for use where extensive aircraft manoeuvring is required and the pilot prefers to be hands-on without attitude retention.

SAS mode is selected by:

- pushing the SAS button on the auto-pilot controller, or
- setting the FORCE TRIM switch on Miscellaneous panel to OFF.

When SAS mode is selected, being it a hands-on control mode:

- the Autotrim is disabled
- the AFCS can be operated with FORCE TRIM switch either ON or OFF
- the ATT OFF caution message is displayed in the CAS window and the SAS annunciator is displayed in the ADI (PFD)

Note: Each autopilot uses the on-side AHRS for both ATT and SAS mode computations; failure of an AHRS causes the disengagement of the on-side Autopilot.

YAW CONTROL

Autopilot Yaw Control channel is engaged automatically when an AP is selected and remains active in both ATT and SAS modes.

It provides airspeed-switched yaw coordination as required to maintain proper high and low speed directional control.
A hysteresis separates the low speed and high speed operating ranges: high speed operation is entered accelerating above 45 KIAS, low speed operation is entered decelerating below 41 KIAS.

The yaw control provides the following functions:

- At any speed:
  - Yaw Rate Damping
  - Collective-to-Yaw Cross-feed
- At low airspeed (IAS < 41 KIAS):
  - Low Speed Yaw Heading Hold
- At high airspeed (IAS > 45 KIAS):
  - High Speed Turn Coordination
  - Lateral Ball Trim
  - High Speed Wings-Level Yaw Heading Hold

YAW RATE DAMPING
Yaw Rate Damping (SAS Yaw channel) is active whenever the Autopilot is engaged; it provides stability augmentation about the yaw axis at all airspeeds and helps preventing the dutch roll tendency of the helicopter.

COLLECTIVE-TO-YAW CROSS-FEED
Collective-to-Yaw Cross-feed function is active whenever the Autopilot is engaged; it automatically compensates any collective position changes to counteract the effect of torque when the collective control is moved.

LOW SPEED YAW HEADING HOLD
Low Speed Yaw Heading Hold function automatically engages at low speed (IAS < 41 KIAS) if roll attitude is less than 3°, or when HOV mode is engaged on FD.
Yaw force trim must be engaged (CLTV/YAW TRIM switch on the Miscellaneous panel is at ON) for the function to operate.
Once the Low Speed Yaw Heading Hold function is active, it does not disengage in case roll attitude exceeds 3°.
The low speed yaw heading hold reference is established at the time this function becomes active and is indicated with a small magenta low speed heading bug displayed on the HSI.
Changes can be performed by repositioning the aircraft acting on the pedals or using the YAW beep switch on the collective grips (±3°/s).

HIGH SPEED TURN COORDINATION
High Speed Turn Coordination function operates at high speed only (IAS > 45 KIAS) during roll controlled turns to minimize slip or skid. A yaw rate command supplies this function from the calculated aircraft turn rate as a function of the bank angle and airspeed.

LATERAL BALL TRIM

Lateral Ball Trim function operates at high speed only (IAS > 45 KIAS); it allows the pilot to keep a constant slip or skid, either to offset a small accelerometer misalignment or to purposely offset the tail alignment with respect to the aircraft horizontal flight path.

Yaw force trim must be engaged (CLTV/YAW TRIM switch on the Miscellaneous panel is at ON) for the function to operate.

The magnitude of the slip or skid is set using the YAW beep switch on the collective grips (±0.5 ft/s²/s).

HIGH SPEED WINGS-LEVEL YAW HEADING HOLD

The High Speed Wings-Level Yaw Heading Hold function is active when the aircraft is flying at high speed (IAS > 45 KIAS) with no FD roll mode engaged and the selected roll attitude is less than 3°: the function automatically reduces the roll attitude to 0° (wings level) and then holds new aircraft heading through the yaw channel.

Yaw channel is also used to correct for heading errors when HDG mode is engaged on FD (coupled), if the roll attitude is less than 0.5° and heading error is less than 2°.

The roll attitude is commanded to wings level while yaw is driving heading.

COLLECTIVE CONTROL

The Collective Control function is activated by selecting 3-cue FD operation or by activating a collective only mode (RHT) while the CLTV/YAW TRIM switch on the Miscellaneous panel is at ON.

The Collective Control function provides automatic vertical control of the helicopter by means of a rotary Collective Actuator connected in parallel with the collective mechanical control system.

When the CLTV/YAW TRIM switch is turned OFF, the collective (and yaw) Force Trims are disengaged.

Changes to collective reference settings can be performed by repositioning the collective lever with the collective FTR pressed or using the CLTV beep switch on the collective grips.

The Collective Control function is disabled by deselecting the mode, uncoupling the FD from the AP or selecting the SAS mode.

The AP that is controlling the collective actuator is the Trim Master.

The Collective Control function limits collective position to prevent engine or mast torque exceedances. This limiting function will only be active when the automatic collective control function is engaged.

AT ANY SPEED

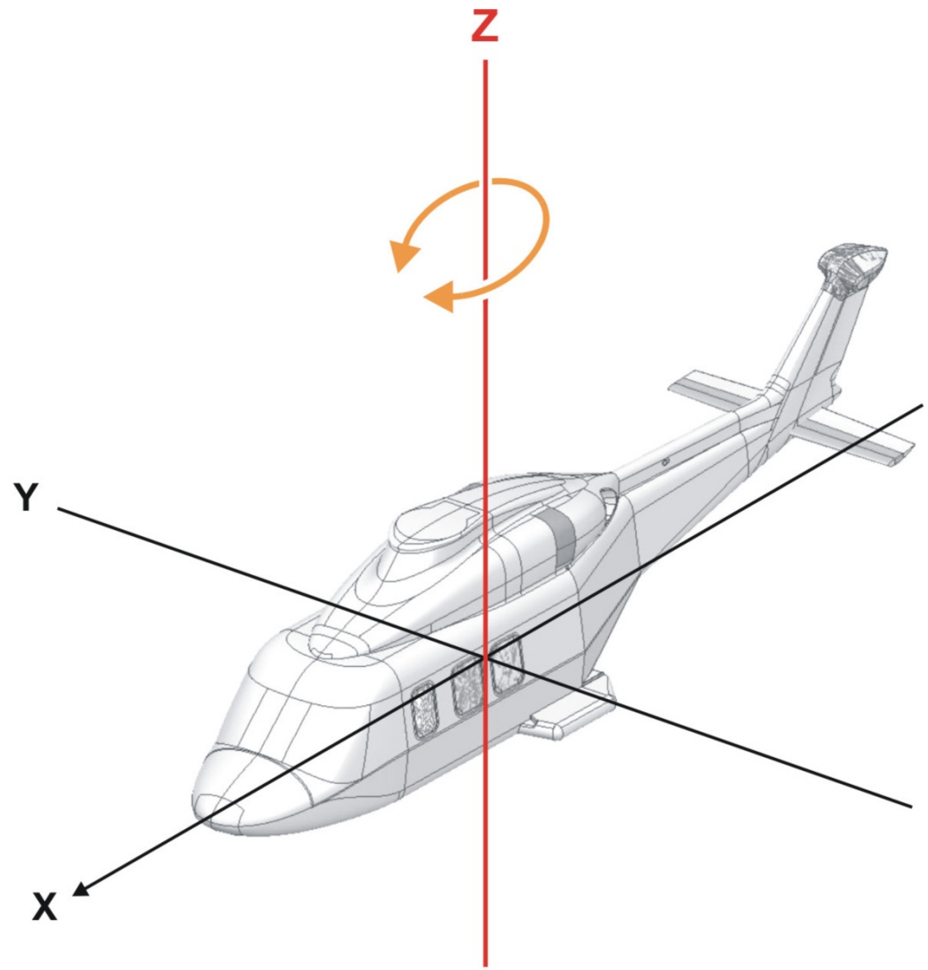
- Yaw Rate Damping (Stabilization)
- Collective-to-Yaw Cross-feed

AT LOW SPEED (IAS < 41 KIAS)

- Low-Speed Yaw Heading Hold

AT HIGH SPEED (IAS > 45 KIAS)

- Turn Coordination
- Lateral Ball Trim
- High Speed Wings-Level Yaw Heading Hold



YAW CONTROL FUNCTIONS

AUTOTRIM

The purpose of the automatic trim (AUTOTRIM) function is to keep the average position of the output shaft of the roll, pitch and yaw linear actuators near center to ensure optimum Linear Actuator authority in response to aircraft control commands; the AP Autotrim function commands the Trim Actuators to achieve that.
The Autotrim function on each axis is active when either or both Autopilots are engaged in ATT mode and the relevant axis Force Trim is engaged.
The Autotrim control reference is the Linear Actuator displacement from the center: when the displacement exceeds a percentage of the Linear Actuator stroke, the Trim Motor is commanded at an appropriate rate in the direction to re-center the Linear Actuator (different percentage values are set for pitch, roll and yaw axes).

TRIM PRIORITY FUNCTION (TRIM MASTER)

The Trim Priority Function has the purpose to provide and retain the Trim Master operation on controlling the Trim and Collective actuators.
For normal operation, when both autopilots are engaged, the autopilot that was engaged first retains the Trim Master function.
The indication of what AP is elected as Trim Master is shown in the Flight Controls synoptic page of the MFD.
As the Trim Master AP is disengaged (either manually or automatically due to a failure), the remaining AP is elected Trim Master.
The Trim Master function switching logic is overridden if one or more linear actuators fail.
A set of relays in the aircraft wiring are used to ensure that only the Trim Master AP provides commands to the pitch, roll, yaw and collective trim actuators so that they cannot be corrupted by the other AP.
Similarly, the trim end-around feedback is provided to identify trim and collective actuator movement and direction.

PRE-FLIGHT TEST (TEST)

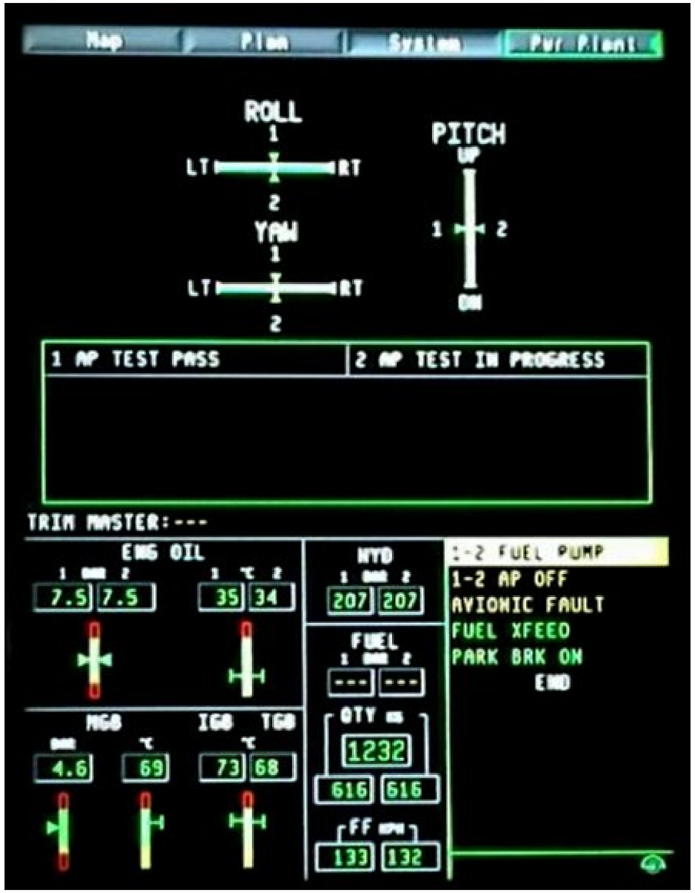
Pre-flight test allows verifying autopilot control and monitoring functionality.
During the pre-flight procedure, as part of the System Checks, pilot is required to perform the AFCS TEST that is initiated by pressing the TEST button on the Autopilot Controller with the following conditions:

- Aircraft on the ground (WOW)
- Electrical and hydraulic power available
- Autopilots disengaged
- Hands off the flight controls (Force Trims on)
- Collective down (<10%)

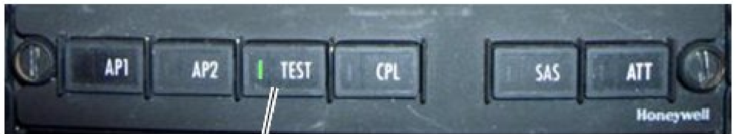
During the self-test, the each AP moves both Linear Actuators and Trim Actuators in opposite directions. This limits the amount of actual flight control movement that occurs during the test.

The pre-flight test verifies autopilot control and monitor functions reducing the possibility of having latent failures in critical system components.

The result of the test is displayed on the Flight Controls synoptic page (that is automatically selected on pilot MFD) and in the CAS window.
Refer to Section 2 of the RFM for the AFCS TEST procedure.



SAS REL SWITCH



TEST BUTTON

AFCS PRE-FLIGHT TEST

PRE FLIGHT TEST ANNUNCIATORS

While this test is running, the FLIGHT CONTROL synoptic page displays a text box containing one or more messages listed below in case of an improper operational condition:

Message Wording	Message Conditions
ACT CPLT POWER FAIL	Copilot's linear actuators are not powered
ACT PLT POWER FAIL	Pilot's linear actuators are not powered
1(2) AUTOTRIM POWER FAIL	Trim actuators are not powered
1(2) HYDRAULIC PRESSURE INVALID	Hydraulic pressure is not in normal operating range
1(2) COLLECTIVE TOO HIGH	Collective positive is too high
1(2) TRIM OFF	Cyclic trim switch is off
1(2) YAW TRIM OFF	Yaw trim switch is off
1(2) COLLECTIVE OFF	Collective switch is off
1(2) FTR ACTIVE	Cyclic FTR switch is pressed
1(2) YFTR ACTIVE	Yaw FTR switch is pressed
1(2) CFTR ACTIVE	Collective FTR switch is pressed
1(2) PITCH OUT OF DETENT	Pitch trim actuator is being held out of detent
1(2) ROLL OUT OF DETENT	Roll trim actuator is being held out of detent
1(2) YAW OUT OF DETENT	Yaw trim actuator is being held out of detent
1(2) COLLECTIVE OUT OF DETENT	Collective actuator is being held out of detent
1(2) PRESS AND RELEASE SAS REL SWITCH	Pre Flight test request for pilot action
1(2) SAS REL SW FAILURE	Pre Flight test did not detect pilot press of the SAS REL switch

Message Wording	Message Conditions
1(2) SAS RELEASE SWITCH ACTIVE	Pre Flight test detected a press of the SAS REL switch when not expected
TEST WAITING	Displayed while waiting the pilot to press the SAS REL switch
TEST IN PROGRESS	Displayed when on-side AFCS is executing the actuator tests
TEST COMPLETE	Displayed when all tests have been completed
TEST STANDBY	Displayed while waiting for the cross-side AFCS to complete actuator tests
TEST INHIBIT	Displayed when test is halted due to an inhibit condition
TEST INVALID	On-side AFCS is invalid, while cross-side AFCS is running test

AUTOPILOT – OPERATION

ENGAGING AND DISENGAGING THE AUTOPILOTS

The autopilots engage and disengage by pushing the AP1 and/or AP2 buttons on the autopilot controller.
Simultaneous disengagement of both AP 1 and AP 2 can also be commanded by pressing the SAS REL push-button on either cyclic stick.

ATT is the default mode when the autopilots are engaged. However, the autopilots are forced to SAS mode when the cyclic trim enable switch is OFF.

To prevent undesirable transients in the controls at engagement or disengagement, an easy-on/easy-off function drives the pitch, roll and yaw Linear Actuators gently back to center.
After disengagement and actuator centering, the 28-Volt brake excitation is removed, locking the Linear Actuators in the center position.

The first autopilot engaged is elected Trim Master and takes control of the four Trim Actuators by switching on or off a set of relays.
As the Trim Master AP is disengaged (either manually or automatically due to a failure), the remaining AP is elected Trim Master.

NORMAL OPERATION

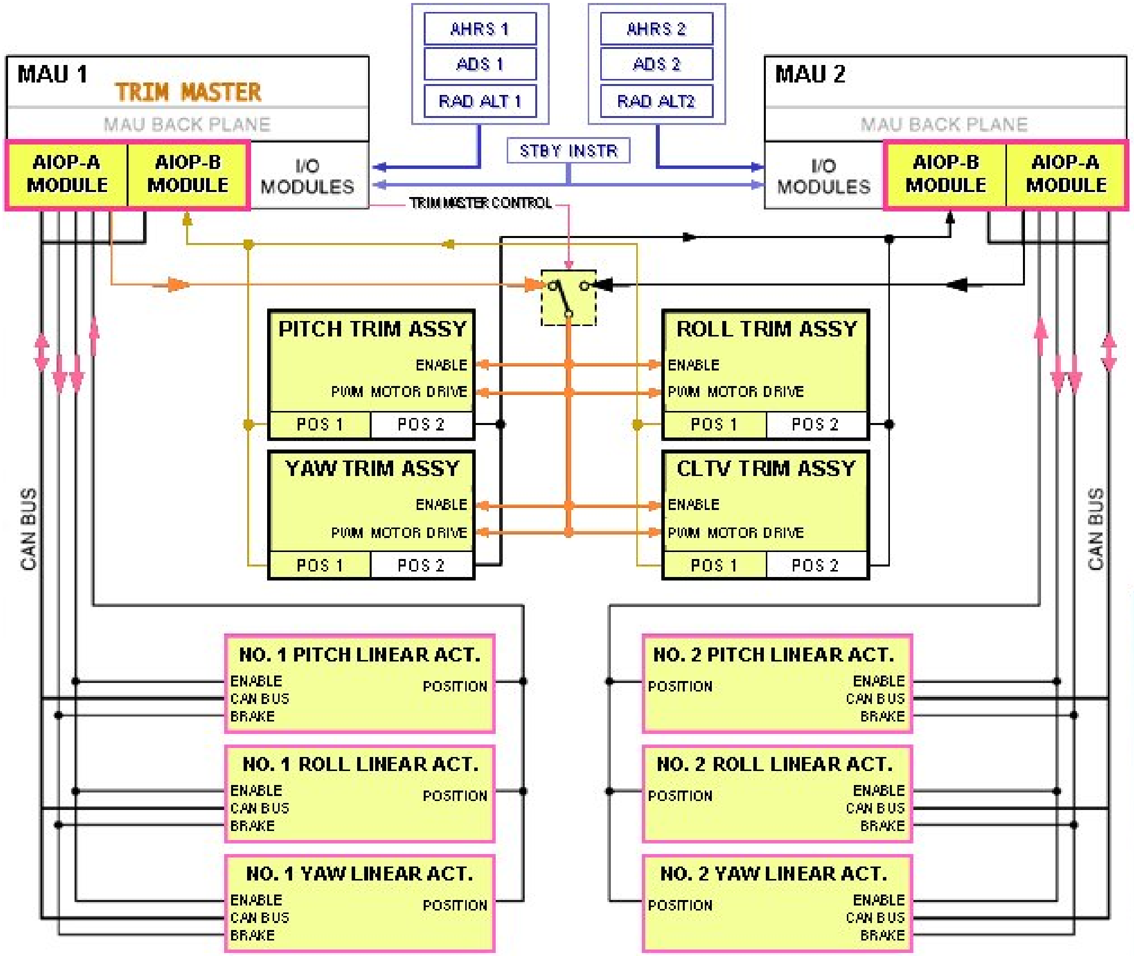
In normal operation, each autopilot supplies half the total linear actuator system authority.
Engaging the AP activates the automated yaw control function and the Autotrim function.

The autopilot control authority as supplied by the dual series Linear Actuators is 20% of the full aircraft control authority for each axis (each section of a Linear Actuator providing 10% of the full aircraft control authority).
Flying manually, the pilot has the same control authority over the aircraft when the autopilot is engaged in SAS or ATT modes, or when the autopilot is disengaged.

AP 1 and AP 2 operate either with or without the flight director guidance modes active.
In normal operation, both autopilots are engaged to supply full dual system performance when coupled to the flight director.

The on-side AHRS is the main reference for AP operation; in case an unflagged miscompare occurs between the two AHRS, the electronic Standby Instrument input is used by the MAU to determine which AHRS has failed.

The on-side ADS is the reference for AP gain computations; in case of ADS dual failure or in case an unflagged miscompare occurs between the two ADS, a fixed airspeed value is used by the AP.



AUTOPILOT OPERATION

AUTOPILOT FAILURE

When detecting failure affecting mode integrity, the AP of the affected single system disengages.
The following events also disengage the on-side AP on the corresponding AHRS:

- Invalid on-side attitude data
- Invalid on-side attitude rate data
- Invalid on-side yaw rate data

Failure of an AP is annunciated by CAS messages and the AUTOPILOT-AUTOPILOT aural warning message.

SINGLE AUTOPILOT OPERATION

With only one autopilot engaged, the single system operates at full gain with half the dual system authority. When both autopilots are engaged, each single system gain is reduced to 50% so that each system supplies half of the required input. This results in full gain control with twice the single system authority.
Full yaw control and Autotrim functions are available also in single system operation.

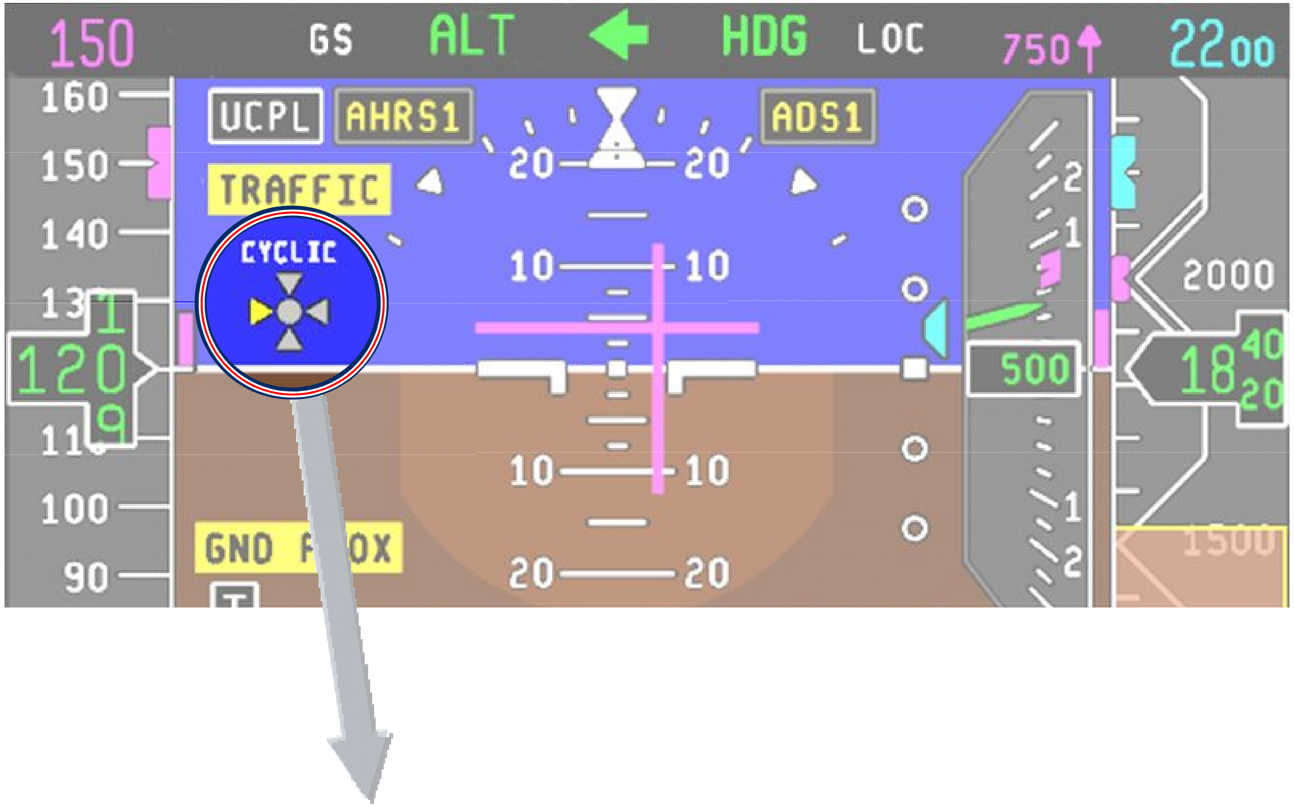
AUTOPILOT OVERRIDE

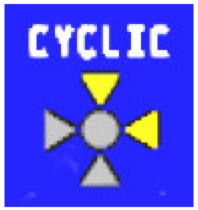
The pilot has full authority with the AP engaged or disengaged and can immediately override the AFCS at any time by simply taking over the controls.

With the autopilots engaged, FTR switches, AP disengage switches (SAS REL), detent switches and actuator circuit breakers offer different ways to partially or completely override the AFCS.

CYCLIC POSITION INDICATOR

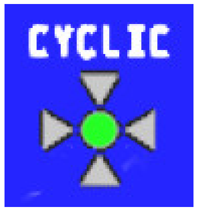
The cyclic position indicator is displayed on the ADI to help pilot center the cyclic controls before starting the engines, to ensure that the main rotor does not hit the static stops when rotating at low speed.
The indicator is only displayed when the helicopter is on the ground (WOW) with collective down (LVDT signal via EECs).
Indication is taken from the position sensors located inside the pitch and roll Trim Actuators.
The cyclic is centered when the indicator shows a green dot; amber arrows indicate which direction the cyclic stick must be moved to center the controls.





CYCLIC NOT CENTERED

Eg: Cyclic must be moved back and to the left



CYCLIC CENTERED

NOTE: Displayed only when:
- On the ground (WOW)
- Collective Lever down

CYCLIC POSITION INDICATOR

AUTOPILOT FAILURES – CAS CAUTION MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
1(2) AP FAIL + aural message + **AFCS DEGRADED**	Associated autopilot failure	AUTOPILOT FAIL	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES
1(2) AP P(R)(Y) FAIL + aural message	Pitch (Roll) (Yaw) axis of AP 1(2) uncommanded disengagement	AUTOPILOT AXIS DISENGAGE	
1(2) AP OFF + aural message + **AFCS DEGRADED**	Associated AP not switched ON	AUTOPILOT OFF	
1(2) AP P(R)(Y) OFF + aural message	Pitch (Roll) (Yaw) axis of AP 1(2) not engaged	AUTOPILOT AXIS OFF	

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
ATT OFF	Attitude system not engaged or Cyclic force trim failure if associated with cyclic freedom of motion in longitudinal and/or lateral direction with loss of function of FTR switch and cyclic beep trim. NOTE: With ATT system not engaged the aircraft flies in SAS mode only (SAS message on PFD)	ATTITUDE SYSTEM OFF and CYCLIC FORCE TRIM FAILURE	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES
1(2) TRIM FAIL	Trim system failure on all axes while AP 1(2) has priority (Trim Master)	AFCS TRIM FAILURE	
1 P(R)(Y) TRIM FAIL	Failure of Pitch (Roll) (Yaw) trim while AP 1 has priority	PITCH, ROLL, YAW TRIM FAIL	
2 P(R)(Y) TRIM FAIL	Failure of Pitch (Roll) (Yaw) trim while AP 2 has priority	PITCH, ROLL, YAW TRIM FAIL	
MISTRIM	Linear actuator(s) not centered	MISTRIM	
1(2) SAS DEGRADED	Associated SAS degraded operation	SAS DEGRADED	
AFCS DEGRADED	Associated SAS degraded operation	SAS DEGRADED	

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
1(2) AP TEST ABORT	AP TEST aborted by pilot action or aircraft lifted off before test completion	AP TEST ABORT	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES
1(2) COLL FAIL + aural message	Collective axis of AP1(2) fail NOTE: Collective FD mode annunciator amber (if a coupled collective mode is active)	SINGLE COLLECTIVE AUTOPILOT FAILURE	Supplement 34 Supplement 40 — Section 3 EMERGENCY AND MALFUNCTION PROCEDURES
1-2 COLL FAIL + aural message	Collective axis of AP1(2) failure on both AP 1 and AP 2 NOTE: Collective FD mode annunciator amber (if a coupled collective mode is active)	DUAL COLLECTIVE AUTOPILOT FAILURE	

COLLECTIVE FORCE TRIM OFF OR FAIL

When a FD collective mode is engaged and coupled and the collective trim is switched OFF (CLTV/YAW TRIM switch on the Miscellaneous Control Panel) or fails:

- a chime sound is generated
- the CLTV annunciation illuminates on the top left of the ADI display
- the CLTV/YAW OFF green advisory illuminates on the CAS
- Collective modes are available uncoupled only

CYCLIC FORCE TRIM FAILURE

Cyclic force trim failure is a disconnection of the longitudinal and/or lateral clutches. The failure is usually joined to the caution

ATT OFF

and the cyclic moves freely in pitch and/or roll axis with loss of function of the cyclic trim release (FTR switch) and cyclic beep trim system.

In these conditions the cyclic must be used hands-on to prevent it from moving away the selected position.

AFCS QUICK DISCONNECT PROCEDURE

For situations where faults are suspected in the AFCS, but with no CAS cautions illuminated, and the AP functions need to be disengaged, all AP/AFCS functions can be disconnected by pressing the SAS REL button on the cyclic grip.

4-AXIS FLIGHT DIRECTOR – GENERAL

The dual Flight Directors (FD1 & FD2) provide lateral and vertical guidance commands that are normally coupled to the Autopilots for automatic flight path control.

The 4-Axis (3-cue) FD provides lateral modes operating on the roll axis and vertical modes operating on the pitch and collective axes.

The 4-Axis Enhanced Flight Director provides all the functions of the 4-Axis Basic Flight Director plus the Auto Hover/Velocity Hold mode (HOV).
References:

- 4-Axis Basic FD: RFM Supplement 40
- 4-Axis Enhanced FD: RFM Supplement 34

The 4-Axis Enhanced FD requires the following optional equipment to be installed in place or in addition to the Basic FD:

- AHRS LCR-93 (in place of LCR-92) to provide AFCS with hybrid Along-Heading and Across-Heading velocities and accelerations (in addition to LCR-92 data)
- 5-way cyclic BEEP TRIM switches (in place of 4-way switches)

FD MASTER

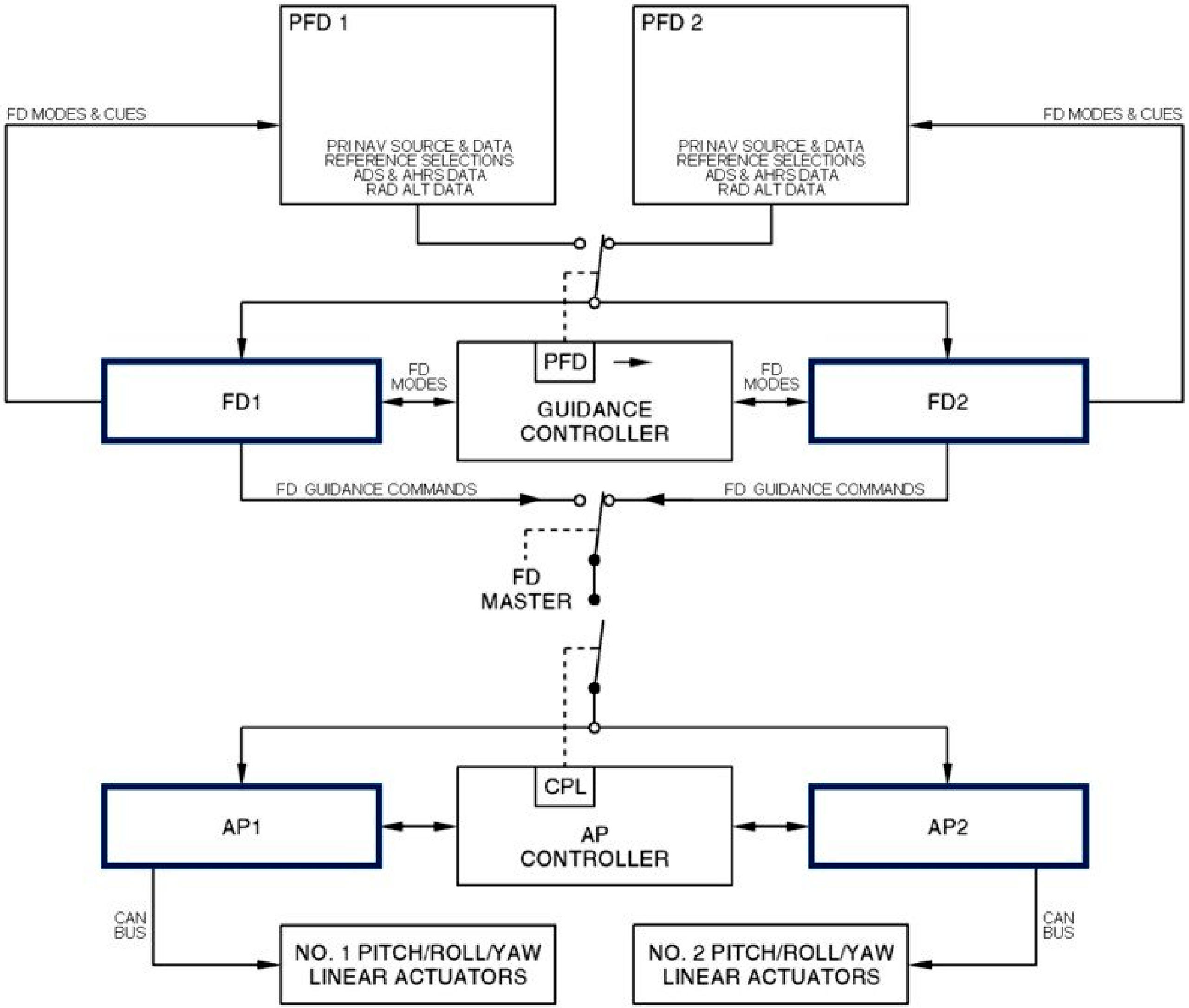
At power up one of the two FD is automatically selected and configured as Master (priority). At every power up the FD selected as Master is alternated: the selection is not visible to the pilot.

PFD SELECTION (PILOT-IN-COMMAND)

Both FD1 and FD2 use the navigation source and data reference presented on the selected PFD.
PFD selection (which typically matches the "Pilot Flying" or "Pilot in Command" determination) is controlled by the PFD pushbutton on the Guidance Controller.
The selected PFD is indicated by the PFD couple arrow in the center of the top line of both PFD's and by either the LH or the RH arrow-shaped LED annunciator aside the PFD pushbutton on the Guidance Controller.

Whenever the PFD pushbutton is pressed to toggle LH/RH PFD selection all FD modes are disengaged (STBY illuminates on the Guidance Controller).

FD modes remain engaged in case the selected PFD fails: the paired MFD reverting in Composite format keeps operating as the selected PFD.



FLIGHT DIRECTOR FUNCTIONAL DIAGRAM

4 AXIS FLIGHT DIRECTOR - OPERATION

ENGAGING AND DISENGAGING FD MODES

Flight Director modes are engaged and disengaged via the Guidance Controller (FD Control Panel). Each mode pushbutton green light illuminates when the mode is armed or engaged (captured).

The engagement of the FD modes is made with the Guidance Controller (GC) and the communication to the AIOP modules is through the CIO modules.

Each mode may be deselected by pressing the associated pushbutton on the Guidance Controller.
All engaged modes may be deselected simultaneously by using the STBY button on the Guidance Controller or the FD STBY button located on pilot or copilot cyclic stick grips.

GA (Go-Around) mode can only be engaged by using the GA button on pilot or copilot collective lever and disengaged by engaging a different vertical mode or setting FD to STBY.

Armed and captured FD modes are displayed with messages along the top line of the PFD:

- collective and pitch mode messages on the left
- lateral mode messages on the right

Armed modes are shown in white small font; captured (or engaged) modes are shown in green medium font.

When transitioning from no mode to engaged, from armed to captured or change from one mode to another mode a chime sounds and the message flashes normal/reverse video for 6 seconds then becomes steady (normal video).
When changing from captured to no mode a chime sounds and the message flashes on and off for 6 seconds then extinguishes.

Pitch and roll command bars and/or collective cue are displayed on the PFD by the AFCS when a Flight Director mode is active on the relevant channel.
When there is no FD mode active the command bars/ collective cue are not displayed.

When HOV mode is the active flight director mode, the pitch and roll command bars are not displayed.

All modes are disengaged when toggling the PFD button on Guidance Controller.

All modes - except HOV, ALT and RHT - must not be engaged below 60 KIAS and are automatically disengaged when airspeed reduces below 55 KIAS. HOV, ALT and RHT can be engaged between 0 KIAS and Vne.

All modes have a Minimum Use Height limit: see RFM Supplement 40 (or 34), Section 1 – Limitations.

FD COUPLING AND UNCOUPLING

When a FD mode is engaged (captured) the coupled (CPL) function is also automatically engaged if the AP is in ATT

mode and the relevant Force Trim is engaged: the CPL pushbutton green light on the Autopilot Controller illuminates.
When coupled, both autopilots drive their actuators to satisfy the references provided by the priority FD.
Selecting a FD mode by pressing the pushbutton on the Guidance Controller with ATT OFF (i.e. SAS mode or both Autopilots off) will engage the FD in the Uncoupled mode.

The FD may be manually uncoupled by pressing the CPL pushbutton on the Autopilot Controller when illuminated: the CPL green light extinguishes and the UCPL annunciator appears on both PFD's; on the ADI the command bars and/or collective cue remain in view and the pilot can manually fly the commands.

When the AP reverts to SAS no UCPL caption will appear on the PFD.

In either case (Coupled or Uncoupled) pitch, roll command bars/collective cue are presented on each PFD, depending on the mode selected.
The coupled function is also automatically forced to Uncoupled whenever:

- the AP is not in ATT Mode
- moving the CLTV/YAW TRIM switch to OFF when a collective FD mode is engaged

In case of single AP operation, when a FD mode is engaged the coupled function remains available.

Autopilots ignore the FD commands when pilot flies the helicopter manually against any artificial feel device while the FD is coupled, because of the detent microswitch inside the Trim Actuator. FD remains coupled and no change in FD settings occurs.

FD AURAL TONES

MODE CHANGE TONE
A chime sounds whenever any of the following occurs:

- Autopilot mode changes from ATT to SAS or vice versa
- A FD mode changes status from disengaged to engaged or vice versa
- A FD mode changes status from armed to captured
- Coupling status (CPL) changes from Coupled to Uncoupled or vice versa

Arming of modes does not trigger the tone.
When a collective FD mode is engaged and coupled, the tone is triggered by moving the CLTV/YAW TRIM switch to OFF.

ALTITUDE REFERENCE CHANGE TONE
An aural tone ("bip-bip") sounds for any change in altitude or radar height mode reference datum.

ALTITUDE ALERTER
When ALT (Altitude Holding) or RHT (Radar Height Holding) modes are engaged, an aural warning message "ALTITUDE – ALTITUDE" is generated whenever aircraft exceeds a predetermined altitude/height deviation.

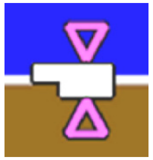
FLIGHT DIRECTOR INDICATORS

1. COLLECTIVE REFERENCE MARKERS

   (In view only if a FD Collective mode is engaged)

   The collective reference markers (fixed) are set as two hollow magenta triangles pointing towards each other: they represent the required position of the collective lever to fulfil the FD collective command

2. COLLECTIVE CUE

   (In view only if a FD Collective mode is engaged)

   The collective cue represents the current position of the collective lever relative to reference markers.

   If collective cue is shown below the reference markers, the collective lever is to be pulled up, and viceversa.

3. ENGAGED VERTICAL COLLECTIVE MODE Annunciator

   Shows what collective mode is engaged (captured)

   Green …………… Flight Director is operating normally

   Amber …………... Collective trim actuator is failed or collective pitch mode (ie ALT) is affected by the FD torque limiting function

4. ARMED VERTICAL MODE Annunciator (white)

   Shows what vertical mode(s) is(are) armed (collective and/or pitch)

5. ENGAGED VERTICAL PITCH MODE Annunciator

   Shows what pitch mode is engaged (captured)

   Green …………… Flight Director is operating normally

   Amber …………... Both linear actuators are failed or vertical pitch mode (ie IAS) is affected by the FD torque limiting function

6. ENGAGED LATERAL MODE Annunciator

Shows what roll mode is engaged (captured)

Green …………… Flight Director is operating normally

Amber …………... Both linear actuators are failed

7. ARMED LATERAL MODE Annunciator (white)

Shows what lateral mode is armed (roll)

Green …………… Flight Director is operating normally

Amber …………... Both linear actuators are failed

8. FLIGHT DIRECTOR SOURCE SELECT ARROW AND STATUS

The source select arrow points towards the PFD that is supplying navigation and mode select data to both flight directors.

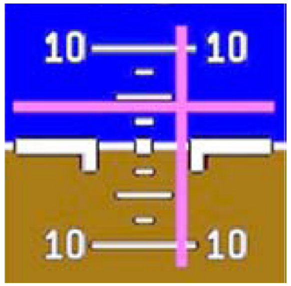
Green …………… Associated Flight Director is valid

Amber …………... Associated Flight Director is invalid

FD FAIL flag …… Both Flight Directors invalid (replaces the arrow annunciator)

9. FD COMMAND BARS

Show the amount of roll (Lateral command bar) or pitch (Vertical command bar) guidance command from the Flight Director.

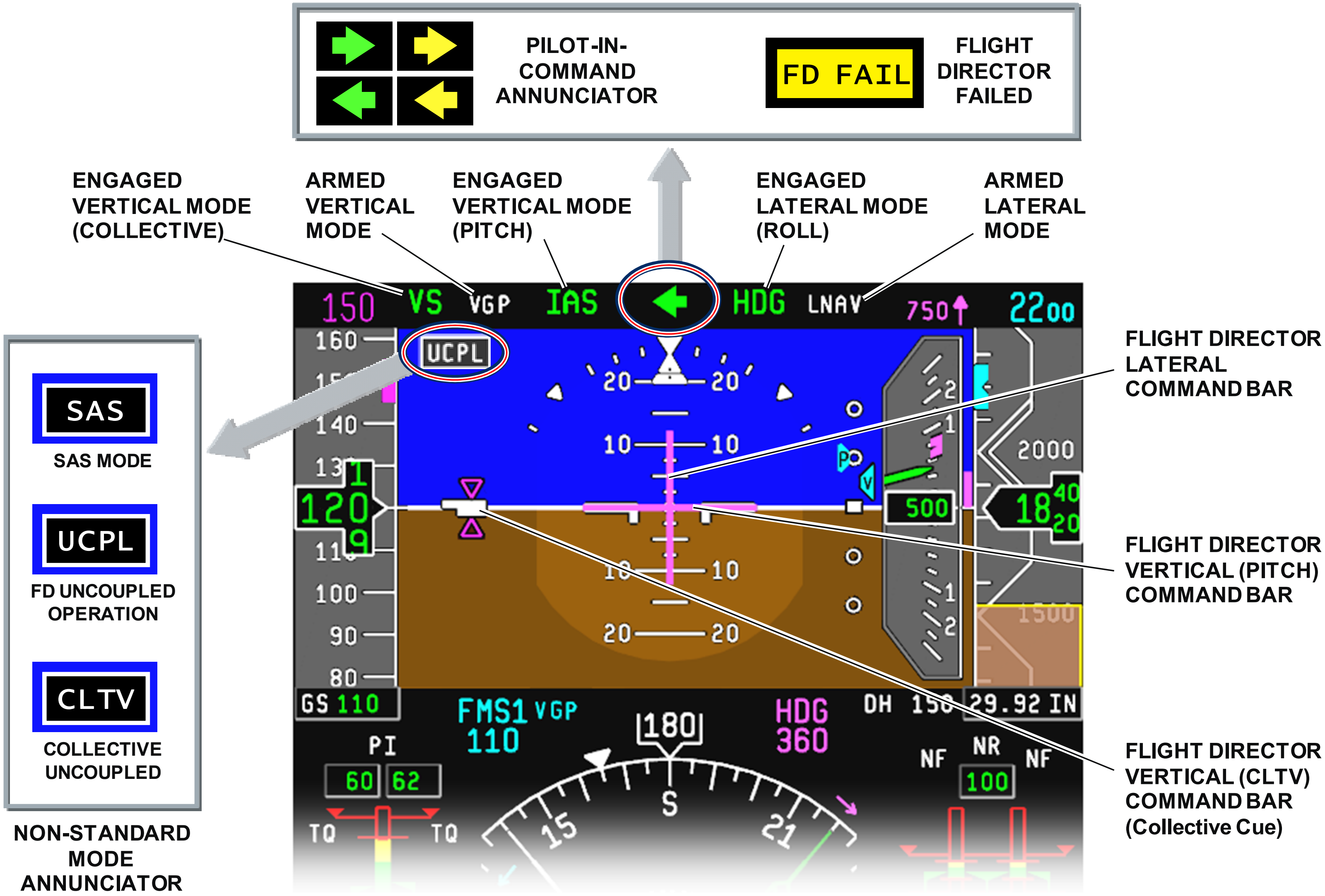


Either command bar is in view only if a relevant channel mode (roll or pitch) is engaged and is read with respect to the ADI aircraft symbol.

When a FD mode is engaged, and remains uncoupled from the AP, it permits the pilot to manually fly the aircraft using the directional cues presented by the command bars.

To follow the command bars, the pilot manually flies the aircraft to where the command bars intersect on the ADI.

In the example aside, the aircraft is below and to the left of the desired course and altitude. The pilot should execute a climbing, right-hand turn to place the aircraft on course and at the designated altitude.

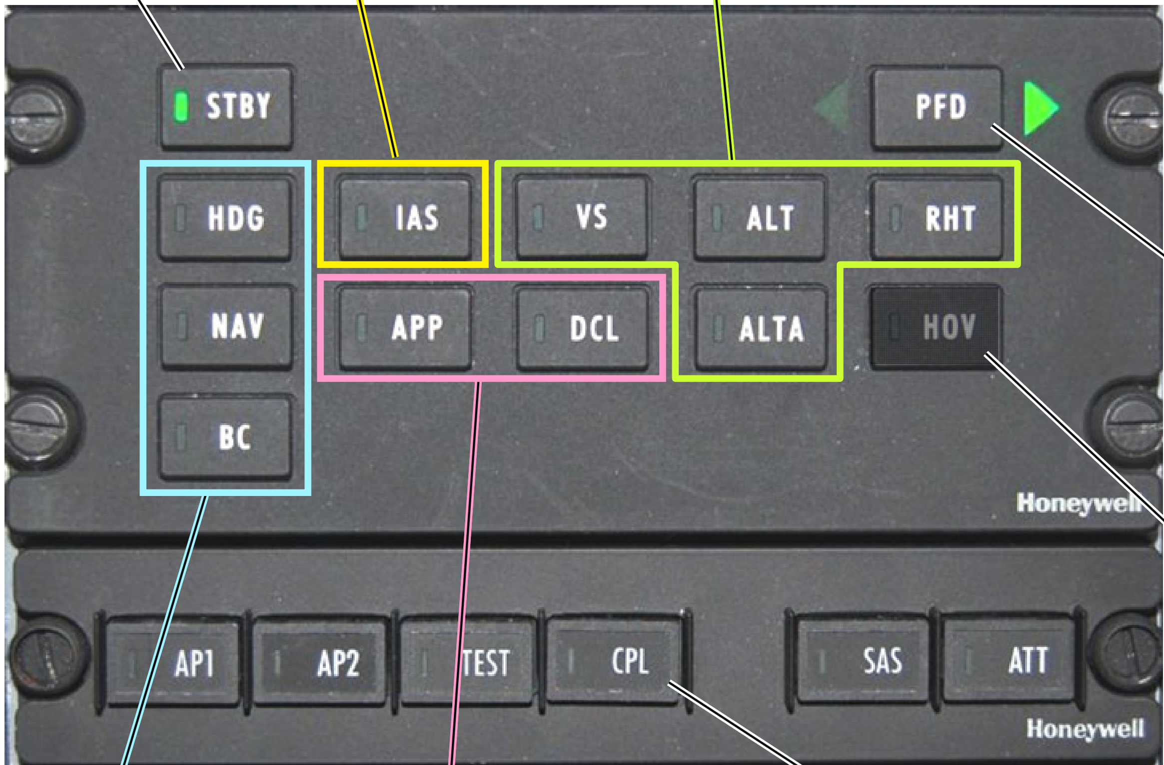


FD INDICATIONS

FLIGHT DIRECTOR STAND-BY

VERTICAL MODES (PITCH)

VERTICAL MODES (COLLECTIVE)



PFD SELECTOR (PILOT-IN-COMMAND)

HOVER MODE (4-Axis Enhanced only)

LATERAL MODES (ROLL)

APPROACH MODES

FLIGHT DIRECTOR COUPLED/DECOUPLED

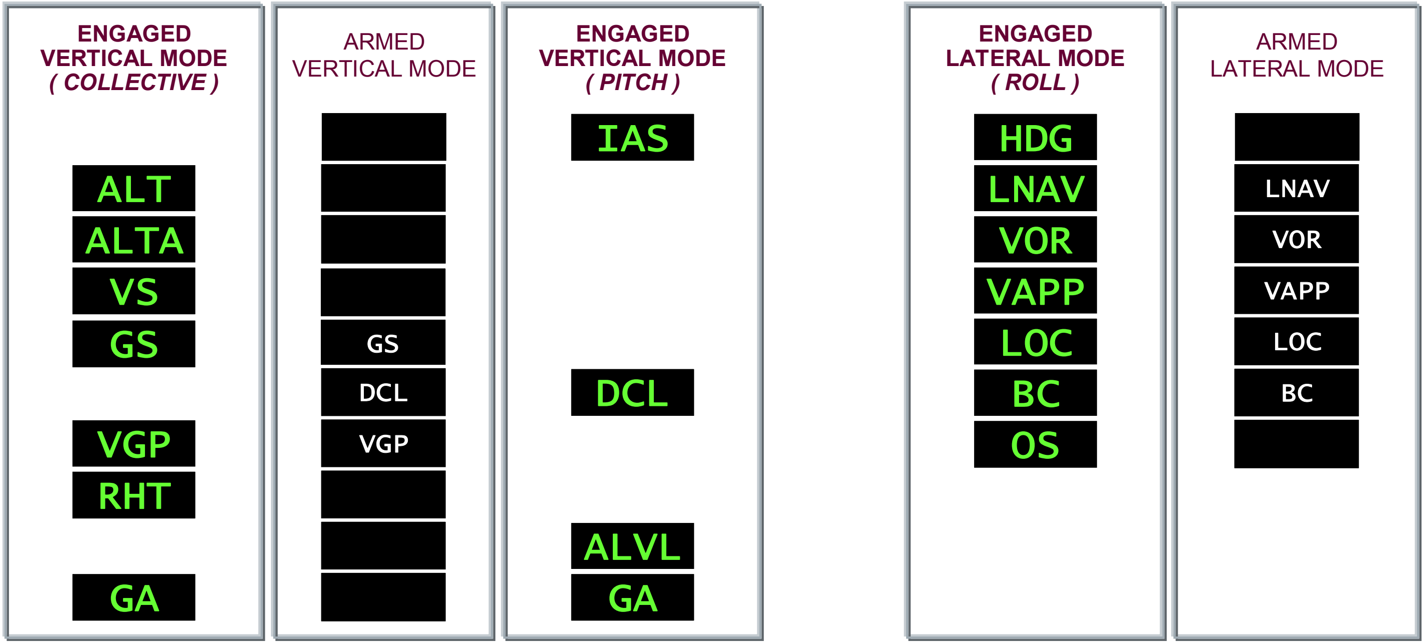
4- AXIS BASIC FLIGHT DIRECTOR MODE SELECTION

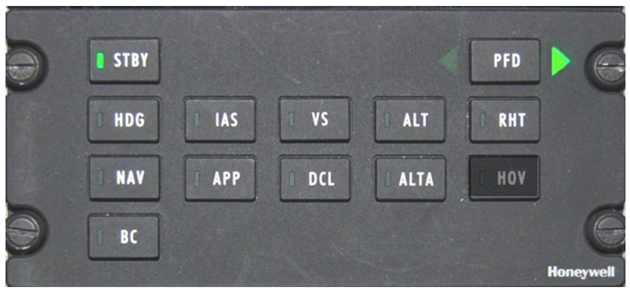
FLIGHT DIRECTOR MODES

Mode	Function	Control Axis	PFD caption ARM	PFD caption CAPTURE
HDG	Heading select	Roll and Yaw	N/A	HDG
ALT	Altitude Hold	Collective	N/A	ALT
IAS	Indicated Airspeed Hold	Pitch	N/A	IAS
NAV	Lateral navigation	Roll	LNAV VOR LOC	LNAV VOR LOC
APP	Lateral Approach	Roll for VOR or LOC	VAPP LOC	VAPP LOC
APP	Vertical Approach	Collective for Glide Slope or VGP	GS VGP	GS VGP
PRV	Preselect VOR-ILS approach while in LNAV or Preselect VGP	N/A	VAPP LOC GS VGP	N/A
DCL	ILS Deceleration VGP Deceleration	Pitch	DCL	DCL
BC	Back Course Approach	Roll	BC	BC
ALTA	Altitude Acquire	Collective	N/A	ALTA
VS	Vertical Speed Hold	Collective	N/A	VS
GA	Go-Around	Pitch for AS function/ Collective for VS function	N/A	GA
VGP	Vertical Glide Path	Collective	VGP	VGP
RHT	Radar Height Hold	Collective	N/A	RHT
HOV *	Hover / Velocity Hold	Pitch / Roll	N/A	HOV
OS **	Over Station mode	Roll	N/A	OS
ALVL **	Auto-Level mode	Pitch / Collective	N/A	ALVL
SBY	Clear all FD modes	N/A	N/A	N/A

* Only for 4-axis Enhanced FD

** Provided automatically





4-AXIS BASIC FD MODES

COLLECTIVE PI LIMITING FUNCTION

During collective coupled operation the collective movement is limited by the following Power Index (PI) values:

- Maximum 97% AEO (reduced to 95% at altitudes above 10000 ft Hp)
- Maximum 106% AEO for airspeed less than 60 KIAS (5 MIN message displayed beside collective cue)
- Maximum 140% OEI
- Minimum 5% AEO
- Minimum 10% OEI

When PI limiting function is active and is limiting maximum collective movement, an amber LIM caption is displayed beside the collective cue.

If PI limiting is active with IAS engaged and the required reference for VS, ALTA, RHT, GA, or ALT cannot be achieved then the airspeed will automatically reduce as necessary to achieve the collective mode reference.

If the collective mode reference cannot be maintained when the airspeed has reduced to a minimum of 80 KIAS, this airspeed will be maintained (IAS caption is displayed in green) and the collective mode reference will be reduced (the collective mode caption is displayed in amber).

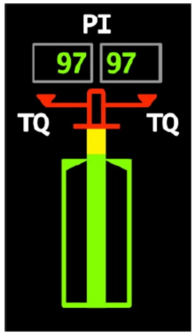
CAUTION

If PI limiting is active with ALT or RHT engaged and the reference barometric altitude or radio height cannot be maintained, the aural warning "ALTITUDE – ALTITUDE" will warn the pilot when the maximum allowed deviation from the reference setting has been exceeded.

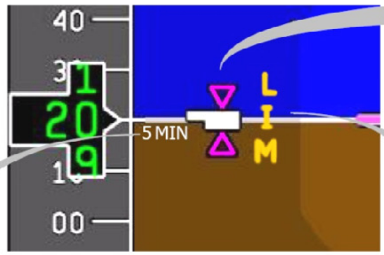
When flying at high altitude (above 15000 ft), selecting the LD SHARE switch on MISC panel to TORQUE improves the helicopter manoeuvering during automatic turns.

Collective PI Limiting active,
IAS reference value not attained

* Collective Cue



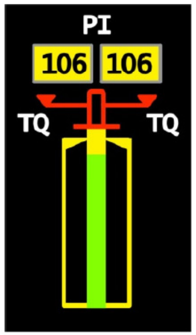
PI LIMITING
▪ **Max 97% AEO** (reduced to 95% at altitudes above 10000 ft Hp)
▪ **Max 106% AEO** for **IAS < 60 KIAS** (5 MIN message displayed beside collective cue)
▪ **Max 140% OEI**
▪ Min 5% AEO
▪ Min 10% OEI
Minimum IAS = 80 KIAS

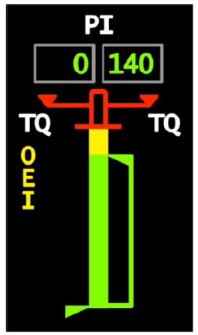


Collective
Cue *

Collective
PI Limiting
active

Collective PI Limiting at 106% and
within 5-minutes Take Off limit (IAS < 60 KIAS)





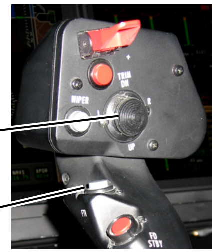
* *In UCPL operation when collective cue is below reference arrows collective must be raised and viceversa*

COLLECTIVE PI LIMITING

FD MODES

The following descriptions assume the modes are coupled, unless stated otherwise.

HDG — HEADING HOLD MODE

Description	The HDG mode provides the capability to steer the aircraft to capture and hold the Selected Heading reference displayed as the Heading Bug and associated digital readout on the HSI. Aircraft axis control is performed via roll attitude. The HDG select mode is supported with turn coordination functionality on the yaw axis.	
Engagement	Direct: Press HDG button on Guidance Controller Automatic: HDG mode automatically engages when any of the following mode is selected (and armed): VOR, VAPP, LOC, BC, LNAV When engaged Heading Bug and Selected Heading readout on both HSIs are shown in magenta.	HDG BUTTON 
References and performance	The Selected Heading reference is the Heading Bug on the HSI. Heading bug can be set and adjusted (even if HDG mode is not engaged) by either: • rotating the HEADING knob on either pilot or copilot Remote Instrument Controller • setting either pilot or copilot cyclic Beep TRIM switches to L or R (Beep rate: ±3°/s for first 3 seconds and then ±10°/s) Note: Only one Selected Heading is available on the helicopter; both pilot and copilot are always enabled to change it.	HEADING KNOB CYCLIC BEEP TRIM CYCLIC FTR

Heading bug can be synchronized with the actual heading (even if HDG mode is not engaged) by either:

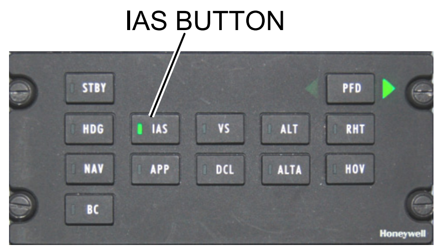
- pressing the PUSH SYNC button on HEADING knob on either pilot or copilot Remote Instrument Controller
- pressing the cyclic FTR button on either pilot's cyclic while HDG mode is engaged

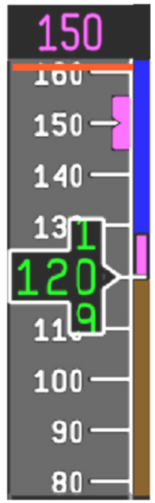
Heading bug is also synchronized with the actual heading when HDG mode is automatically engaged.

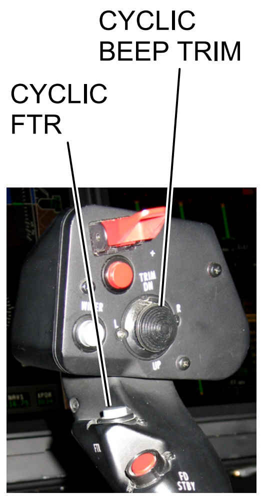
In HDG mode turns are performed at standard rate 1 (i.e. 3°/s).

— INDICATED AIRSPEED HOLD MODE

Description	The IAS hold mode generates pitch commands to maintain a selected aircraft airspeed.
Engagement	Direct: Press IAS button on Guidance Controller Automatic: IAS mode automatically engages when ALTA mode is selected or DCL mode is armed
Initial conditions	IAS mode maintains the airspeed existing at the time of engagement.
References and performance	When engaged the airspeed reference is displayed as a set bug and a digital readout in magenta on the Airspeed Indicator on PFDs. The airspeed reference bug is adjusted by either: • setting either pilot or copilot cyclic Beep TRIM switches to UP (to reduce IAS) or DN (to increase IAS) (Beep rate: ±3.5 kts/s) • repositioning the cyclic, with the FTR button depressed, to the required airspeed and releasing FTR button. The airspeed reference is adjustable via the cyclic beep switch between 60 KIAS and (Vne – 5) KIAS. If airspeed, at time of engagement, is higher than (Vne – 5) KIAS, then it is automatically reduced to (Vne – 5) KIAS. Note that the IAS reference will not increase when Vne increases.

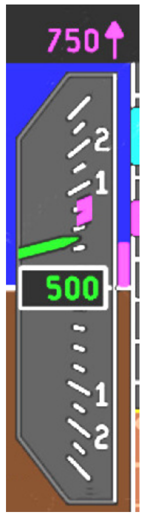






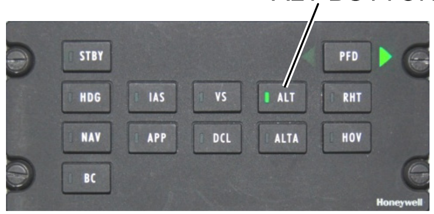
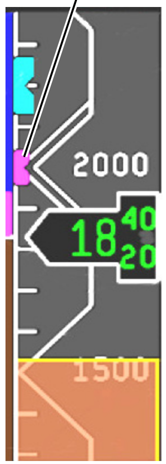
	The IAS reference is resynchronized when there is a transition from FD Uncoupled to FD Coupled.	

VS — VERTICAL SPEED HOLD MODE

Description	The VS hold mode generates collective commands to maintain a selected aircraft vertical speed.	
Engagement	Direct: Press VS button on Guidance Controller Automatic: VS mode automatically engages when ALTA mode is selected	VS BUTTON 
References and performance	VS mode maintains the vertical speed existing at the time of engagement. When engaged the vertical speed reference is displayed as a set bug and a digital readout in magenta on the Vertical Speed Indicator on PFDs. The vertical speed reference bug is adjusted by either: • setting either pilot or copilot collective Beep Trim CLTV/YAW switches to UP (to increase VS) or DN (to reduce VS) (Beep rate: ±150 fpm/s) • pressing the collective FTR switch while flying to the desired vertical speed, and then releasing the FTR. The VS mode reference can be set in the range of –1500 fpm to +2000 fpm. The VS reference is resynchronized when there is a transition from FD Uncoupled to FD Coupled.	CLTV/YAW TRIM    COLLECTIVE FTR


150
VS
VGP
IAS
HDG
LNAV
750
2200
160
150
140
120
100
90
80
20
10
2000
500
1840
20
1500
GS 110
FMS1 VGP
110
180
HDG
360
DH 150
29.92 IN
PI
60
62
TQ
TQ
S
15
21
NF
NR
NF
100

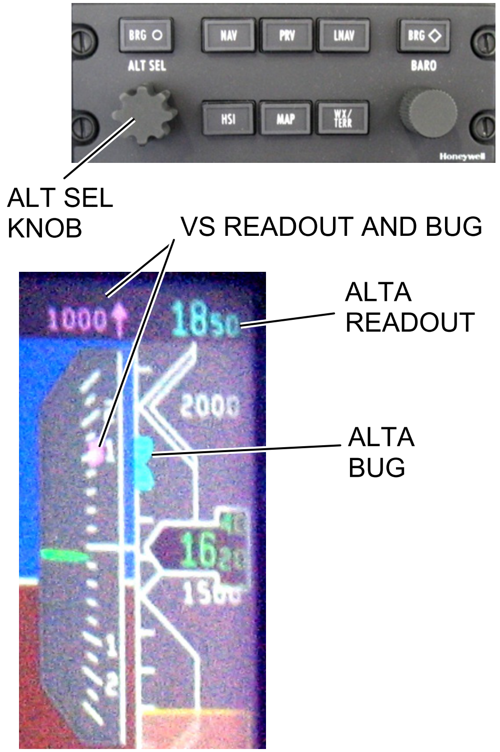
ALT — BARO ALTITUDE HOLD MODE

Description	The ALT hold mode generates collective commands to maintain a selected aircraft barometric altitude.	
Engagement	Direct: Press ALT button on Guidance Controller Automatic: ALT mode automatically engages when ALTA mode has acquired the pre-selected altitude	ALT BUTTON 
References and performance	ALT mode maintains the barometric altitude existing at the time of engagement. When engaged the baro altitude hold reference is displayed as a small magenta bug on the Barometric Altimeter on PFDs. The altitude hold reference bug is adjusted by either: • setting either pilot or copilot collective Beep Trim CLTV/YAW switches to UP (to increase altitude) or DN (to reduce altitude) (Beep rate: ±50 ft/s) • pressing the collective FTR switch while flying to the desired barometric altitude, and then releasing the FTR. Changing the ALT reference causes a Reference Change aural tone to be played when the reference begins to change. While the reference is being changed and for 5 seconds after finishing, no Reference Change aural tone is played.	ALT HOLD BUG CLTV/YAW TRIM  COLLECTIVE FTR

Adjustment of the barometric setting value displayed on the selected PFD will induce a corresponding change of the indicated altitude.
When changing the barometric setting value with the ALT mode engaged, the mode will command a climb or descent as necessary to return to the barometric altitude corresponding to the respective value indicated at the time ALT mode was engaged, last synchronized or beeped.

ALT mode can be engaged with HOV mode as an alternative to the RHT mode.

ALTA — ALTITUDE ACQUIRE MODE

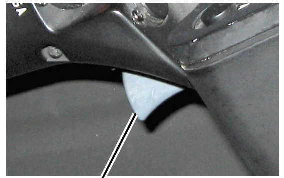
Description	The ALTA mode generates collective commands to climb or descend towards the pre-selected barometric altitude reference set with the Altitude Pre-Select knob (ALT SEL) on the Display Controller.	
Engagement	Direct: Press ALTA button on Guidance Controller	
References and performance	At power up, the digital readout for the ALTA reference will be displayed as dashes (invalid) until the ALT SEL knob is moved at least one click. When ALTA mode is engaged: • a vertical speed of either +1000 fpm or –750 fpm in the direction of the target altitude is automatically set • IAS mode is automatically engaged holding the current airspeed (60 KIAS minimum) The ALTA mode vertical speed reference is displayed as a set bug and as a digital readout on the PFD. The vertical speed reference for the ALTA mode can be changed by the pilot using one of the following means: • setting either pilot or copilot collective Beep Trim CLTV/YAW switches to UP (to increase VS) or DN (to reduce VS) (Beep rate: ±150 fpm/s) • pressing the collective FTR switch while flying to the desired vertical speed, and then releasing the FTR. The VS mode reference can be set in the range of –1500 fpm to +2000 fpm.	

The minimum vertical speed reference is 100 fpm in the direction of the target altitude

Changing the altitude preselect reference while ALTA is engaged, will cause the mode to attempt to capture the new reference once the altitude preselect knob is released.

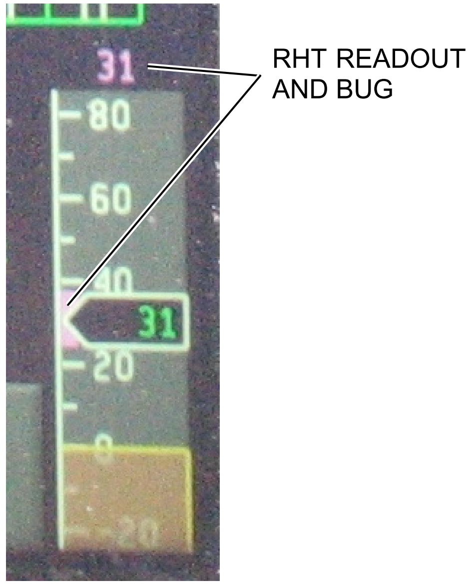
As the aircraft approaches and captures the pre-selected altitude, the ALTA mode automatically transitions to the ALT mode and IAS mode remains engaged.

CLTV/YAW TRIM



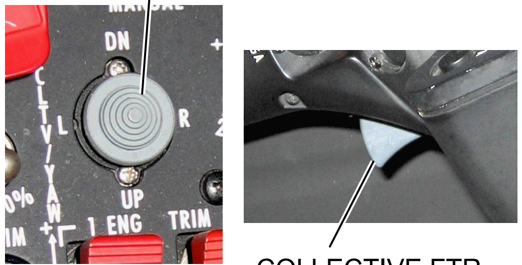
COLLECTIVE FTR

RHT — RADIO HEIGHT HOLD MODE

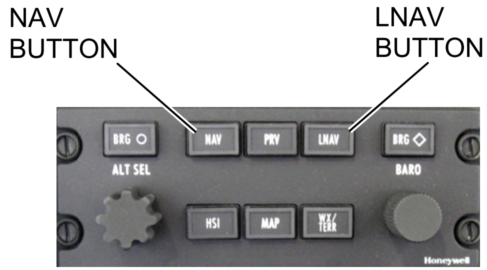
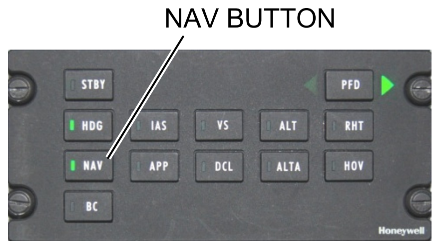
Description	The RHT hold mode generates collective commands to maintain a selected aircraft radio height.	
Engagement	Direct: Press RHT button on Guidance Controller Automatic: RHT mode automatically engages when • HOV mode has engaged • Collective is beeped in ALVL mode	
References and performance	RHT mode maintains the radio height existing at the time of engagement. When engaged the radio height hold reference is displayed set bug and a digital readout in magenta on the Radio Altimeter on PFDs.	

The radio height hold reference bug is adjusted by either:

- setting either pilot or copilot collective Beep Trim CLTV/YAW switches to UP (to increase height) or DN (to reduce height) (Beep rate: ±50 ft/s)
- pressing the collective FTR switch while flying to the desired radio height, and then releasing the FTR.



NAV MODE — VOR / LOC / LNAV FUNCTIONS

Description	The NAV mode generates roll commands to steer the aircraft to capture and hold the Selected VOR Course, ILS Localizer or FMS Desired Track dependent on the Navigation system selected as Primary Navaid. Primary Navaid is selected by the pilot in command via the Display Controller (DC) and displayed as the CDI and the associated digital readouts on the HSI. NAV button on DC: VOR or LOC LNAV button on DC: FMS	
Engagement	Arming: Press NAV button on Guidance Controller. ▪ NAV mode is armed (VOR, LOC or LNAV) ▪ HDG mode is automatically engaged and the present heading held Note: If LOC is selected as Primary Navaid, NAV mode selection does not provide Glide Slope guidance	
References and performance	The VOR / LOC Course can be set and adjusted by the COURSE knob on the Remote Instrument Controller (RIC). The FMS Desired Track is automatically computed by the FMS. VOR Course only can be synchronized with the current VOR bearing value by pressing the PUSH DIR button on COURSE knob on the RIC. Capture: Automatic capture of the Selected VOR Course, ILS Localizer or FMS Desired Track • occurs at lateral beam sensing point (navigation source must be valid), • disengages the HDG mode • activates the VOR, LOC or LNAV mode, respectively	

When VOR mode is captured, the VOR deviation (difference between VOR bearing and course selection) is gain-programmed as a function of distance from the station (DME and/or FMS). If distance is not available, the gain-programming uses the default values optimized for cruise.

The AFCS uses a course error signal to immediately correct the short-term heading disturbances such as wind gust. With a crosswind, a course error offset (crab angle) is computed and used by the FD to keep the aircraft on course.

APP MODE — VOR APPROACH FUNCTION

Description	The VAPP (VOR Approach) mode provides the capability to steer the aircraft to fly a VOR non-precision approach. VOR must be selected by the pilot in command as the Primary Navaid via the NAV button on the Display Controller (DC) and displayed as the CDI and the associated digital readout on the HSI. • Aircraft axes control is performed via roll attitude for VAPP.	NAV BUTTON 
Engagement	Arming: Press APP button on Guidance Controller ▪ VAPP mode is armed ▪ HDG mode is automatically engaged and the present heading held	APP BUTTON 
References and performance	The VOR Course can be set and adjusted by the COURSE knob on the Remote Instrument Controller (RIC). VOR Course can also be synchronized with the current VOR bearing value by pressing the PUSH DIR button on COURSE knob on the RIC. At VOR radial capture, HDG mode automatically disengages and VAPP mode automatically engages, so the helicopter turns to track the selected radial.	COURSE KNOB 

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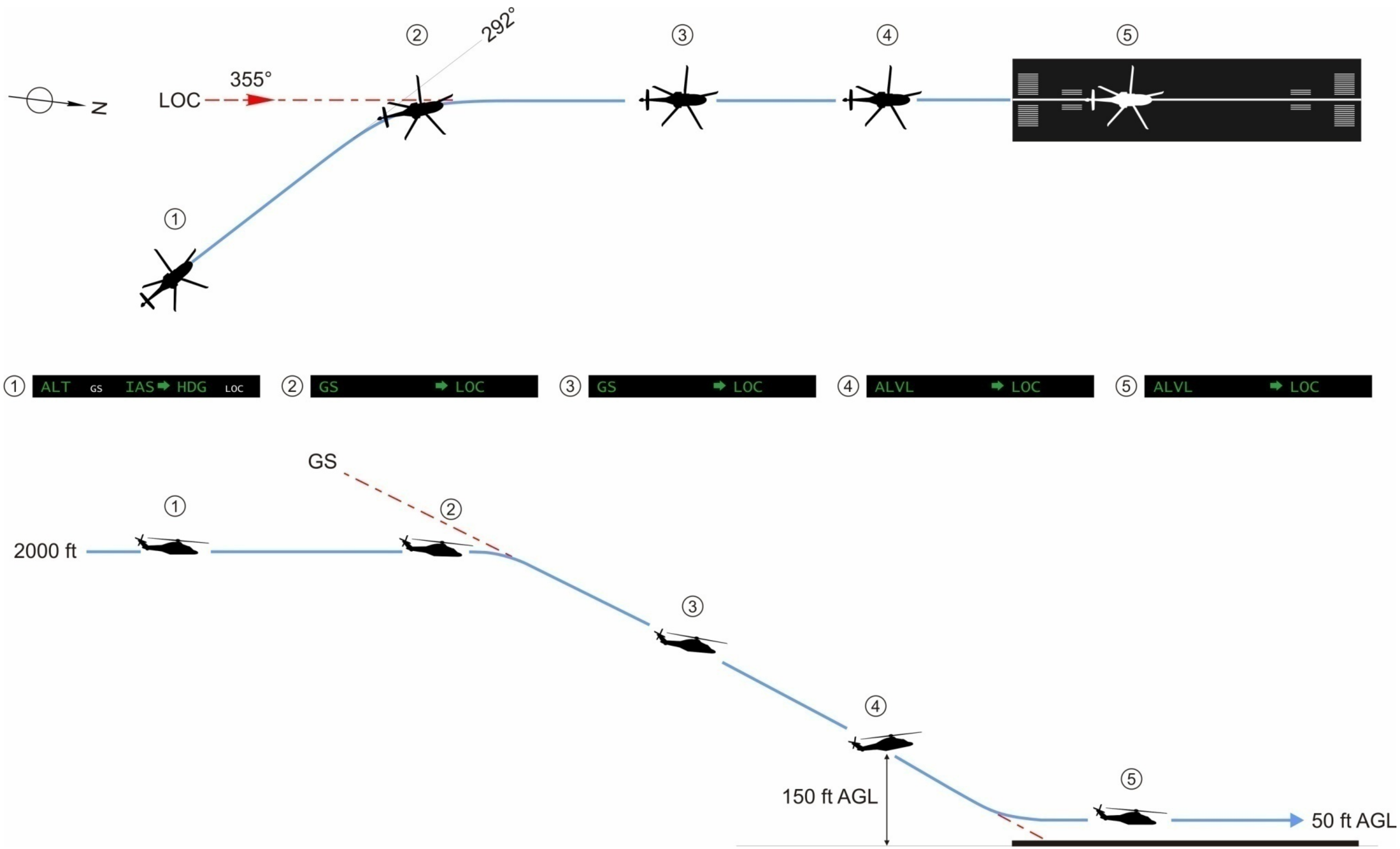
APP MODE — ILS APPROACH FUNCTION

Description	The APP mode provides the capability to steer the aircraft to capture and hold the ILS Localizer (LOC) and Glide Slope (GS). LOC must be selected by the pilot in command as the Primary Navaid via the NAV button on the Display Controller (DC) and displayed as the CDI and the associated digital readout on the HSI. Aircraft axes control is performed via: • roll attitude for LOC • collective input for GS	NAV BUTTON 
Engagement	Arming: Press APP button on Guidance Controller. ▪ GS and LOC modes are armed ▪ HDG mode is automatically engaged and the present heading held	APP BUTTON 

References and performance	The LOC inbound (front) course can only be set and adjusted by the COURSE knob on the Remote Instrument Controller (RIC). At LOC capture, HDG mode automatically disengages and LOC mode automatically engages, so the helicopter turns to track the selected localizer. At GS capture, any selected collective mode automatically disengages and GS mode automatically engages, so the helicopter adjusts collective to hold the glide slope. Upon completion of ILS Approach, ALVL mode is automatically engaged.

COURSE KNOB

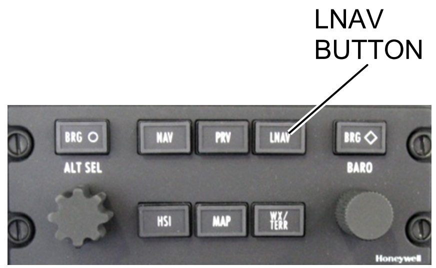

N
LOC
355°
292°
1
2
3
4
5
1 ALT GS IAS ➡ HDG LOC
2 GS ➡ LOC
3 GS ➡ LOC
4 ALVL ➡ LOC
5 ALVL ➡ LOC
GS
2000 ft
150 ft AGL
50 ft AGL

ILS PATTERN

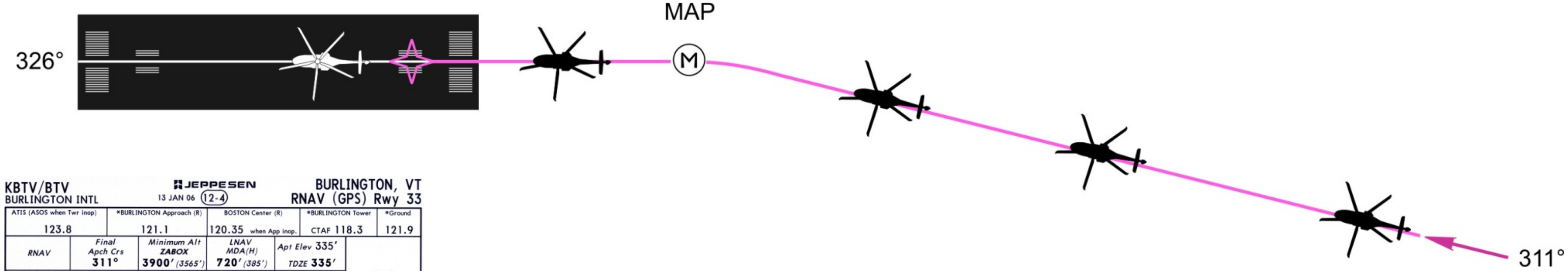

100
ALT
GS
IAS
HDG
LOC
TAWS FAIL
GS 103G
LOC1
355
HDG
292
DH 200
1009 HPA
PI
106
0
TQ
TQ OEI TNG
NF
NR
100
1180
--- NM

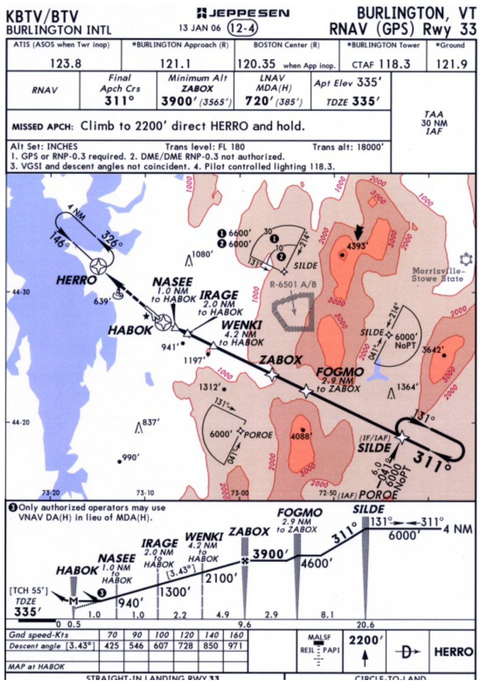

100
GS
IAS
LOC
TAWS FAIL
500
GS 100G
LOC1
355
HDG
292
DH 200
1009 HPA
PI
58
0
TQ
TQ OEI TNG
NF
NR
100
1100
--- NM

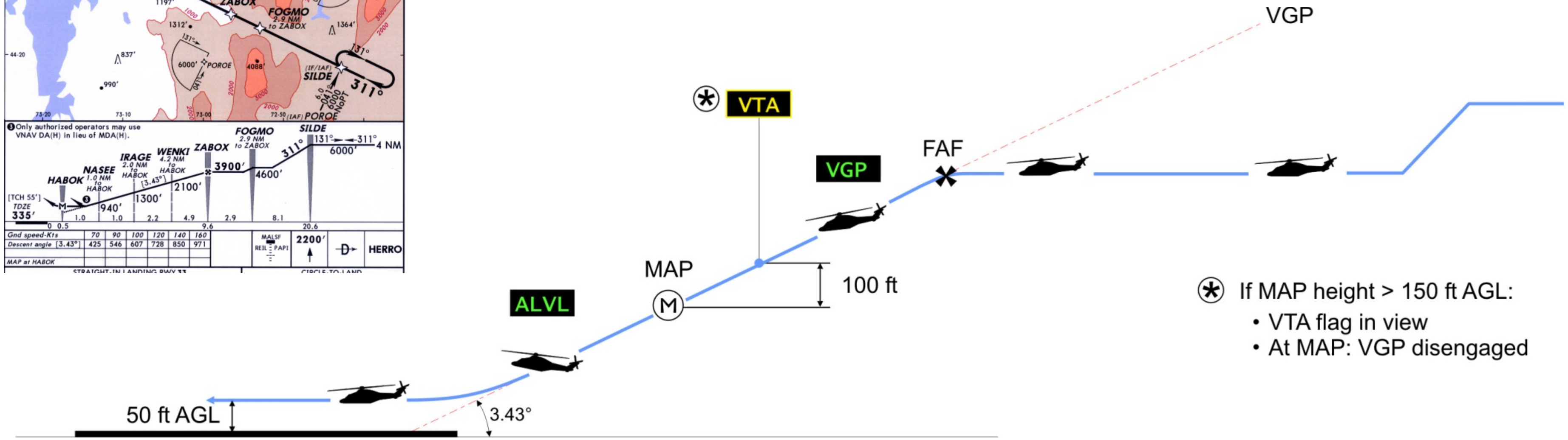
APP MODE — VGP APPROACH FUNCTION (GPS/RNAV APPROACH)

Description	The APP mode produces a non precision approach similar to an ILS approach when using a non localizer based approach from the data base with FMS selected as the Primary Navaid (LNAV button on DC). The Vertical Glide Path (VGP) is anchored at the BOD (Bottom of Descent) —ie the touchdown point— with an angle equal to that specified in the data base at the corresponding waypoint. Aircraft axes control is performed via: • roll attitude for LNAV lateral steering • collective input for VGP	
Engagement	Arming: Press APP button on Guidance Controller ▪ VGP mode is armed ▪ HDG mode is automatically engaged and the present heading held Conditions for arming VGP mode: • FMS is selected as Primary Navaid and a published non-Localizer approach is selected as Arrival in the Active FPL • FMS is not in DR (Dead Reckoning) mode • Helicopter is within 30 nm of destination • Altitude and angle constraint values have not been changed • If there is an altitude constraint at the FAF (Final Approach Fix), the helicopter must be at the FAF altitude • No vertical direct-to the FAF has been executed Note: If APP pushbutton is pressed when Primary Navaid is FMS and VGP is not available, the VGP UNAVAILABLE message is	

	displayed in the MCDU scratchpad.	
References and performance	On the glide slope pointer a letter P is displayed to indicate VGP. When VGP mode is armed, both letter P and pointer appear on the left hand side of vertical deviation scale. When captured the symbol moves to the right hand side of the vertical deviation scale. When flying to the MAP (Missed Approach Point) or within 5 nm from the FAF, any selected collective mode automatically disengages and the VGP mode is automatically captured, so the helicopter adjusts collective to hold the VGP. Conditions for capturing VGP mode: • VGP mode must be armed • FMS active waypoint is the MAP or helicopter is within 5 nm of the FAF • If holding, helicopter must be established on the inbound course to the FAF • Helicopter is able to capture the final approach slope As the helicopter approaches the runway threshold, ALVL mode is automatically engaged and collective is commanded for an asymptotic flare. If MAP height (in the Navigation Data Base) is greater than 150 ft AGL: • when helicopter is 100 ft above the MAP the VTA (Vertical Track Alert) flag appears on the ADI • when helicopter is at the MAP the VGP mode disengages	


326°
MAP
M
311°

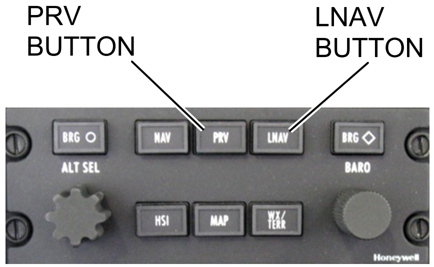
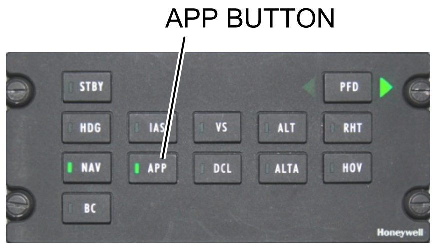

KBTV/BTV
BURLINGTON INTL
JEPPESEN
13 JAN 06 (12-4)
BURLINGTON, VT
RNAV (GPS) Rwy 33
123.8
121.1
120.35
CTAF 118.3
121.9
RNAV
Final Apch Crs 311°
Minimum Alt ZABOX 3900′ (3565′)
LNAV MDA(H) 720′ (385′)
Apt Elev 335′
TDZE 335′
MISSED APCH: Climb to 2200′ direct HERRO and hold.
TAA 30 NM IAF
HERRO
HABOK
NASEE
IRAGE
WENKI
ZABOX
FOGMO
SILDE
POROE
311°
326°
146°
6000′
3900′
4600′
2100′
1300′
940′
HERRO


VGP
VTA
VGP
FAF
MAP
M
100 ft
ALVL
50 ft AGL
3.43°
If MAP height > 150 ft AGL:
• VTA flag in view
• At MAP: VGP disengaged

VGP PATTERN

VGP CAPTURED

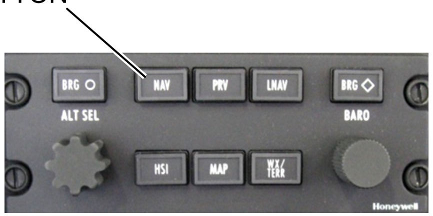
APP MODE — PREVIEW FUNCTION

Description	The Preview function is a way to preselect ILS or VOR Approach functions while FD is still engaged in LNAV. The Preview function is selected by pressing the PRV pushbutton on the Display Controller with FMS already selected as the Primary Navaid (LNAV button on DC).	
Engagement	Arming: Press APP button on Guidance Controller. ▪ GS and LOC modes are armed ▪ Helicopter keeps flying the LNAV leg	
References and performance	Selected Preview Course is displayed on the HSI as an additional thin cyan CDI and a small-font digital readout. Preview VOR/LOC Course can be set and adjusted by the COURSE knob on the Remote Instrument Controller (RIC). Preview VOR Course can also be synchronized with the current VOR bearing value by pressing the PUSH DIR button on COURSE knob on the RIC. If the APP pushbutton is pressed the FD arms LOC and GS or VAPP functions depending on the NAV receiver tuning (ILS or VOR). At the capture of the LOC or VAPP functions the FD disengages LNAV and engages LOC-GS or VAPP modes.	


120 ALT GS IAS LNAV LOC 1800
TAWS FAIL
GS 122G
FMS2 330
LOC2 355
HDG 331
DH 200 1009 HPA
PI
TQ
NF NR NF
100
GABC2
3.9 NM
02 ETE
RNP 1.00 NM
OAT 18°C
VOR1
FMS2
COM1
124.925


120 ALT GS IAS LOC 1800
TAWS FAIL
GS 125G
LOC2 355
HDG 331
DH 200 1009 HPA
PI
TQ
NF NR NF
100
--- NM
OAT 18°C
VOR1
FMS2
COM1
124.925

DCL — ILS DECELERATION MODE

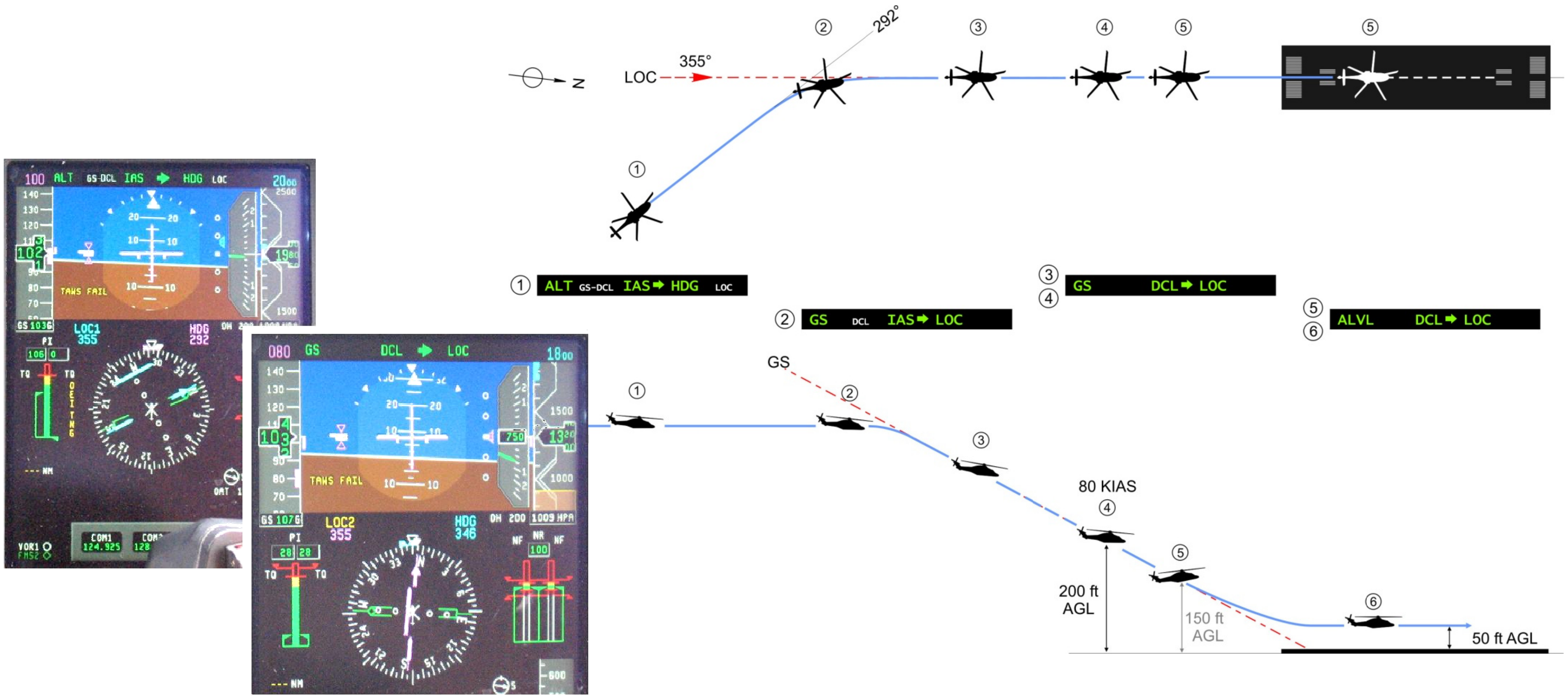
Description	The DCL-ILS mode is the same as the APP mode plus automatic deceleration from present speed down to approximately 80 KIAS as the aircraft reaches 200 ft AGL. LOC must be selected by the pilot in command as the Primary Navaid via the NAV button on the Display Controller (DC) and displayed as the CDI and the associated digital readout on the HSI. Aircraft axes control is performed via: • roll attitude for LOC • collective input for GS • pitch attitude for airspeed	NAV BUTTON 
Engagement	Arming: Press DCL button on Guidance Controller. ▪ GS-DCL and LOC modes are armed ▪ HDG mode is automatically engaged and the present heading held ▪ IAS mode is automatically engaged and the present airspeed held	DCL BUTTON 
References and performance	The LOC inbound (front) course can only be set and adjusted by the COURSE knob on the Remote Instrument Controller (RIC). At LOC capture, HDG mode automatically disengages and LOC mode automatically engages, so the helicopter turns to track the selected localizer. At GS capture, any selected collective mode automatically disengages and GS mode automatically engages, so the helicopter adjusts collective to hold the glide slope. After GS capture and at a computed condition for correct approach and	COURSE KNOB 

deceleration the IAS mode is disengaged and the DCL mode engaged.

The airspeed reference bug moves to 80 KIAS and the digital reference is set to 80 KIAS.

These values can be adjusted using the cyclic beep trim at which time the IAS mode automatically engages replacing the DCL mode.

Upon completion of ILS Approach, ALVL mode is automatically engaged.



BACK COURSE (BC) MODE

The BC mode allows the aircraft to capture and track the inbound back course of the localizer for approach to a runway by providing the lateral guidance in the roll axis.
After tuning, the course pointer on the PFD must be set to the runway inbound course, with the heading bug set to the necessary intersection heading. When the BC button is pressed on the GC, HDG mode is automatically engaged and BC armed. The BC mode operates in the same way as the LOC mode with the deviation and course signals that are opposite to make a BC approach to the localizer.

GA — GO AROUND MODE

The Go-Around mode (GA) provides aircraft control for an automatic ascent profile. The function is intended primarily for use during the approach phase of flight to initiate a missed approach.
GA mode is engaged by pressing he GA pushbutton on either collective; any other mode is then disengaged.
Upon engagement the GA mode commands collective for a 1,000 ft/min climb, roll for wings-level and pitch for an airspeed of 80 KIAS or the current airspeed, whichever is higher.

OS — OVER STATION MODE

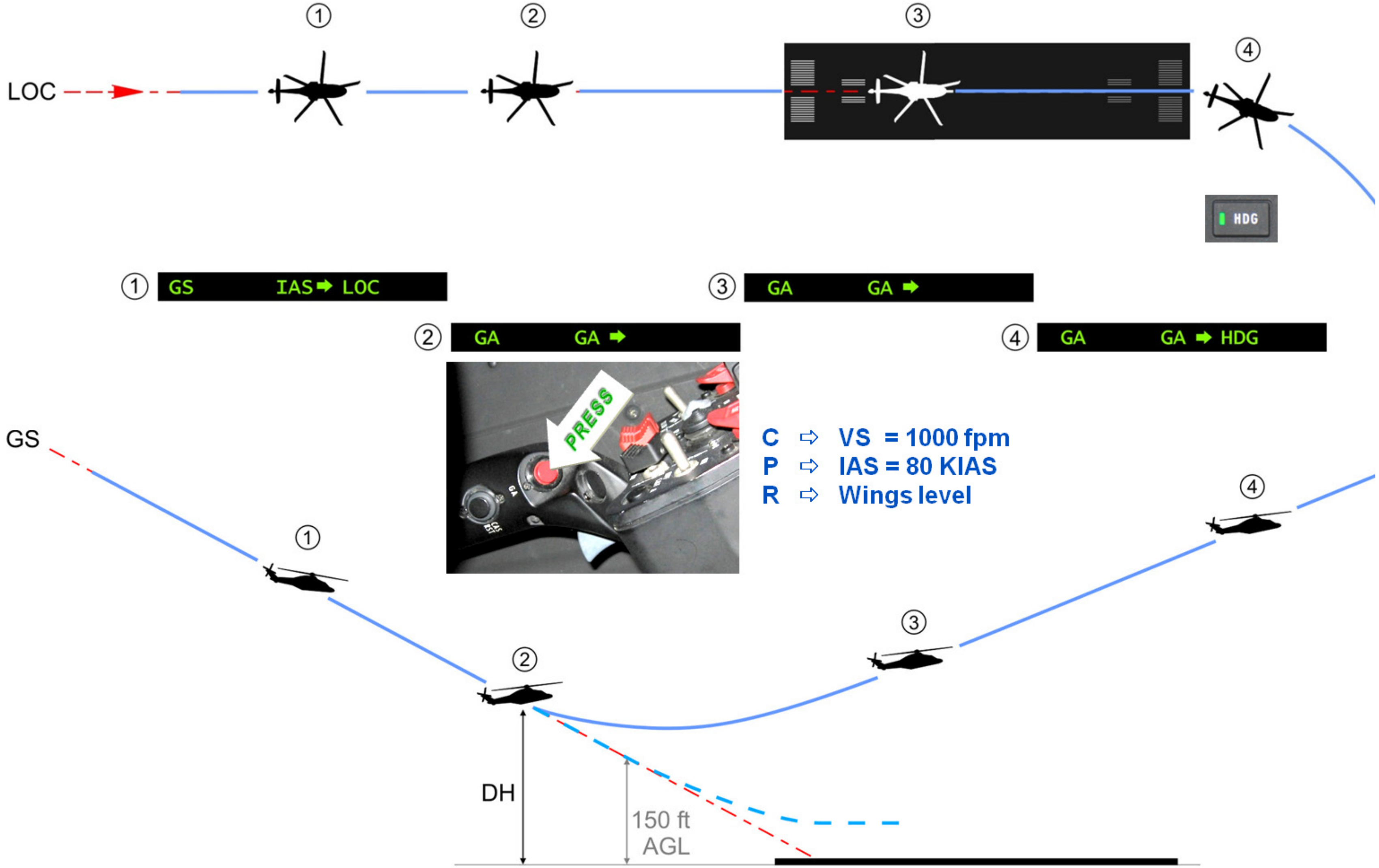
When overflying a VOR station, with VOR or VAPP mode engaged, the VOR equipment detects the aircraft being within the zone of ambiguity.
The VOR deviation signal is removed from the command until the VOR signal has stabilized after passing over station, when the OS mode disengages returning to VOR or VAPP mode automatically.
While over the station, course change may be made by selecting a new course via the COURSE knob on the RIC: aircraft is steered heading towards the selected course until OS mode is replaced by VOR or VAPP.

ALVL — AUTOLEVEL MODE

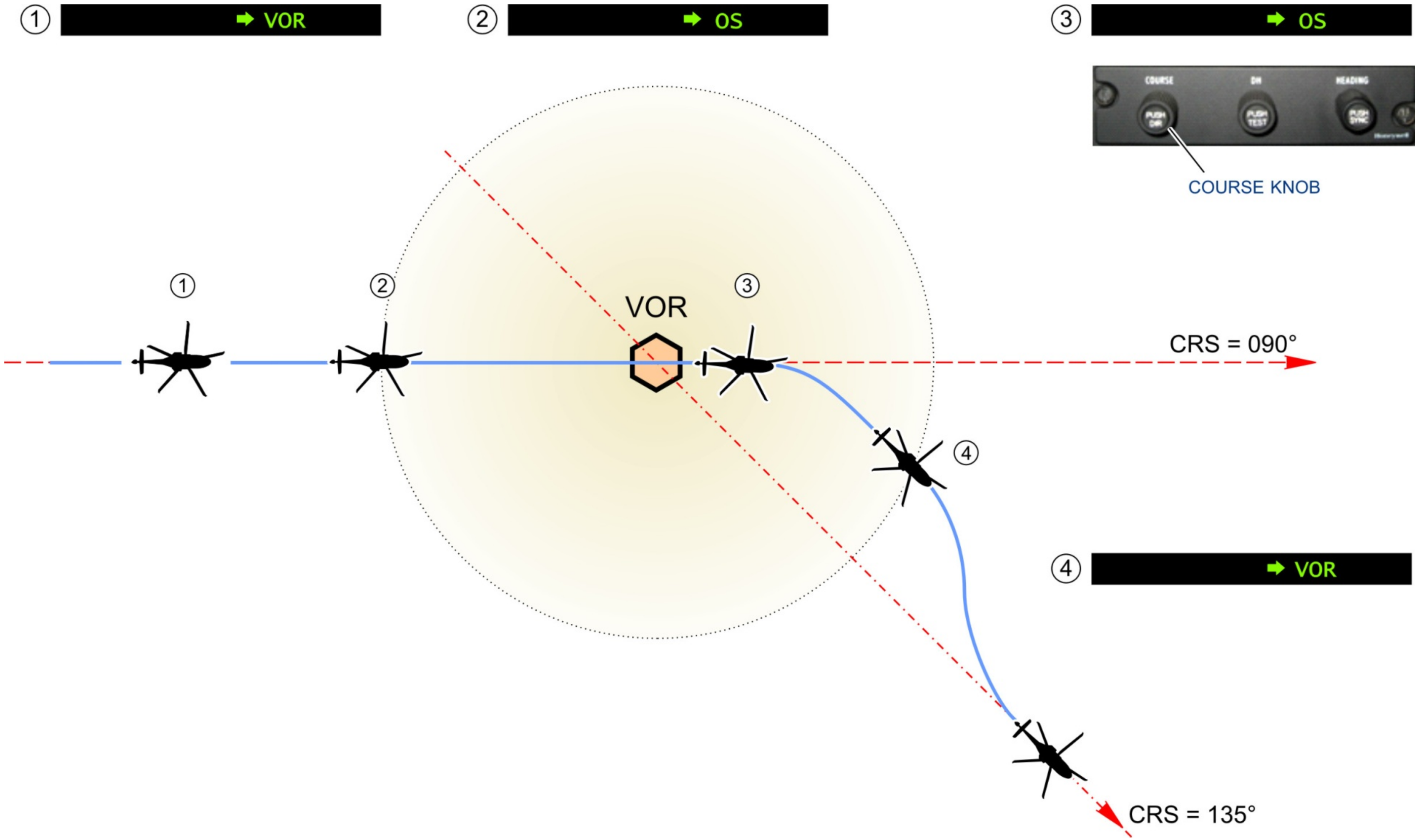
If the APP (LOC/GS or VGP) mode is continued, to below landing minimum, this safety feature is automatically activated at 150 ft AGL and an asymptotic flare to a radar height of 50 ft AGL will be initiated.
The green GS or VGP caption on the PFD will be replaced by ALVL.

CAUTION

In VGP mode the ALVL will not activate if the missed approach point (MAP) is higher than 150 ft AGL. For this case the VGP mode will disengage at the MAP (preceded by a Vertical Track Alert (VTA) caption displayed above the vertical guidance scale on PFD) and a chime is generated.



GO-AROUND MODE



OVER STATION MODE

HOV — HOVER/VELOCITY HOLD MODE

The HOV mode utilizes the blended AHRS-GPS ground velocity information to provide commands that maintain longitudinal and lateral aircraft ground velocities for hovering and low speed flying.
The HOV can be manually engaged by pressing the HOV push button on the Guidance Controller or by pressing the beep trim (Chinese-Hat fifth position) on the cyclic hand-grip.
At the engagement the HOV mode velocity references are set to zero. The engagement of HOV mode will automatically engage RHT mode if a valid radar altimeter signal is available and within the threshold limits.
The HOV mode velocity references may be changed by the pilot using any of the following means:

- pressing the cyclic FTR switch, flying to the desired lateral and longitudinal velocities and then releasing the cyclic FTR switch
- pressing the cyclic beep switch forward, aft, right, or left to increase the velocity reference in the direction of the switch press
- pressing the cyclic beep fifth position switch to promptly zero the velocity reference

The HOV mode velocity references are limited to a maximum of 60 knots forward and 40 knots left, right, and aft (while aft the velocity reference is limited to 40 knots total vector amplitude).

CAUTION

In ALT mode the voice message "Altitude Altitude" is triggered when altitude exceeds the reference altitude by ± 150 ft. Therefore, if ALT mode is engaged as an alternative to RHT at a height below 300 ft, set DH at a value 10 ft below the reference height in order to have an additional height deviation exceedances cue.

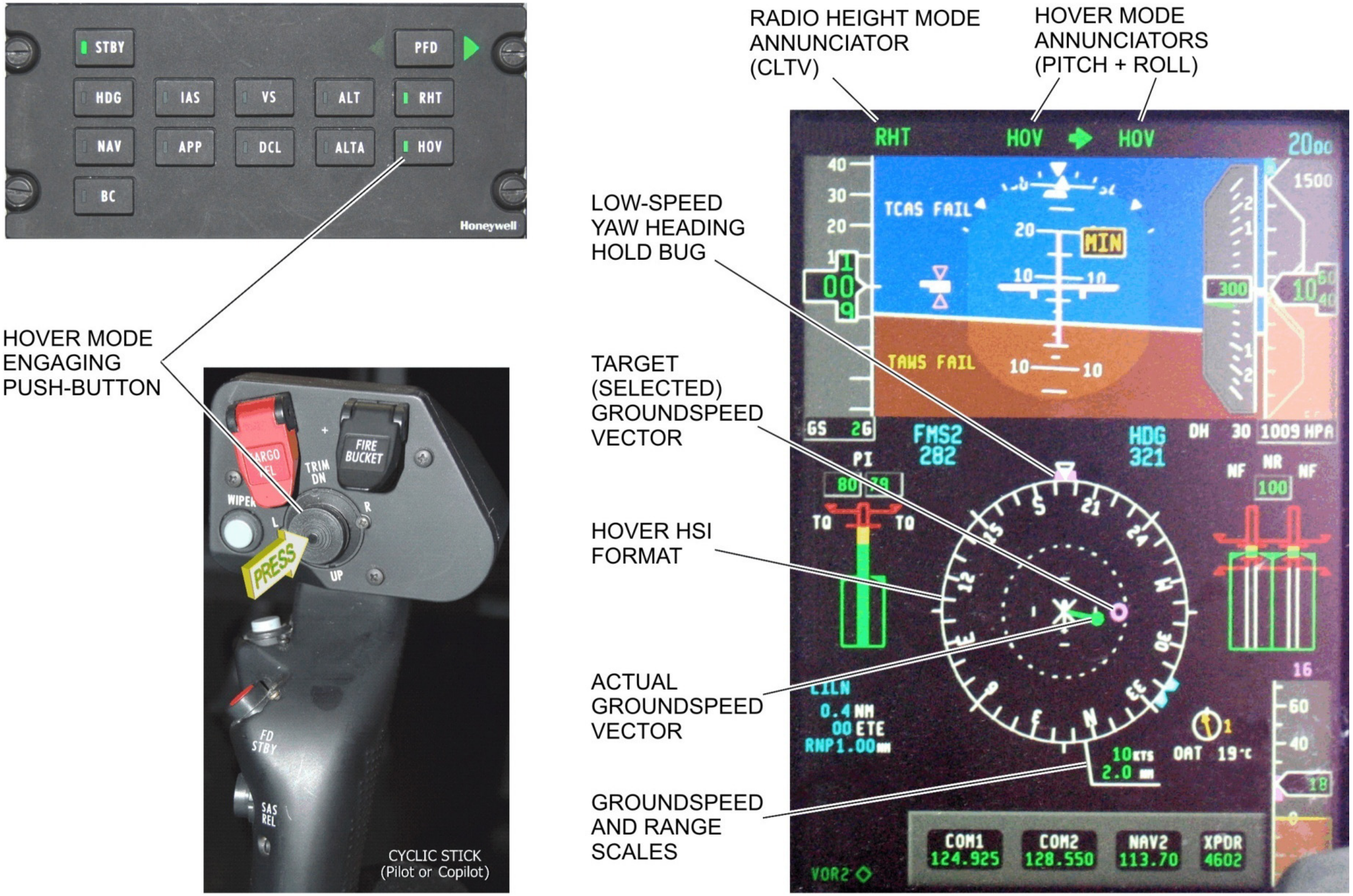
CAUTION

The HOV mode maintains a groundspeed reference therefore pilot must ensure that crosswind and rear wind speed limits are not exceeded. If wind limits are exceeded directional control may not be maintained.

CAUTION

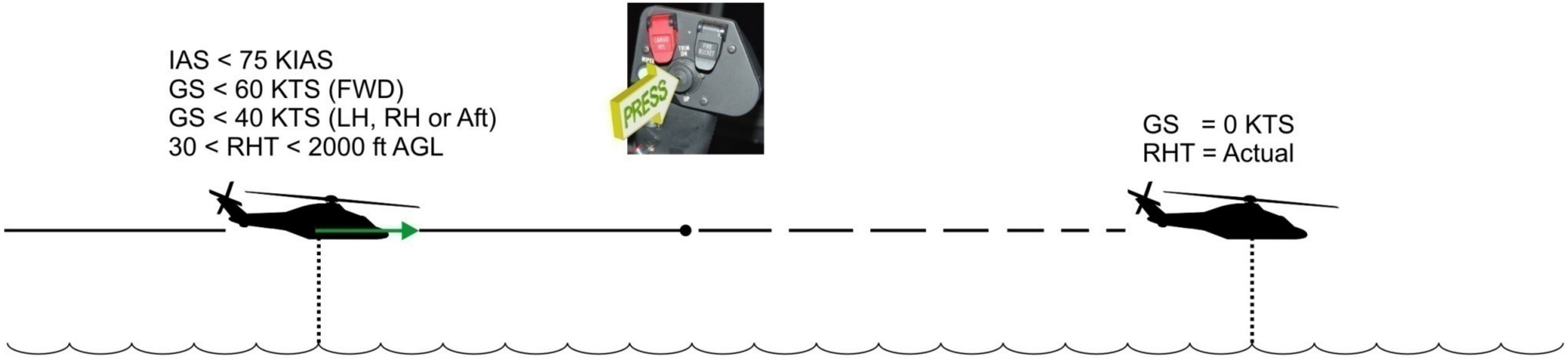
When HOV mode is engaged above 2000 ft AGL the ALT mode does not automatically engage. Therefore the pilot must control collective manually or engage ALT mode.

The hover symbology automatically appears on the PFD when HOV mode is engaged. The display shows the aircraft velocity vector and the reference velocity symbol.


STBY
PFD
HDG
IAS
VS
ALT
RHT
NAV
APP
DCL
ALTA
HOV
BC
Honeywell
HOVER MODE
ENGAGING
PUSH-BUTTON
FIRE
BUCKET
TRIM
DN
WIPER
PRESS
UP
FD
STBY
SAS
REL
CYCLIC STICK
(Pilot or Copilot)
RADIO HEIGHT MODE
ANNUNCIATOR
(CLTV)
HOVER MODE
ANNUNCIATORS
(PITCH + ROLL)
LOW-SPEED
YAW HEADING
HOLD BUG
TARGET
(SELECTED)
GROUNDSPEED
VECTOR
HOVER HSI
FORMAT
ACTUAL
GROUNDSPEED
VECTOR
GROUNDSPEED
AND RANGE
SCALES
RHT
HOV
HOV
TCAS FAIL
TAWS FAIL
MIN
FMS2
282
HDG
321
DH
1009 HPA
OAT 19°C
COM1
124.925
COM2
128.550
NAV2
113.70
XPDR
4602

HOVER MODE

IAS < 75 KIAS
GS < 60 KTS (FWD)
GS < 40 KTS (LH, RH or Aft)
30 < RHT < 2000 ft AGL



GS = 0 KTS
RHT = Actual

HOVER MODE OPERATION

FLIGHT DIRECTOR AURAL MESSAGES

VOICE ONLY MESSAGES	DESCRIPTION	
FD CHIME	Change state of any FD mode: • engaged to disengaged or viceversa • arm to captured • coupled to uncoupled or viceversa	
BIP-BIP	Change in altitude or radar height reference datum (target altitude or height)	
ALTITUDE - ALTITUDE	Aircraft exceeded the altitude deviation, dependent on the selected mode, by: ALT ± 150 ft from reference RHT Reference Height (ft) / Deviation (ft) 30 / 11 50 / 15 100 / 20 150 / 25 200 / 30 250 / 40 500 / 70 1000 / 135 1500 / 200 2000 / 260	

FLIGHT DIRECTOR FAILURES – PFD INDICATIONS

PFD INDICATION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
	replaces green arrow above attitude indicator on PFD that has failed Flight Director	SINGLE FLIGHT DIRECTOR FAILURE	Supplement 34 Supplement 40 — Section 3 EMERGENCY AND MALFUNCTION PROCEDURES
FD FAIL	replaces FD arrow above attitude indicator on both PFD displays when both FD have failed	DUAL FLIGHT DIRECTOR FAIL	
VTA	on attitude indicator during VGP approach when RAD ALT height between Missed Approach Point (MAP) and MAP+100 ft	VERTICAL TRACK ALERT CAPTION	
LOSS OF PFD WITH FD ENGAGED 	loss of PFD: the associated MFD will automatically configure to Composite mode maintaining the FD engaged on the same references as the PFD	FAILURE OF SELECTED PFD UNIT WITH FLIGHT DIRECTOR ENGAGED	
RAD	replaces RAD ALT height information on both PFD when both RAD ALT 1 and 2 have failed RHT mode disengages with audio chime	DOUBLE RAD ALT FAILURE WITH RHT ENGAGED	

PFD INDICATION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
RAD	on RAD ALT display when miscompare between RAD ALT 1 and 2 information RHT mode disengages with audio chime	RAD ALT MISCOMPARE WITH RHT ENGAGED	
VEL FAIL	below and to the left of the compass rose when the valid velocity data from both AHRS is not available. The velocity vector on the hover display is removed and the HOV mode disengages automatically.	VELOCITY VECTOR FAIL	Supplement 34 — Section 3 EMERGENCY AND MALFUNCTION PROCEDURES
VEL TEST	below and to the left of the compass rose when velocity is reported in test from an AHRS source.	None	

GUIDANCE CONTROLLER FAILURE

In case of guidance controller failure, recognised as non functionality of panel pushbuttons (that is modes cannot be changed or disengaged using the panel pushbuttons), the FD may be disengaged using the cyclic FD STBY button.

FLIGHT DIRECTOR FAILURES – CAS MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
1(2) COLL FAIL + aural message	Collective axis of AP1(2) fail NOTE: Collective FD mode annunciator amber (if a coupled collective mode is active)	SINGLE COLLECTIVE AUTOPILOT FAILURE	Supplement 34 Supplement 40 — Section 3 EMERGENCY AND MALFUNCTION PROCEDURES
1-2 COLL FAIL + aural message	Collective axis of AP1(2) failure on both AP 1 and AP 2 NOTE: Collective FD mode annunciator amber (if a coupled collective mode is active)	DUAL COLLECTIVE AUTOPILOT FAILURE	
1(2) AHRS FAIL **AVIONIC FAULT** **AFCS DEGRADED** **1(2) AP FAIL** + **ATT FAIL** **HDG FAIL** on PFD + aural message	associated AHRS failure and subsequent loss of AP 1(2) and loss of attitude, heading and slip skid data on Left (Right) PFD	AHRS FAILURE WITH FD ENGAGED	

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
ATT OFF	+ SAS message displayed on both PFD's Note With ATT system not engaged the aircraft flies in SAS mode only	ATTITUDE SYSTEM OFF	Supplement 34 Supplement 40 — Section 3 EMERGENCY AND MALFUNCTION PROCEDURES
1(2) ADS FAIL	On-side ADS failure on the selected PFD. Disengagement of FD vertical modes (ALT, IAS, VS) + Audio Chime and loss of Airspeed Altitude Vertical speed Data on Left(Right) indicators	ON SIDE ADS FAILURE ON SELECTED PFD WITH FD ENGAGED	

LIMITATIONS

Refer to AW139-RFM-4D, Supplement 34 or Supplement 40, SECTION 1 for:

- WEIGHT AND CENTRE OF GRAVITY LIMITATIONS
- FLIGHT DIRECTOR MODES LIMITATIONS
- FLIGHT DIRECTOR MODES ENGAGEMENT LIMITS AND MINIMUM USE HEIGHT (MUH)
- REQUIRED EQUIPMENT
- VOR LIMITATIONS
- ILS APPROACH MODE LIMITATIONS
- VGP LIMITATIONS

CHAPTER 23

COMMUNICATIONS

SECTION 00 – GENERAL

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COMMUNICATIONS – GENERAL

The communication system contains the equipment for communicating inside the helicopter, with other aircraft and with ground stations.

The communication system consists of the following main sub-systems:

- the Modular Radio System
- the Audio Integrating System

MODULAR RADIO SYSTEM – GENERAL

The Modular Radio System includes the Modular Radio Cabinet (MRC), which is the core of the communication and navigation system, with related controls, displays, and antennas. The MRC uses modules from the PRIMUS EPIC® system that supply Communication and Navigational data to the aircrew. The MRC contains the standard navigation and communication systems, including VOR, ILS, DME, ADF and VHF communication and the transponder system.

MODULAR RADIO SYSTEM – MAIN COMPONENTS

The MRC accommodates radio modules from the PRIMUS EPIC® system into one Line Replaceable Unit (LRU). The MRC is usually installed in a dual system configuration with MRC 1 on the left side (copilot) and MRC 2 on the right side (pilot). The two MRC doesn't however contain all the same radio modules. Each radio module consists of circuit cards, backplane connectors and front mounting plate with connectors. In addition, each radio module contains its own power supply and self-test circuitry that can be activated as an initiated built-in test (IBIT).

Usually the MRC 1 contains the following modules:

- Network Interface Module (NIM)
- VHF-NAV module (VIDL module = VOR/ILS/Data Link module) (VOR = Very high frequency Omni-directional radio Range; ILS = Instrument Landing System)

Usually the MRC 2 contains the following modules:

- Network Interface Module (NIM)
- VHF-NAV module (VIDL module = VOR/ILS/Data Link module)
- DME (Distance Measuring Equipment) module
- ADF (Automatic Direction Finder) module
- VHF-COMM module (VDL module = VHF Data Link module)
- XPDR (Transponder) module

MODULAR RADIO CABINET (MRC)

The MRC consists of a cabinet that houses multiple line replaceable modules providing aircraft radio functions.

NI-900 NIM MODULE

The NIM module consists of two sections:

- the NIC (Network Interface Controller) section that comprises the standard NIC circuit as used on all other ASCB-D connected units
- the central cabinet processing section (NIM CPU) which uses an Intel 80486 processor, DEOS (synchronized to ASCB-D) and PRIMUS EPIC core software components

NV-75A VIDL MODULE

The VIDL gives VOR/ILS and data link functionality (future implementation). It contains a VOR/LOC receiver, GS (Glide Slope) receiver and a marker beacon (MKR BCN) receiver.

DF-855 ADF MODULE

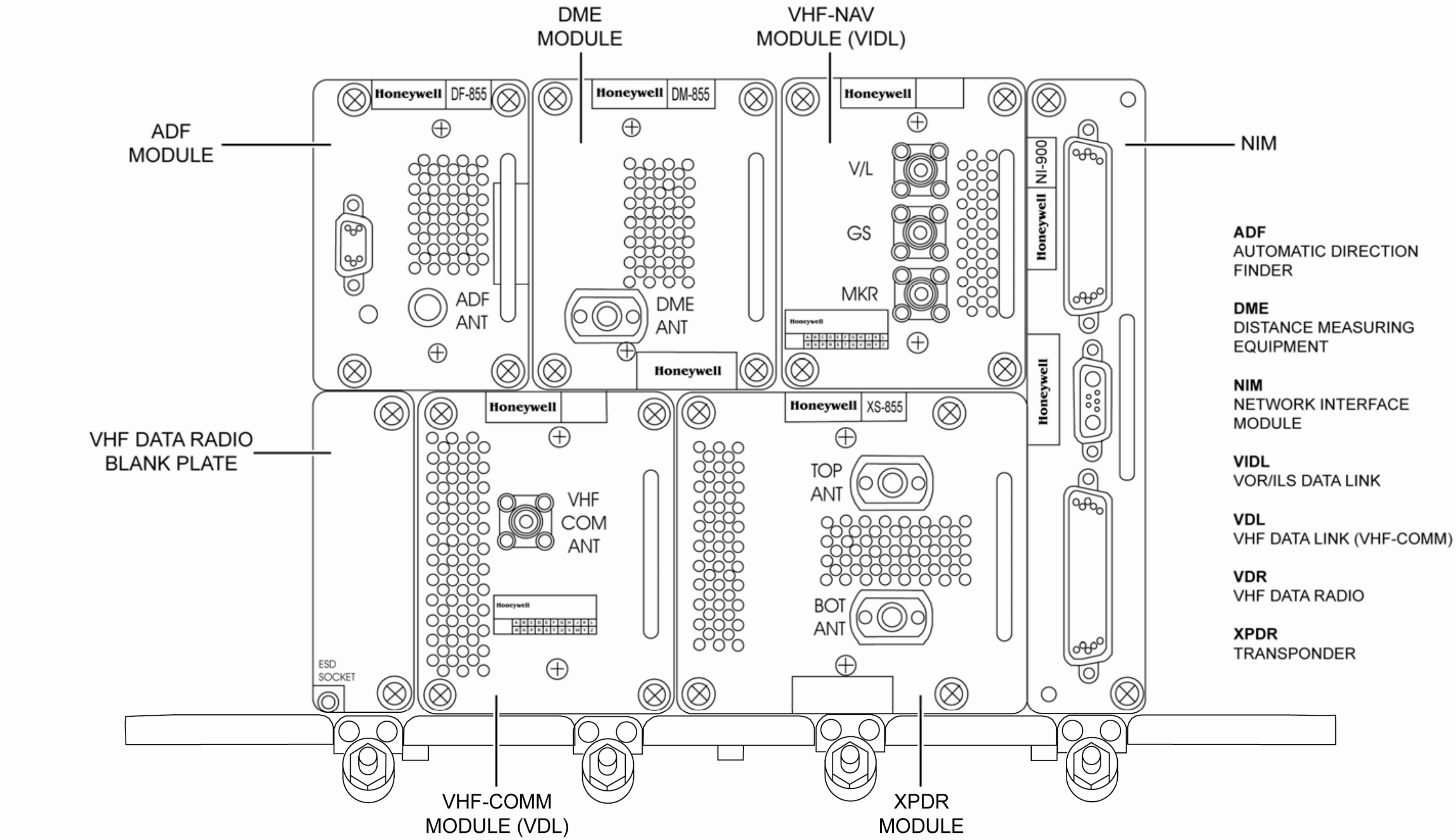
The ADF contains an ADF receiver that enables en route and terminal navigational and area guidance.

DM--855 DME MODULE

The DME contains a DME receiver that enables en route and terminal navigational and area guidance.

XS--856A XPDR Module

The XPDR contains a transceiver that gives air traffic control radar beacon system (ATCRBS), Mode S and diversity transponder capability



ADF
AUTOMATIC DIRECTION FINDER

DME
DISTANCE MEASURING EQUIPMENT

NIM
NETWORK INTERFACE MODULE

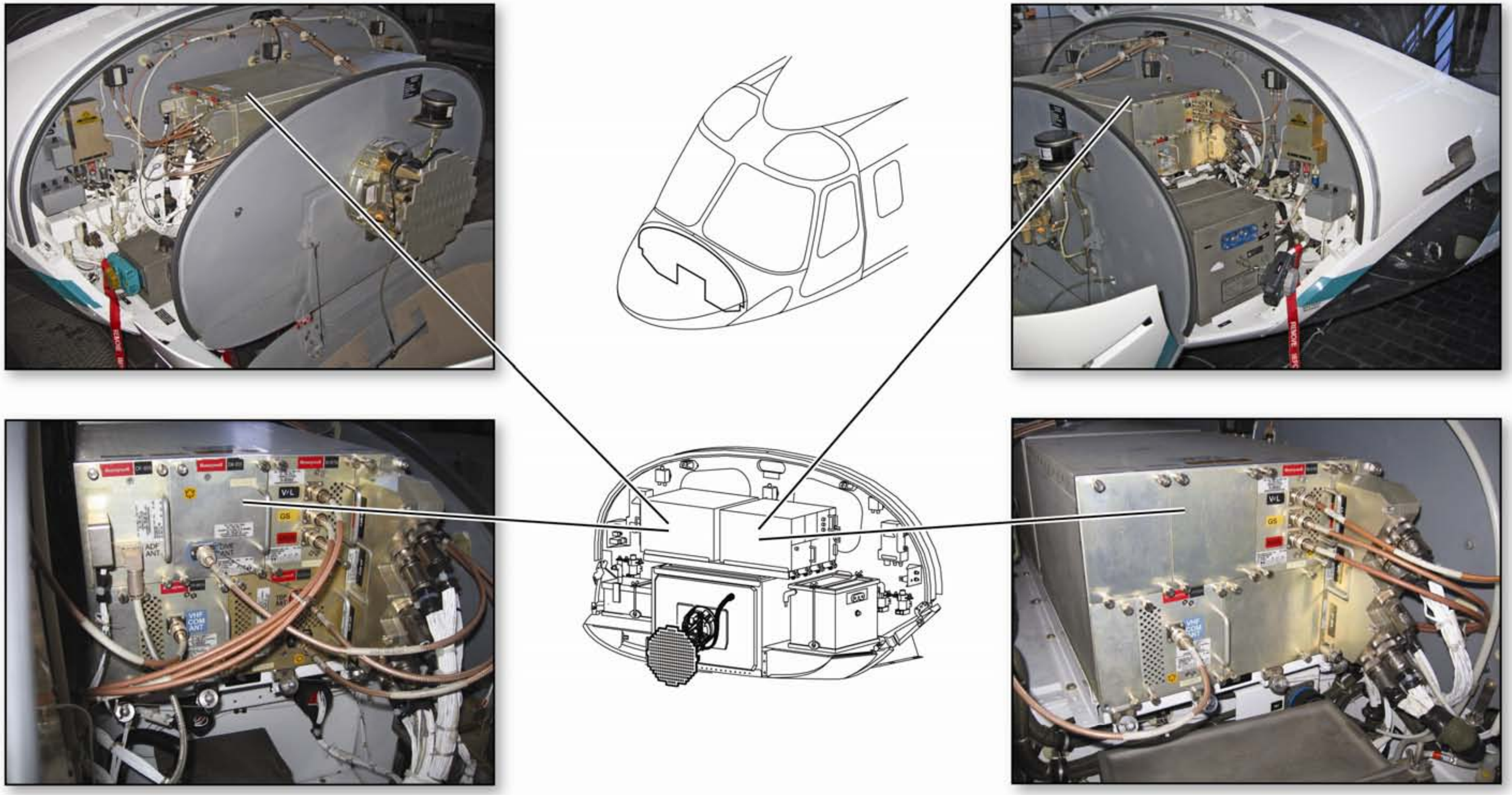
VIDL
VOR/ILS DATA LINK

VDL
VHF DATA LINK (VHF-COMM)

VDR
VHF DATA RADIO

XPDR
TRANSPONDER

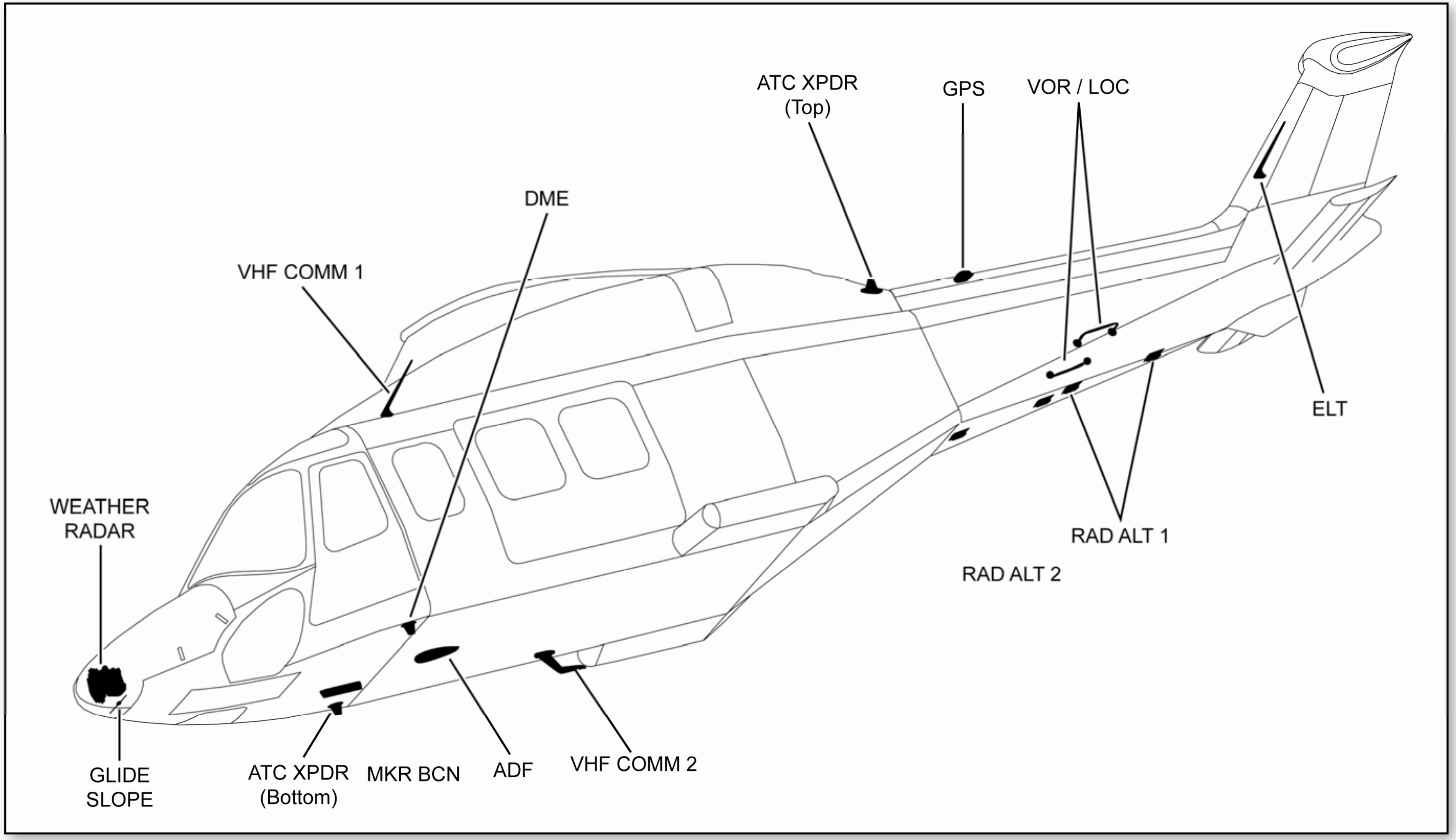
MODULAR RADIO CABINET (MRC) – LOCATION OF MODULES



MODULAR RADIO CABINET 2 (MRC2)

MODULAR RADIO CABINET 1 (MRC1)

MRC 1 AND MRC 2 INSTALLATION



LOCATION OF ANTENNAS

MODULAR RADIO SYSTEM – PRINCIPLE OF OPERATION

The Modular Radio System has standard navigation and communication functions that include VOR, DME, ADF, Instrument Landing System (ILS), Very High Frequency COMmunications (VHF COM) and Air Traffic Control transponder (ATC XPDR). Most functions are contained in the MRC and the NIM is the interface between the radio modules and the other components of the Modular Radio System. Each radio module communicates to and from the NIM through separate serial receive (RX) and transmit (TX) data bus called the Radio Communication Bus (RCB).

The NIM supplies radio data conversion between RCB and ASCB-D in order to supply:

- frequency, channel, code and mode control data to the radios
- NAV and status data to the avionics system

The NIM include a sensor that monitors the temperature of the radio modules and operates the cooling fans to maintain the temperature within limits.

Frequency, channel and mode control of the Modular Radio System are supplied by the Multifunction Control Display Unit (MCDU) and the Cursor Control Device (CCD). The CCD with the PFD or the MCDU can tune, control the modes, and show the displays for the modular radio system. The MCDU is the primary tuning source for the modular radio system.

Radio navigation such as VOR and ADF bearings, ILS deviations, DME distance, and other navigation data can be sent to the MAU and shown on the PFD.

NETWORK INTERFACE MODULE (NIM)

The NIM module most important functions are the following:

- to supply a central processor and an interface between the radio modules in the MRC and the other units of the PRIMUS EPIC® system
- to supply an external interface with the PRIMUS EPIC® system through the ASCB-D and LAN networks
- to receive data from the ASCB-D and LAN and supply data to the radio modules through the Radio Communication Bus (RCB) and viceversa
- to supply analog to digital conversion of radio audio on the digital audio bus and to provide inputs for audio signals from optional equipments like a third COM and high frequency (HF) radio
- to connect the offside MRC to each audio panel in the cockpit through the digital audio bus (the audio bus is really constituted by two audio buses, the audio bus 1 and audio bus 2). This gives the pilot and copilot independent control of both sides of the audio

VIDL MODULE

The VIDL module receives data from ground installations and use these data for landing and taxiing aids. In particular:

- the VOR receiver supplies bearing in degrees to and from the ground stations
- the ILS gives approach and landing navigation guidance information. The ILS radio components include LOC (Localizer), GS, and MKR BCN (Marker Beacon). These radio functions give azimuth, elevation angular deviation and discrete position fixes relative to the runway threshold
- the VDB receiver supplies reception of GPS (Global Positioning system) correction data and runway flight-paths for the GPS landing system

The VOR/LOC receiver operates over the frequency band from 108.00 MHz to 117.95 MHz in 50 kHz increments.

The LOC from 108.10 MHz to 111.90 MHz in 50 kHz increments.

The GS receiver operates over the frequency band 329.15 MHz to 335.0 MHz in 150 kHz increments. The receiver system automatically pairs localizer and glide slope channels to assigned frequencies.

The marker beacon receiver operates at 75 MHz.

Analog NAV audio signals from the module are digitized by the NIM and transmitted on the digital audio bus. Audio outputs from the VIDL include NAV IDENT and MKR tone signals are transmitted from the digital audio bus to each of the two audio panels in the system. The VIDL has an ARINC 429 backup bus for radio tuning when the NIM fails.

DME MODULE

The DME module supplies the data for in-flight navigation, terminal navigation and area guidance. The system measures the distance from the ground station, ground speed and time-to-station.

The DME system operates with radio pulses in the frequency range from 960 MHz to 1215 MHz. The airborne system transmits from 1025 MHz to 1150 MHz to a ground station. The receiver frequency range is from 962 MHz to 1213 MHz.

The DME system is connected to the display system and to the Flight Management System through the ASCB-D bus.

The digital audio bus transmits the DME identification audio signals (IDENT) to each audio panel in the audio system.

ADF MODULE

The ADF module supplies data for in-flight navigation, terminal navigation and area guidance. The ADF uses a narrow-band and wide-band to reduce noise and improve the quality of voice signals during navigation. The selectable modes are ANT, ADF, BFO (Beat Frequency Oscillator) for the narrow band and VOICE for the wide band.

When the ADF is used with the Electronic Flight Instrument System (EFIS) display, the ADF gives the radio-bearing relative to the rotorcraft-heading.

A second ADF module should be installed in the MRC 1 (copilot side) while the single ADF is installed is in the MRC 2 (pilot side).

VHF-COMM (VDL) MODULE

The VHF communications module gives two-way, air-to-air, and air-to-ground communication in the frequency range of 118.000 MHz to 136.975 MHz with 8.33 kHz or 25 kHz channel spacing, which is selectable through the MCDU.

The system has an automatic transmit time-out function to prevent blockage of a communication channel when a push-to-talk (PTT) switch is stuck closed.

The VDR supplies the following voice and data radio functions:

- 8.33 kHz channel ARINC 716 compatible analog voice communications transceiver
- 25 kHz channel ARINC 716 compatible analog voice communications transceiver

The VDR has its own power supply and can operate independently from the NIM. In fact, when a NIM fails the ARINC 429 bus maintains the control of the module. An analog audio backup control is available (headphones) by selecting the backup button on the audio panels. This function bypasses the MIC bus in the NIM and audio panels.

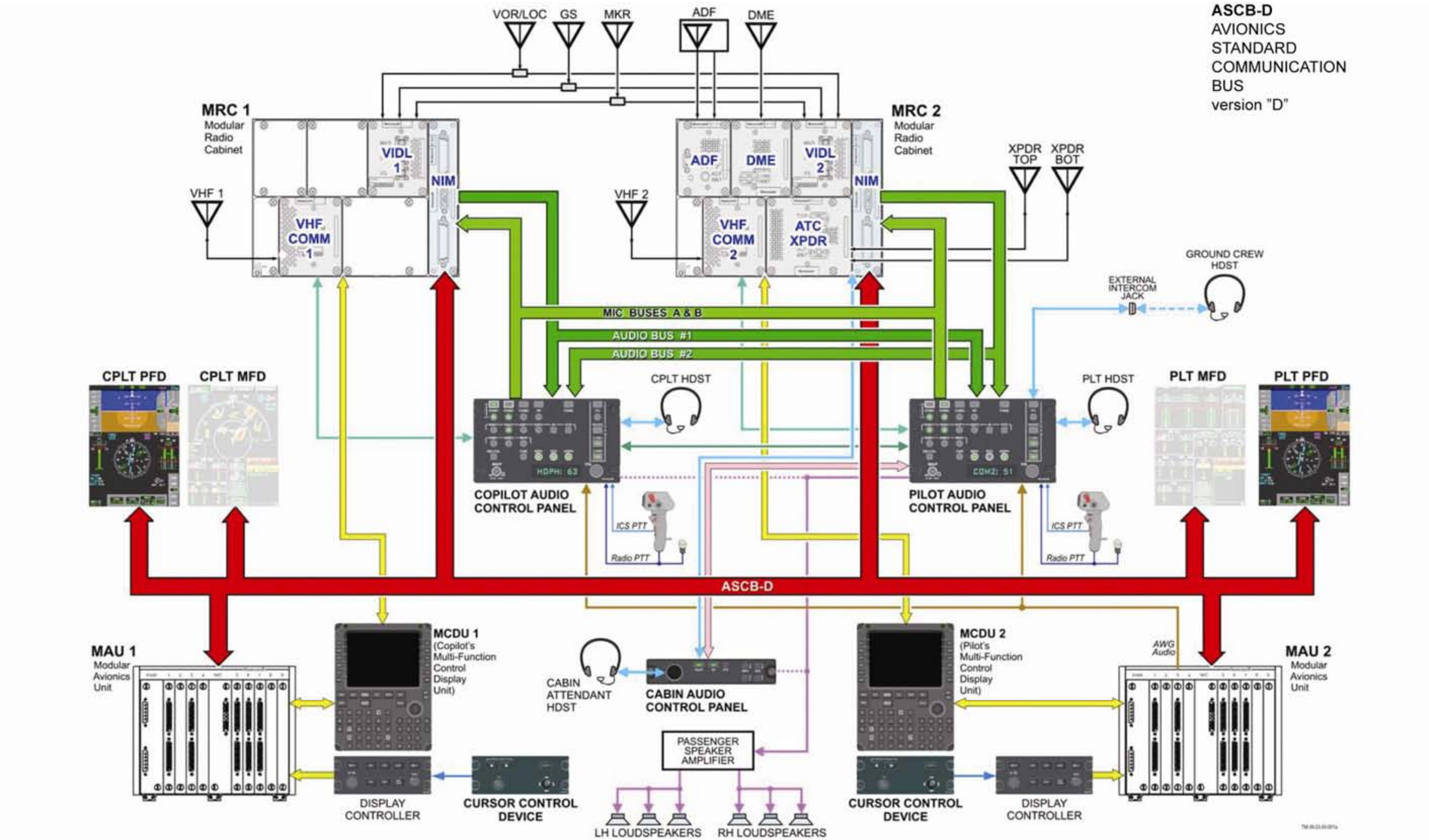
XPDR MODULE

The XPDR module modes of operation are mode A, C and S. Mode S enables secondary surveillance by transmission of aircraft identification information, altitude (barometric) and coded message data to air traffic control (ATC) ground stations and Traffic Collision Avoidance System (TCAS) installations on other aircrafts. The XPDR supports basic downlink aircraft parameters. The NIM supplies barometric altitude data from the Air Data System (ADS). The transponder is provided with an ARINC 429 backup bus in case of the NIM fails.

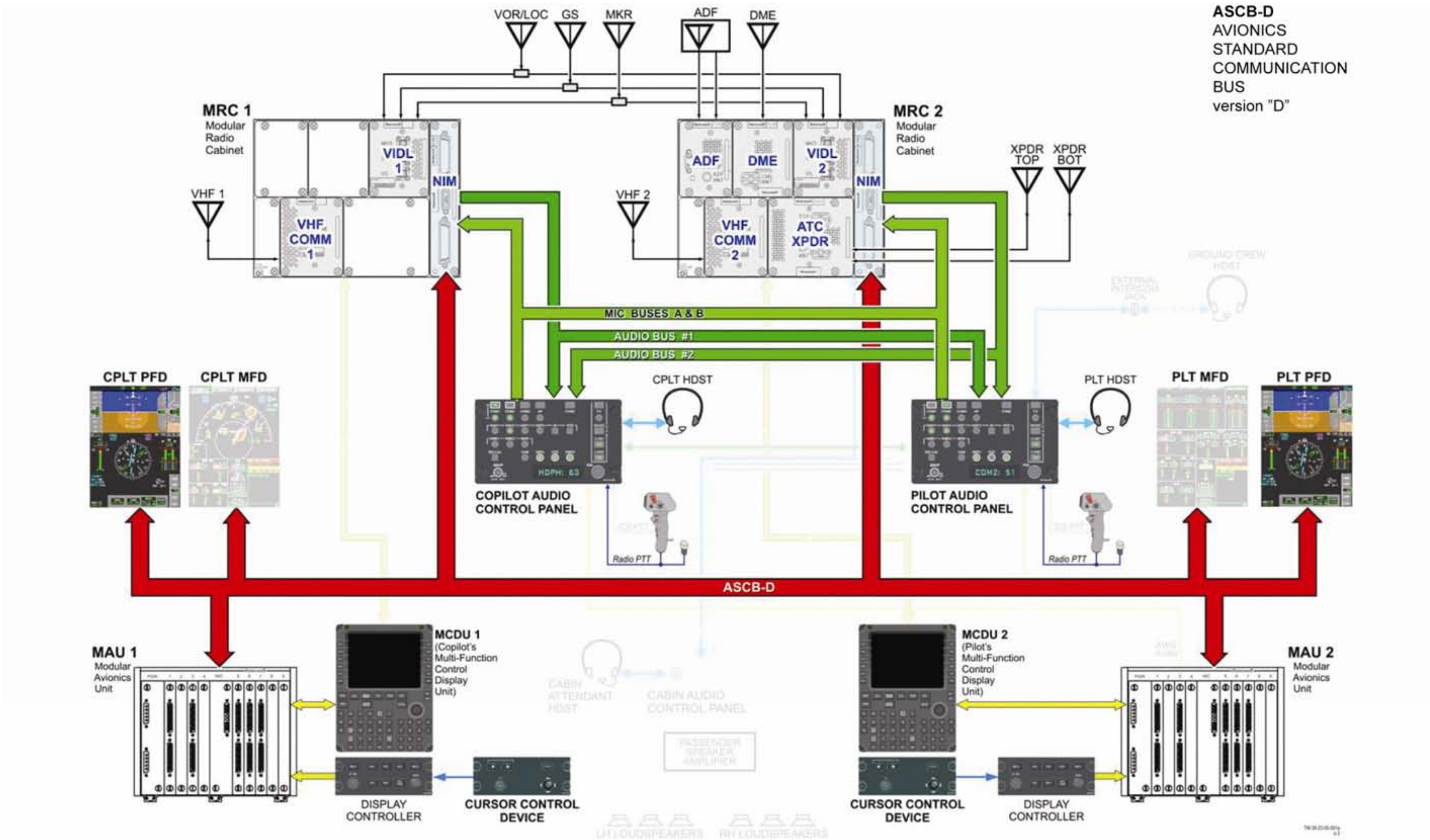
The flight ID information is given by the FMS or it can be entered by the pilot. The XPDR receives the ICAO address programmed into the aircraft personality module (APM) and the pilot enters the squawk code.

The transponder connects to the display system through the ASCB-D, following the same arrangement as the other radio systems.

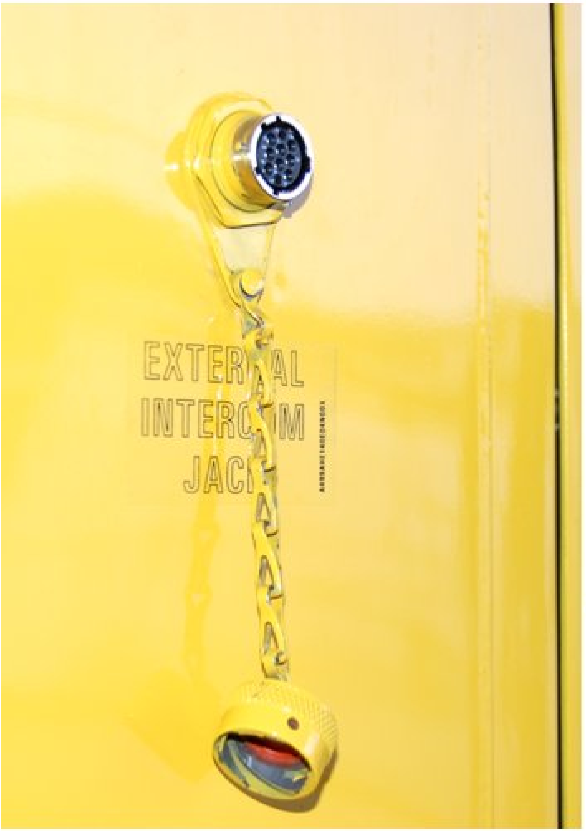
On the following schematics red, light green and yellow larger lines represent digital buses; light blue, green, brown (AWG audio) and purple thinner lines represent analog signal lines. The pink line between the cabin audio control panel and the pilot audio control panel represent a direct connection of the units.



MODULAR RADIO SYSTEM – GENERAL LAYOUT



MODULAR RADIO SYSTEM – NORMAL RADIO OPERATION



EXTERNAL INTERCOMMUNICATION JACK

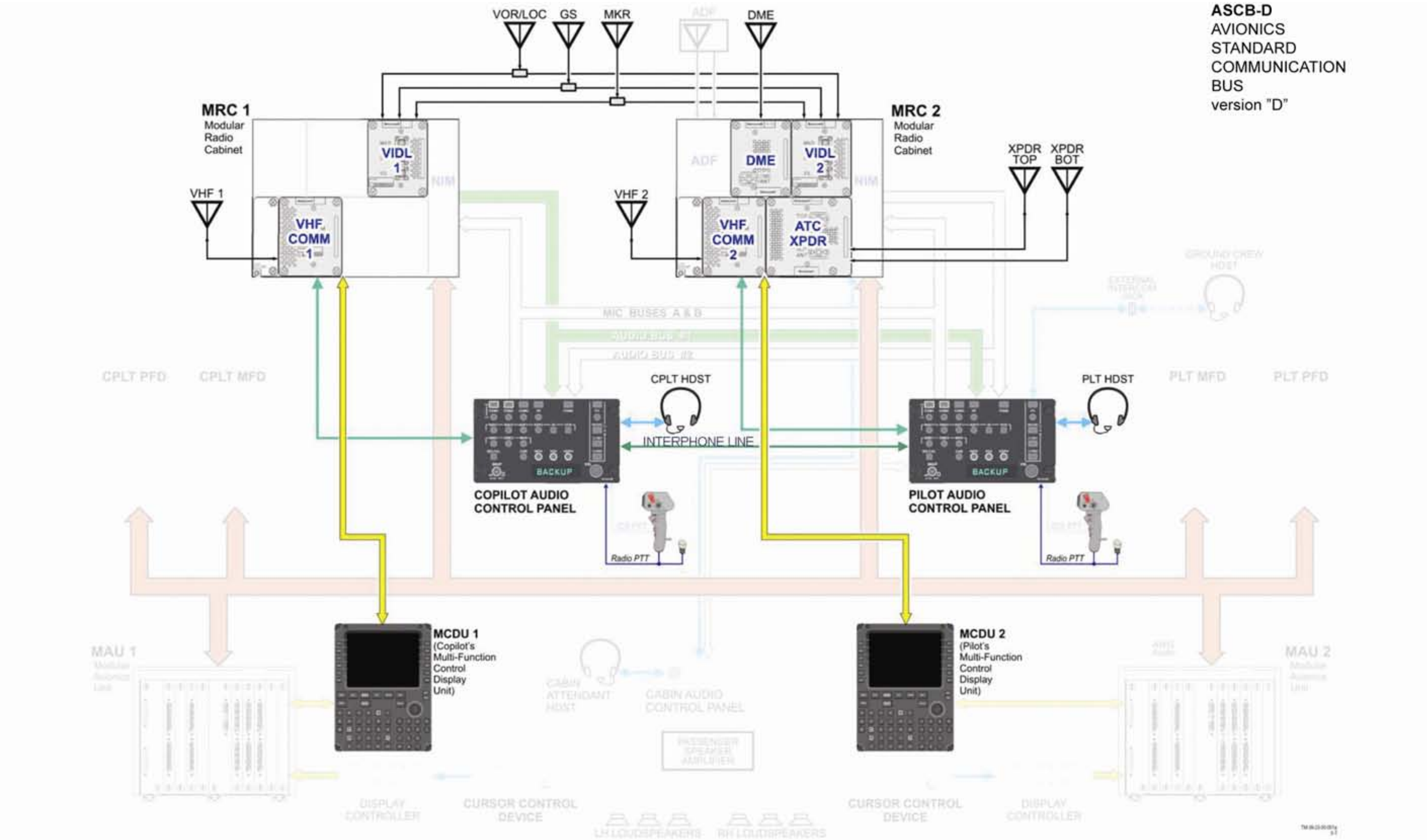
BACKUP ICS MODE

Each audio panel has a backup mode that can be selected when one of the audio panels fail. The backup mode permits the pilot headset to be connected directly to the on--side VHF COMM or off-side audio panel ICS.

No power is required to the on-side audio panel. Switching the pilots MIC between the COM radio and the off-side maintenance port is performed by an external relay controlled by the PTT button.

The audio panel backup mode controls the ICS and COM audio volume through the use of a single volume control knob. The control knob is located on the lower left corner of the audio panel and labelled BKUP.

The ICS and COM audio volumes cannot be adjusted separately.



MODULAR RADIO SYSTEM – BACKUP MODE

MULTIFUNCTION CONTROL DISPLAY UNIT (MCDU) - GENERAL

The Multifunction Control Display Unit (MCDU) is the primary radio controller. The PFD and Cursor Control Device (CCD) can give quick access to change configured COM and NAV frequencies. To access the radio system from the MCDU the pilot has to push the **RADIO** button.

Radio frequencies are displayed along the bottom edge of both PFD displays. Select the radio to be adjusted with the joystick on the CCD. Adjust the frequency using the concentric knobs on the CCD. Radios displayed on the PFD are configured on the radio setup pages in the MCDU.

Any of the components of the radio system that are not supplying valid data to the MCDU or PFD are removed from the associated display and replaced with dashes.

MCDU – PRINCIPLE OF OPERATION

The MCDU uses Line Select Buttons (LSB) adjacent to the display to simplify its operation. Any selectable parameter an be changed. Push the corresponding line select button to place the cursor box around the parameter on the display. Rotate the dual concentric tuning knobs to change the boxed parameter value or load a new value directly from the keypad. The MCDU has a menu paging system with edge control buttons. The sections are used to identify and control both the frequency and operating mode of the radio functions.

Color of characters is used with two font sizes. The active choice is shown in the larger font and in a color other than white. Display pages are normally white. Selections are in larger **green characters** and annunciators can be in green, amber or red.

In normal use, the line select button adjacent to the displayed function executes the function described by the prompt. The top line is the page title.

The scratchpad is used to enter data from the keypads. For example, the pilot enters **124.45** using the keypad. The scratchpad shows **124.45**. When the pilot tries to transfer the frequency to a NAV radio, the frequency is replaced with **Invalid data** annunciator in the scratchpad (such as, 124.45 is not a valid NAV frequency). Pushing the CLR button deletes the **Invalid data** annunciator and restores **124.45** to the scratchpad. When the pilot transfers **124.45** to a VHF COM radio, the frequency is valid, the transfer is made and the scratchpad is cleared.

Control prompts help the pilot navigate through the pages. They indicate what action is required and give some indication of the expected result once the prompt is activated. The prompts are located in the display columns immediately next to the line select buttons. The prompts are white.

When the scratchpad is empty, pushing a line select key moves the format cursor to the adjacent field or performs the function indicated by the icon that appears near the key. The icons and their functions are described in the following table.

RADIO - CONTROLS AND INDICATORS

1. RADIO pushbutton

   pressed …….. displays the RADIO 1/2 page

2. PREV pushbutton

   pressed …….. allows to return to the previous page

3. LINE SELECT KEY

   pressed …….. allows to operate the MCDU menu items

4. ALPHA keypad

   pressed …….. allows to insert alphabetical characters in the scratchpad

5. NUMERIC keypad

   pressed …….. allows to insert numerical characters in the scratchpad

SWAP FREQUENCIES

The symbol indicates exchanges between the active and preset frequencies for the associated radio that can be made. This effectively saves the currently active frequency in the preset memory and tunes the radio to the frequency previously stored as the preset.

PAGE INDICATOR

When the icon is displayed, pushing the adjacent Line Select Key (LSK) changes the display to another page. The page to be displayed is labelled explicitly or it is a detail page for the radio in the associated field.

EXCLUSIVE SELECTION

The icon is displayed next to a list of mutually exclusive options. Each time the adjacent LSK is pushed, the next item in the list is selected, wrapping around to the first when the last option is reached.

The selected value is displayed in **green large** font.

The unselected values are displayed in **small white** fonts.

IMMEDIATE FUNCTION

The icon indicates the function identified in the field executed immediately after the associated LSK key is pushed.

COPY VALUE

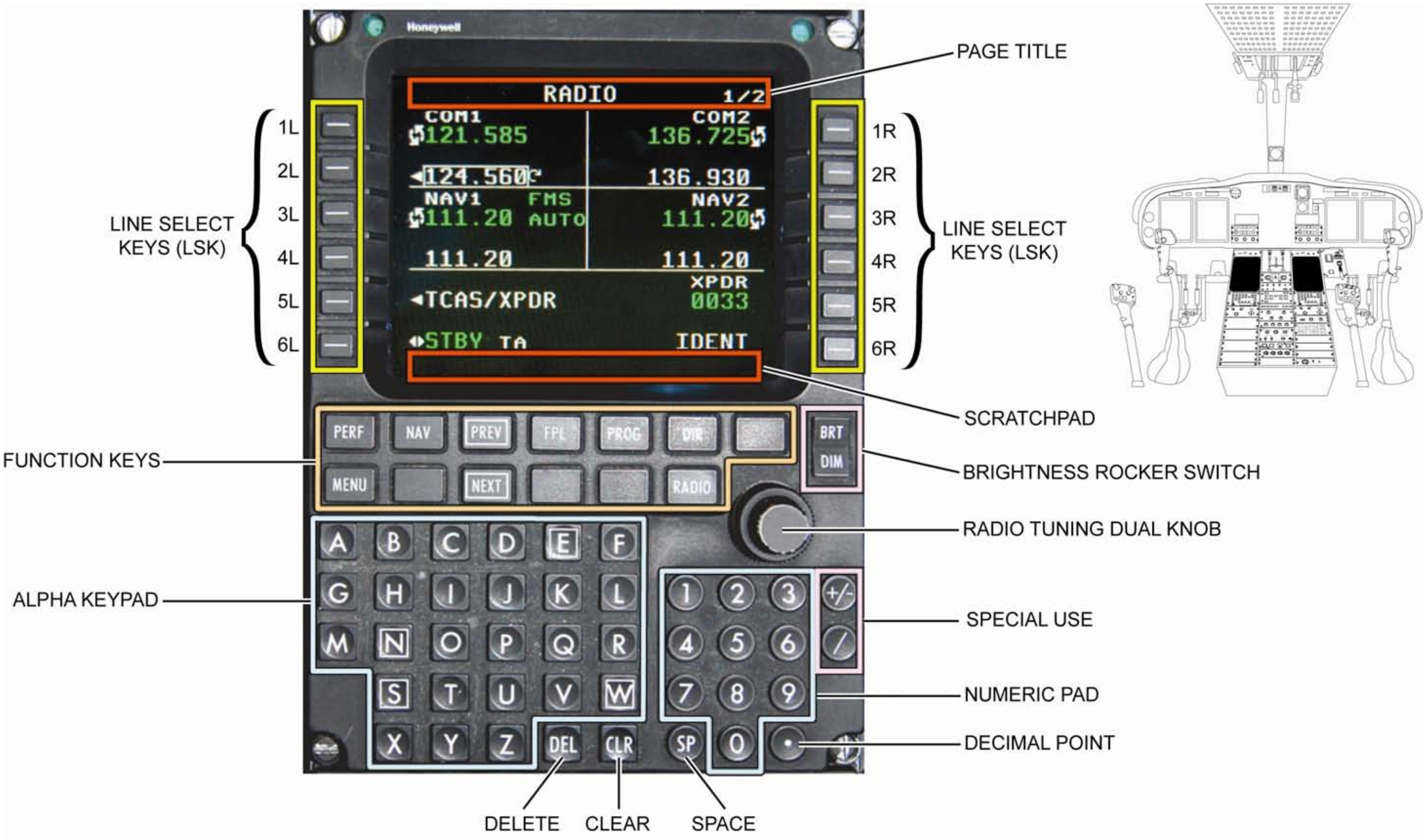
The icon is used on the memory pages to indicate that the frequency highlighted by the cursor is copied into the active frequency for the associated radio.

CURSOR

The cursor box highlights the value in the active field.

TUNING CURL

The icon indicates that the data value highlighted by the format cursor can be changed by turning the MCDU tuning knob.



MULTIFUNCTION CONTROL DISPLAY UNIT (MCDU)

MCDU DISPLAY PROMPTS

Three types of radio frequencies can be displayed:

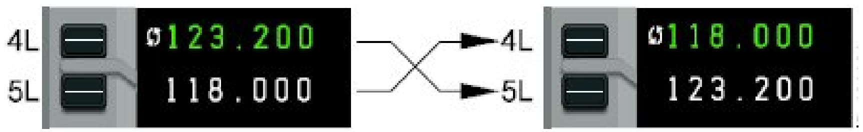
- Active frequency is the one that the radio is currently set to for receiving or transmitting
- Standby frequency is the one waiting to be used next. The Standby frequency can be changed using the tuning knobs or the scratchpad
- Memory frequency is the one with a list of frequencies that is stored for recall

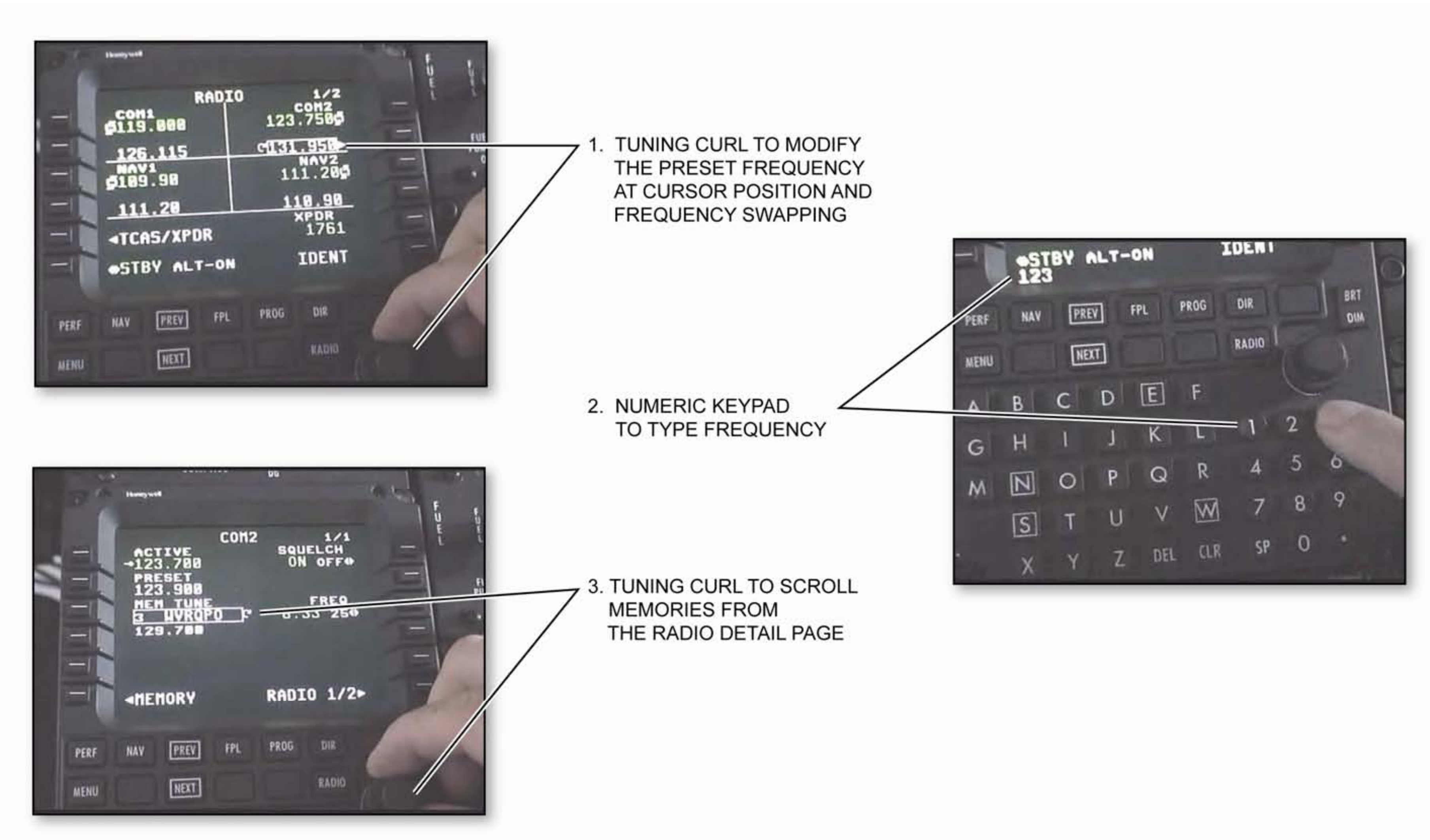
Two swap functions are used. The first one uses the standby frequency. The second one uses the memory frequency.

- With the cursor around the standby frequency and the swap icon displayed when the line select key is pushed, the standby and active frequencies are switched.
- With the cursor around the Memory frequency and the swap icon displayed when the line select key is pushed, the Memory and Active frequencies are switched.

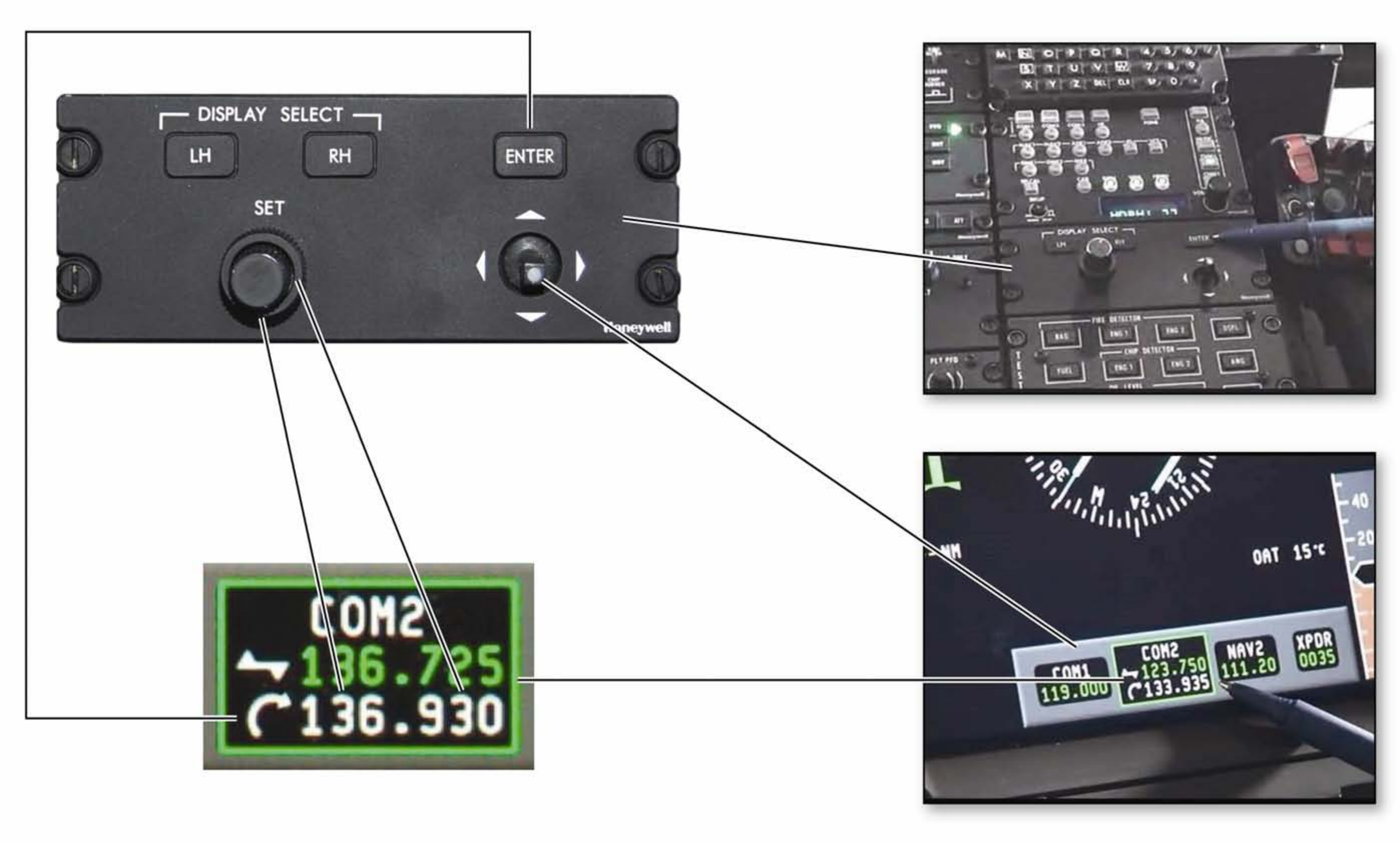
To make the standby frequency active, push the 4L button next to the active COM frequency. This swaps the Standby and Active frequencies.

VHF COM and HF COM radio pages use Active, Standby, and Memory frequencies. They do not permit the use of the tuning knobs to change the Active frequency.





RADIO TUNING USING MCDU

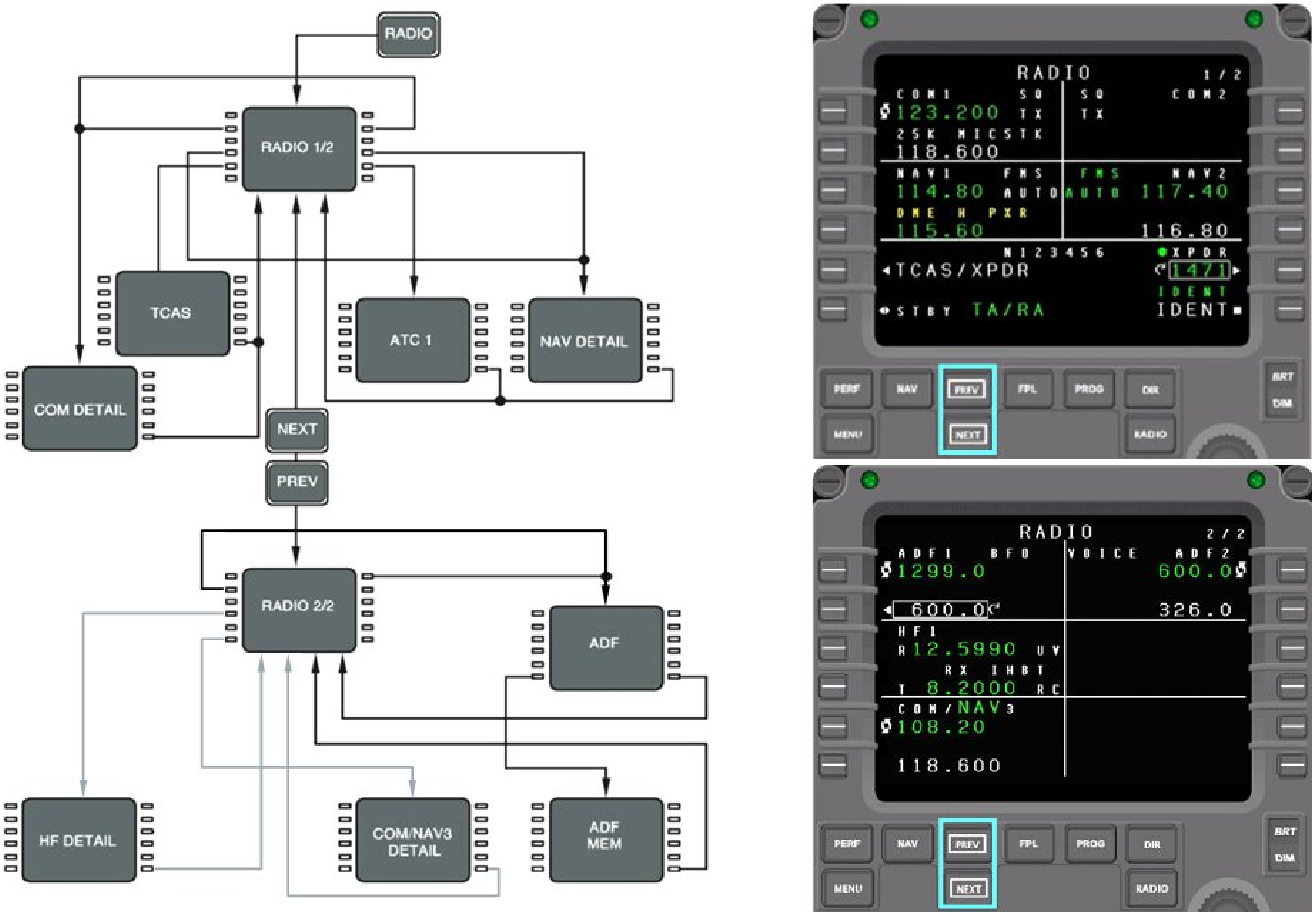


RADIO TUNING USING CCD

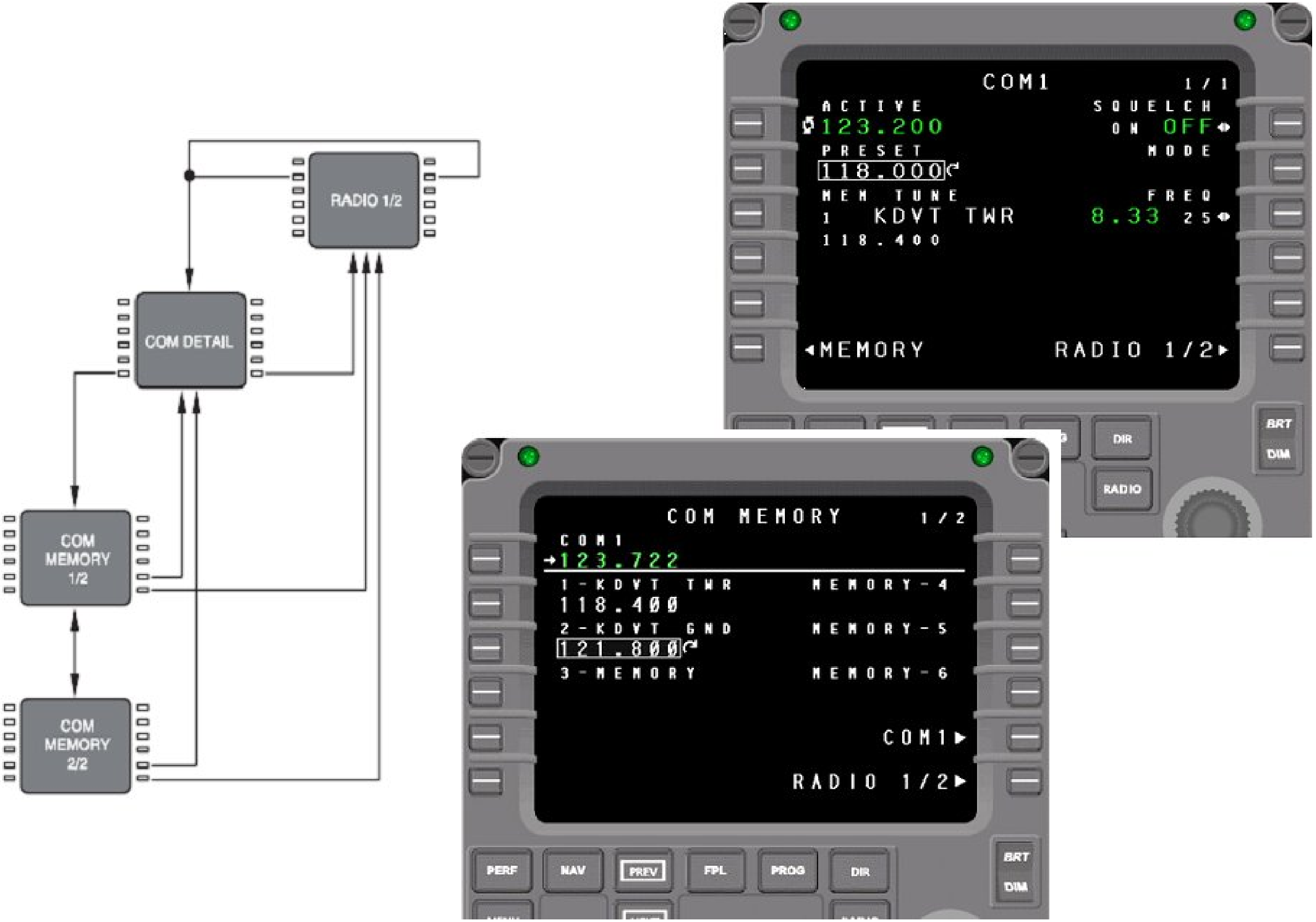
RADIO TUNING

The radio tuning function is accessed by pushing the RADIO button on the MCDU. That shows the RADIO 1/2 page. All other pages are accessed from the RADIO 1/2 page using the LSK or the NEXT and PREV function buttons as shown in the following figures.

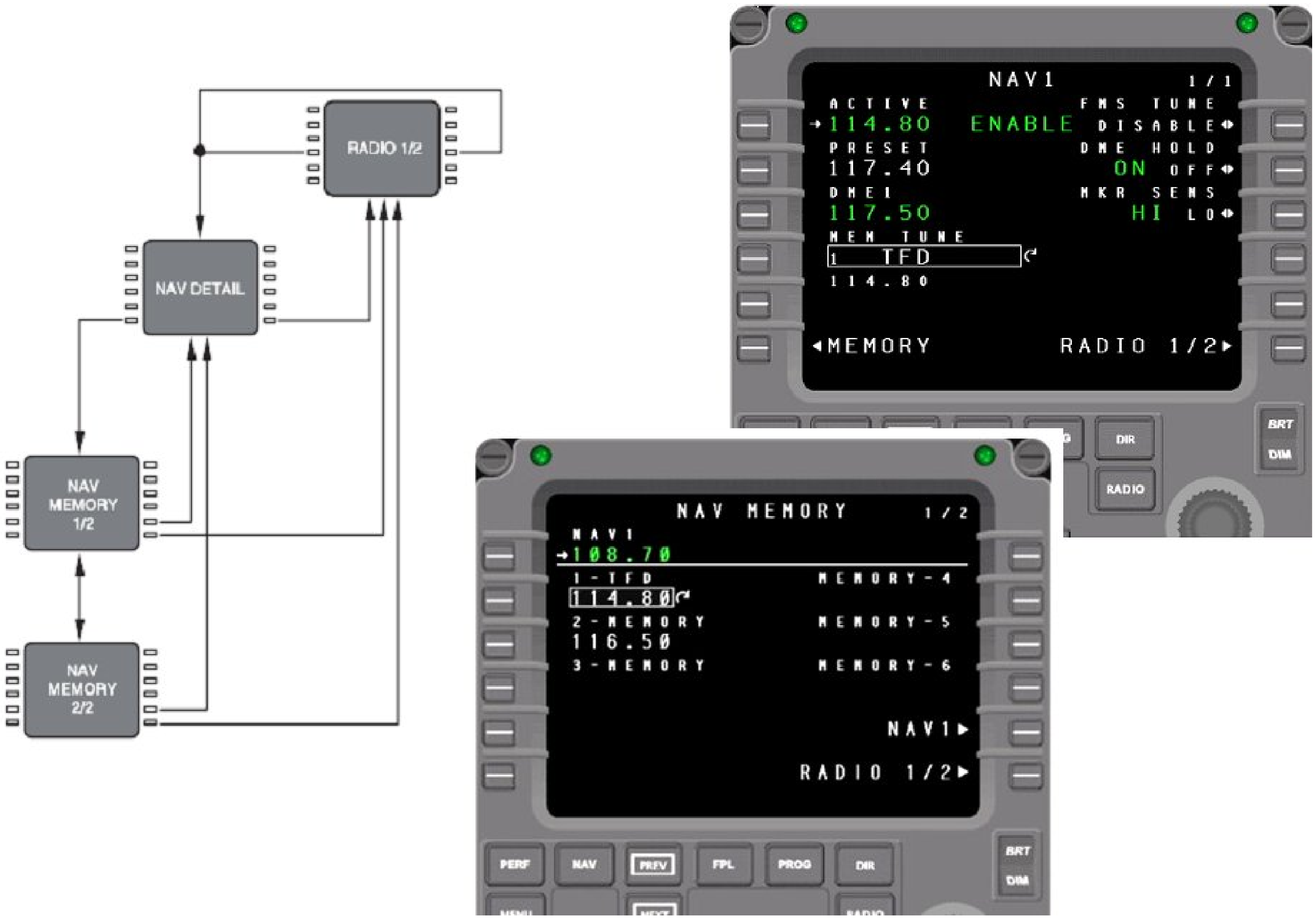
The HF and COM NAV3 apparatus shown faded the figure are optional.



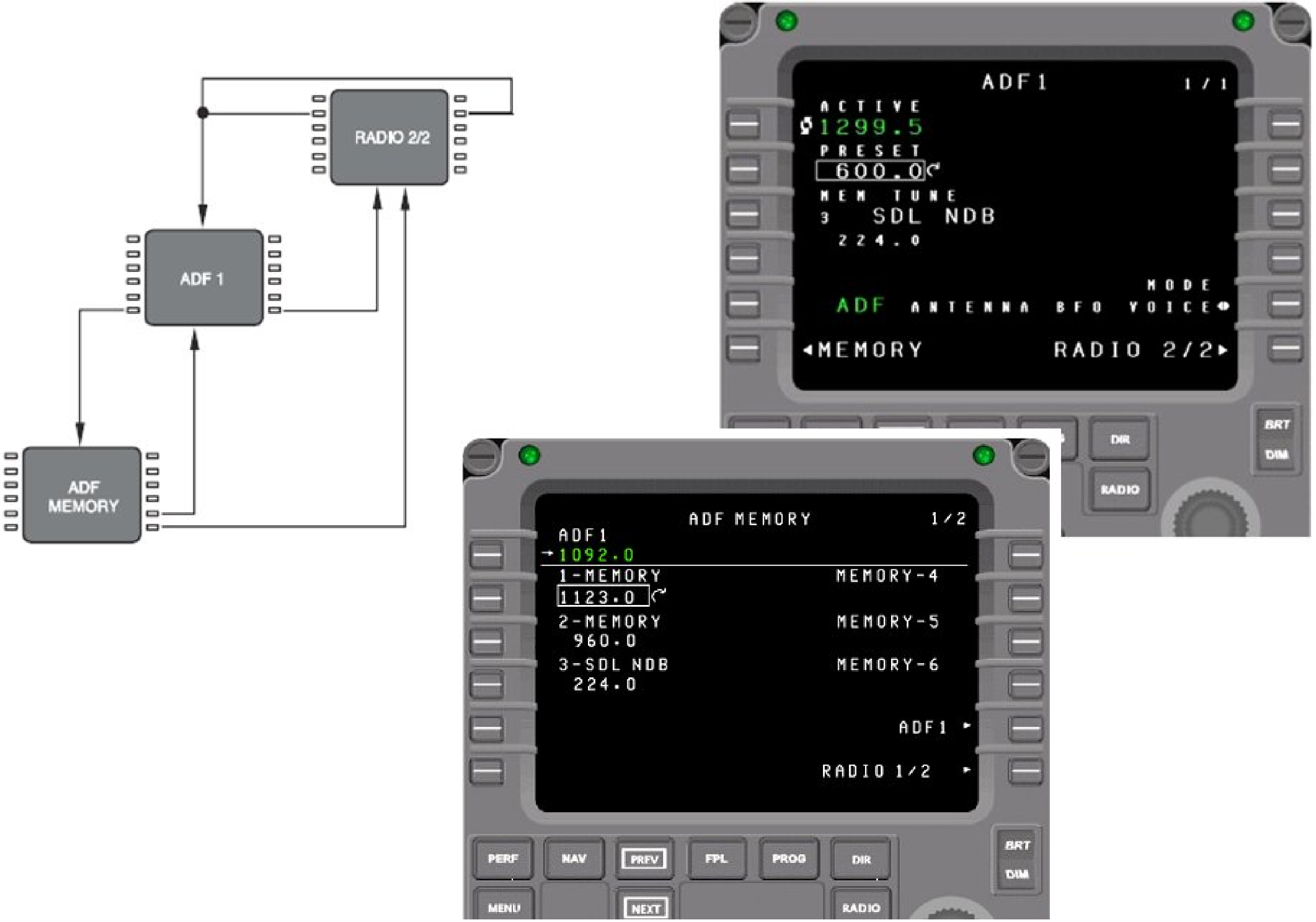
VHF RADIO PAGES



VHF COM PAGES



VHF NAV PAGES


RADIO 2/2
ADF 1
ADF MEMORY
ADF1 1/1
ACTIVE
1299.5
PRESET
600.0
MEM TUNE
3 SDL NDB
224.0
MODE
ADF ANTENNA BFO VOICE
MEMORY
RADIO 2/2
BRT
DIM
DIR
RADIO
ADF MEMORY 1/2
ADF1
1092.0
1-MEMORY
1123.0
2-MEMORY
960.0
3-SDL NDB
224.0
MEMORY-4
MEMORY-5
MEMORY-6
ADF1
RADIO 1/2
PERF
NAV
PREV
FPL
PROG
DIR
MENU
NEXT
RADIO
BRT
DIM

ADF PAGES

Annunciator	Description
25K	Indicates that the associated VHF COM radio is set to 25 kHz frequency spacing. When not present, the radio is tuning with 8.33 kHz frequency spacing. This is selected on the COM 1/2 page.
DME H xxx	This alert indicates that the VHF navigation radio is tuning the corresponding DME receiver independently of the primary navigation frequency.
IHBT	This annunciator indicates that tuning of the radio is inhibited, usually from a remot source (such as an emergency tuning function).
MICSTK	Indicates that the microphone button on the radio has been down long enough that the radio has identified it as stuck in the transmit position.
SQ	This annunciator indicates that the squelch feature for the radio is active. Squelch is set turned ON and OFF by the COM 1/2 page.
TX	This annunciator indicates that the radio is currently transmitting.
FMS AUTO	This annunciator indicates that the autotuning feature is active with the VHF NAV radio.
●	This annunciator indicates that the XPDR is replying to interrogations.
IDENT	This annunciator indicates that the XPDR is transmit-ting IDENT replies.

VHF RADIO ANNUNCIATORS

ALERT	DESCRIPTION
ANT	The ADF radio is in antenna mode.
BFO	The ADF radio is operating in BFO mode.
CW	This annunciator indicates that the radio is in a continuous wave mode.
VOICE	The ADF radio is in voice mode.

ADF RADIO ANNUNCIATORS

CURSOR CONTROL DEVICE (CCD) – CONTROLS AND INDICATORS

1. DISPLAY SELECT button

RH pushed …….....	gives the pilot control of the system and moves the cursor to the PFD
LH pushed ……….	moves the CCD cursor from PFD to MFD

2. ENTER button

…………………….	activates or deactivates the selected submenu function

3. JOYSTICK

	allows to position the cursor within the selected Display Unit (DU). Moving the joystick to the left-of-center or right-of-center moves the cursor left or right, respectively, in the active display page
…………….	Moving the joystick forward or aft of center moves the cursor up and down, respectively, on the active display page. The cursor movement (left, right, up, and down) on the display page permits the pilot to select and control functions on the MFD and PFD. The joystick moves the cursor through the MFD menu selections and operates the MFD designator. It also is used for the radio tuning function.

4. dual concentric rotary knob SET knob

inner knob ……….	MFD selected: scrolls the CAS messages PFD selected: adjusts decimals of the highlight standby frequency or the last two digits of XPDR code
outer knob ……….	MFD selected: sets the scale range (Map or Plan formats) or the highlighted value in the drop-down menu PFD selected: adjusts units of the highlight standby frequency or the first two digits of XPDR code

CURSOR CONTROL DEVICE – CONTROLS AND INDICATORS

PRIMARY FLIGHT DISPLAY (PFD) RADIO

The PFD is a secondary means of radio tuning the selected COM/NAV channels and the transponder.

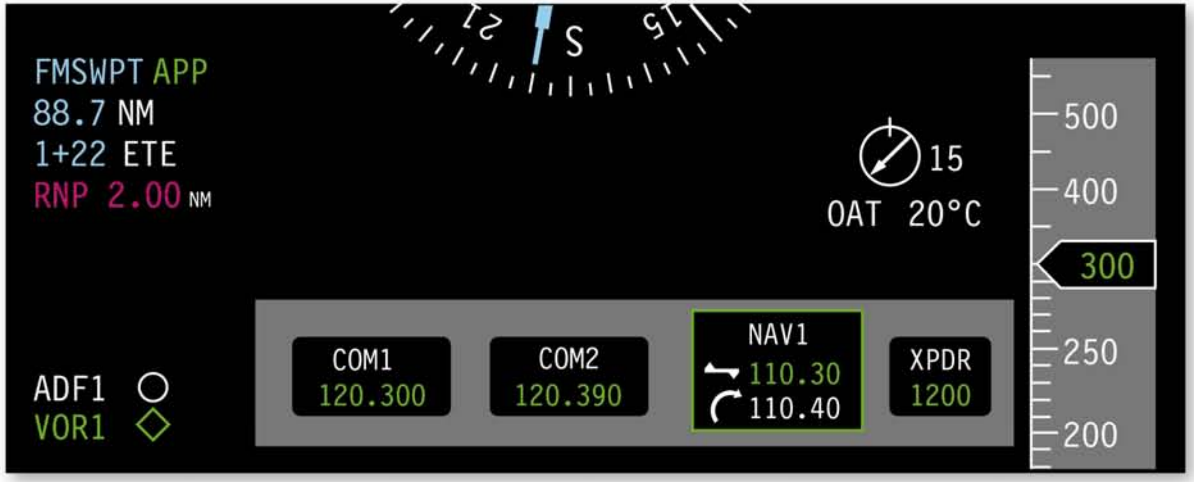
The COM and NAV active and standby frequencies are shown in boxes at the bottom of the PFD display.

The active frequency is displayed in green (110.30).

The standby frequency is displayed in white (110.40).

The purpose of this is to meet the availability requirements for the failure of a loss of all NAV and COM. In addition, it is to increase pilot awareness of the state of the primary NAV and COM.

The frequencies are selected by the CCD. The item currently selected is shown in an enlarged and highlighted box (the PFD cursor). The frequencies are tuned using the CCD knobs. The outer knob is used for radio frequency tuning integer portion and the inner knob is used for the decimal portion. The CCD ENTER button selects a tuned frequency and swaps the active and standby frequencies.



PFD RADIO TUNING

AUDIO INTEGRATING SYSTEM - GENERAL

The Audio Integrating System connects the communication and navigational radios to the radio headsets, microphones and loudspeakers of the helicopter.

The Audio Integrating System includes the Airborne Audio System that transmits communication and navigational audio data through two isolated buses that gives a high level of failure protection.

If one Audio panel fails or is set to back-up mode, the system can supply the pilots with the ICS and on-side radio receive-transmit (using PTT) which takes the role of emergency communications.

There are two audio panels mounted at the top left and right of the cockpit center console in the AW139/AB139, one for the pilot and one for the copilot.

AUDIO INTEGRATING SYSTEM – MAIN COMPONENTS

NETWORK INTERFACE MODULE (NIM)

The NIM module has been already described. Here we remember that the NIN connects the offside MRC to each audio control panel in the cockpit through the digital audio bus. This gives the pilot and copilot independent control of both sides of the audio.

AUDIO CONTROL PANEL

The audio control panel is provided with four rows of round audio selection buttons, grouped side-by-side that allow the pilot to receive audio signals from the selected radio. Multiple radios can be selected enabling the pilot to maintain listening on other radio frequencies while communicating on a selected primary frequency.

At the centre of each button a green annunciator alerts the pilot to the condition of the radio.

The microphone selection button connects the pilot to the associated radio enabling voice communications and lighting the green annunciator.

Pushing the button a second time, the pilot disconnects the radio disabling voice communications and the green annunciator lights off.

Different audio control panel are shown in the figure.

STANDARD CONFIGURATION

EMS CONFIGURATION

AUDIO CONTROL PANEL

AUDIO INTEGRATING SYSTEM – PRINCIPLE OF OPERATION

The audio system permits switching microphones to various radios, distribution of audio to headsets and interphone functions.

The primary interfaces in the audio system are the two digital audio buses from the MRC (one from each MRC) and the two digital microphone (MIC) buses. Both MIC buses are connected to each Audio Control Panel and MRC in the system and contain identical data. The audio panel is the central point of access for controlling the audio system. The audio panel supports connection of up to eight crew-members audio panels on the digital audio and microphone buses.

It converts digital audio signals from the communication and navigation radios into analog signals that are audible in headphones, speakers and cockpit voice recorder (CVR) outputs.

The audio panel supports non-integrated radios such as the high frequency communications (HF COMM) radio.

Applying power to the avionics system powers up the COM, NAV, XPDR, DME, and ADF radios along with the audio panel and each MCDU simultaneously.

The presence of a single volume knob implies that the selection of an audio channel is required before the volume can be adjusted or before it can be deselected.

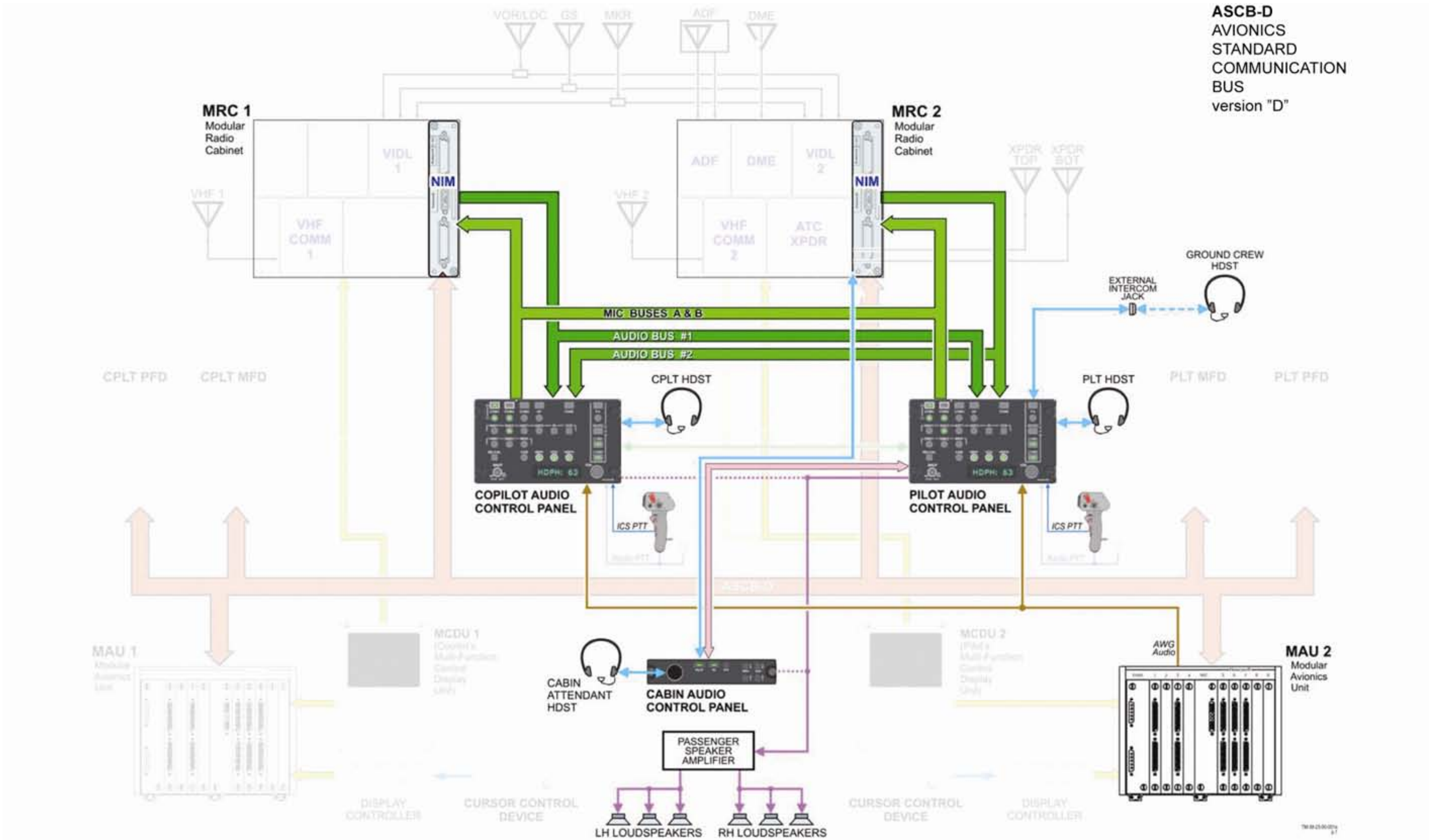
One microphone transmit selection is possible at a time, but listening on more than one audio channel at a time can be done. The default volume adjustment selection that appears in the LCD window is the headphone volume.

The operativity follows the subsequent steps:

1. The pilot selects COM1 to transmit and receive by pushing the MIC and adjusts the volume. After 15 seconds without further knob adjustments it defaults to the headphone (HDPH) in the LCD window.

2. The pilot tunes another frequency on COM2 and requests and receives a frequency change approval. The pilot selects the COM2 MIC button on the audio panel and adjusts the COM2 volume.

3. Suppose the pilot wants to monitor transmissions on COM1. This is accomplished by selecting the audio (AUD) button of COM1. The COM1 volume can be adjusted. With this setting, the pilot can listen to the transmission on COM1 and listen and transmit on COM2.



MODULAR RADIO SYSTEM – ICS & PASSENGER ADDRESS (PA)

AUDIO SYSTEM - CONTROLS AND INDICATORS

1. COM1 button

   ……………... no. 1 VHF communications radio. Pushing the COM1 AUD button activates/deactivates COM1 audio

2. COM2 button

   ……………... no. 2 VHF communications radio. Pushing the COM1 AUD button activates/deactivates COM1 audio

3. COM3 button

   ……………... no. 3 VHF communications radio. Pushing the COM1 AUD button activates/deactivates COM1 audio

4. HF button

   ……………... pushing the HF button activates and deactivates the HF audio (option)

5. FONE button

   ……………... the FONE microphone selection is for a full duplex communication device such as a SATCOM and does not have an audio selection button that is independent of the microphone. When a SELCAL call is received by any of the radios configured to receive SELCAL, the SELCAL button and the corresponding microphone button for that radio flashes

6. NAV1 button

   ……………... pushing the NAV1 button activates and deactivates audio from the navigation radio no.1

7. NAV2 button

   ……………... pushing the NAV2 button activates and deactivates audio from the navigation radio no.2

8. ADF1 button

   ……………... pushing the ADF1 button activates and deactivates audio from the ADF no.1

9. ADF2 button

……….……... pushing the ADF2 button activates and deactivates audio from the ADF no.2

10. ID button

………….…... the ADF and NAV audio filter attenuates the voice audio so the Morse Code ident can be prominently heard.

11. VCE button

…………..…... the ADF and NAV audio filter attenuates the IDENT audio so the voice audio or Morse Code IDENT can be prominently heard

12. DME1 button

……………..... activates/deactivates audio from the DME source no.1

13. DME2 button

……………….. activates/deactivates audio from the DME source no.2

14. MRK (Marker Beacon) button

……..……... activates and deactivates audio from the marker beacons

15. SELCAL (Select calling) button (option)

…………..…... the radios can be configured with select calling capability. When a SELCAL is received, the SELCAL button and the annunciator light for the proper radio flashes. The SELCAL function decodes all the VHF and HF COMM digital audio signals

16. CAB (Cabin) Intercommunication System (ICS) volume control button

……………..….. connects pilot to the cabin ICS. Pushing the PILOT button on the cabin audio panel flashes the CAB button annunciator light. The CAB button on the pilot-side audio panel flashes and generates a single tone that is sent through the PA system and to the cabin audio controller.

The acknowledgement of the cabin call by the cabin audio controller is indicated as a fast flash on the CAB button. When the CAB button is flashing fast, it establishes an audio connection between the cockpit ICS channel and cabin audio controller. This is indicated by a steady lighting of the CAB annunciator. The annunciator on the CAB button slowly flashes and a single audio tone sounds in the pilots' headset when a request for cabin ICS is initiated by the cabin audio system.

When the pilot pushes the CAB button after a call from the cabin, it establishes an audio connection between the cockpit and cabin. This is indicated by a steady light.

17. INPH Pilot Intercommunication System (ICS) volume control button

……………….... connects and disconnects the pilot to the ICS

18. VOX Voice- Activated Squelch (VOX) system button

……………….. turns the VOX system and the associated annunciator light ON and OFF. With the VOX system ON, tuning the VOL knob in the lower right corner of the audio panel increases or decreases the VOX system sensitivity

19. HDPH (Headphone) Pilot Master volume control button

……..………... turns the headphone master volume control ON and OFF. With the master volume control system ON, turning the VOL knob in the lower right corner of the audio panel increases or decreases the volume of all the active radios simultaneously

20. PA button

………………... Pushing the PA button connects and disconnects the pilot microphone to the passenger compartment public address system. Pushing the button turns OFF any other previously connected microphone.

21. MUSIC button

……….……... Pushing the MUSIC button connects the cabin crew or passengers to the cabin entertainment system.

22. CHM1 – CHM2 button

……….……... When the CHM1 or CHM2 button is selected, the audio system turns ON the corresponding button annunciator and activates cabin visual annunciators, such as seat belts. These two buttons toggle ON and OFF. The chime buttons operate when CHM1 is pushed on the pilot--side audio panel, the CHM1 button on both the audio panels lights. The same is true for CHM2. When one pilot turns a chime OFF, the annunciator goes out on both panels.

23. VOL control knob

……….……... changes the audio volume of a radio.

1. Push the button associated with the desired radio to do the following:

- Annunciate the selected radio and current volume setting in the display window
- Activate audio
- Light the green annunciator on the button

2. Change the volume of the selected radio by turning the volume control knob clockwise (CW) to increase volume and CCW to decrease volume.

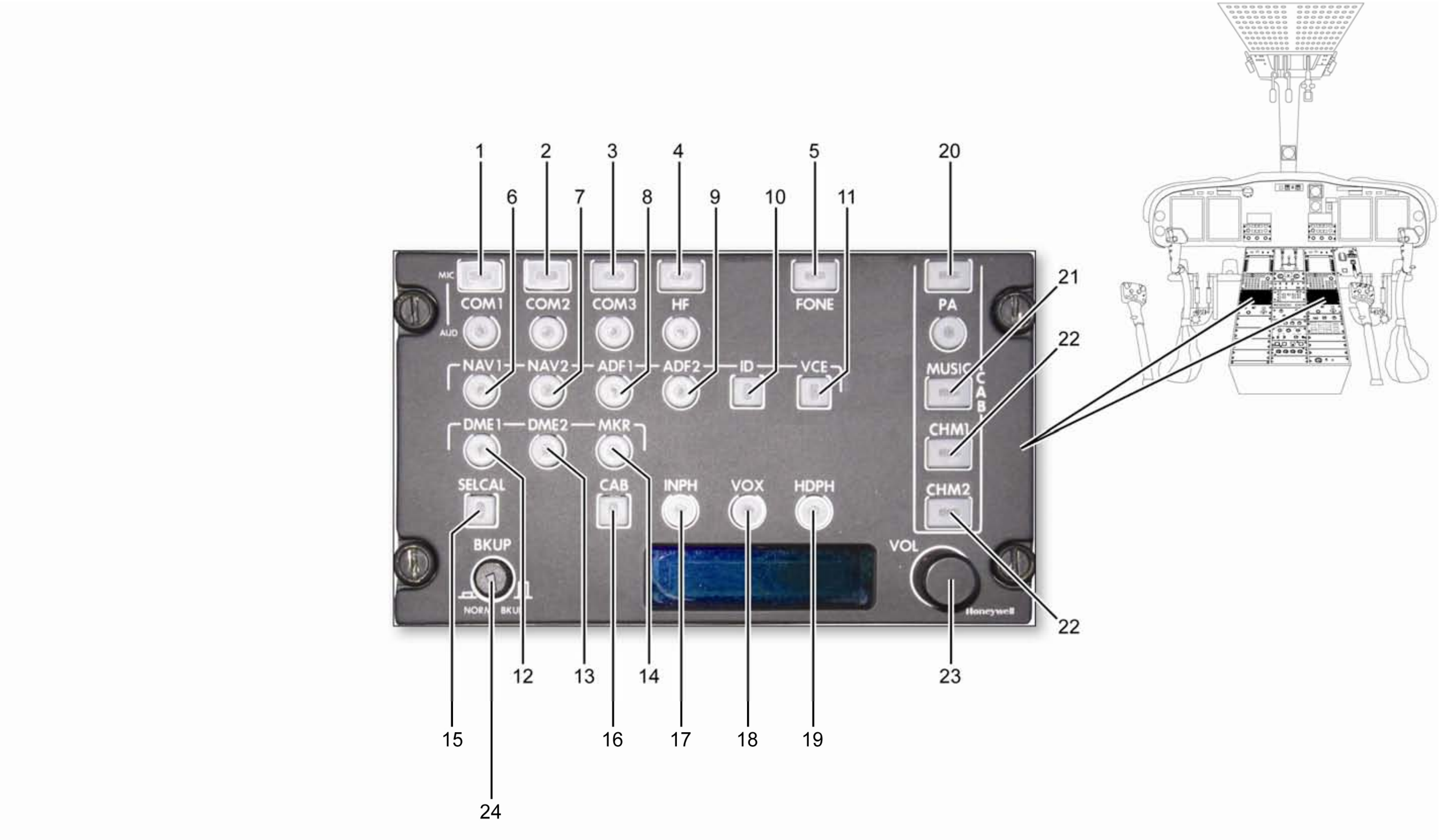
NOTE. The volume control knob controls the volume of the radio displayed in the control window. The exception is selecting the headphone (HDPH) button. When **HDPH** is displayed in the control window, the volume control knob adjusts the volume of all selected radios simultaneously.

24. BKUP button

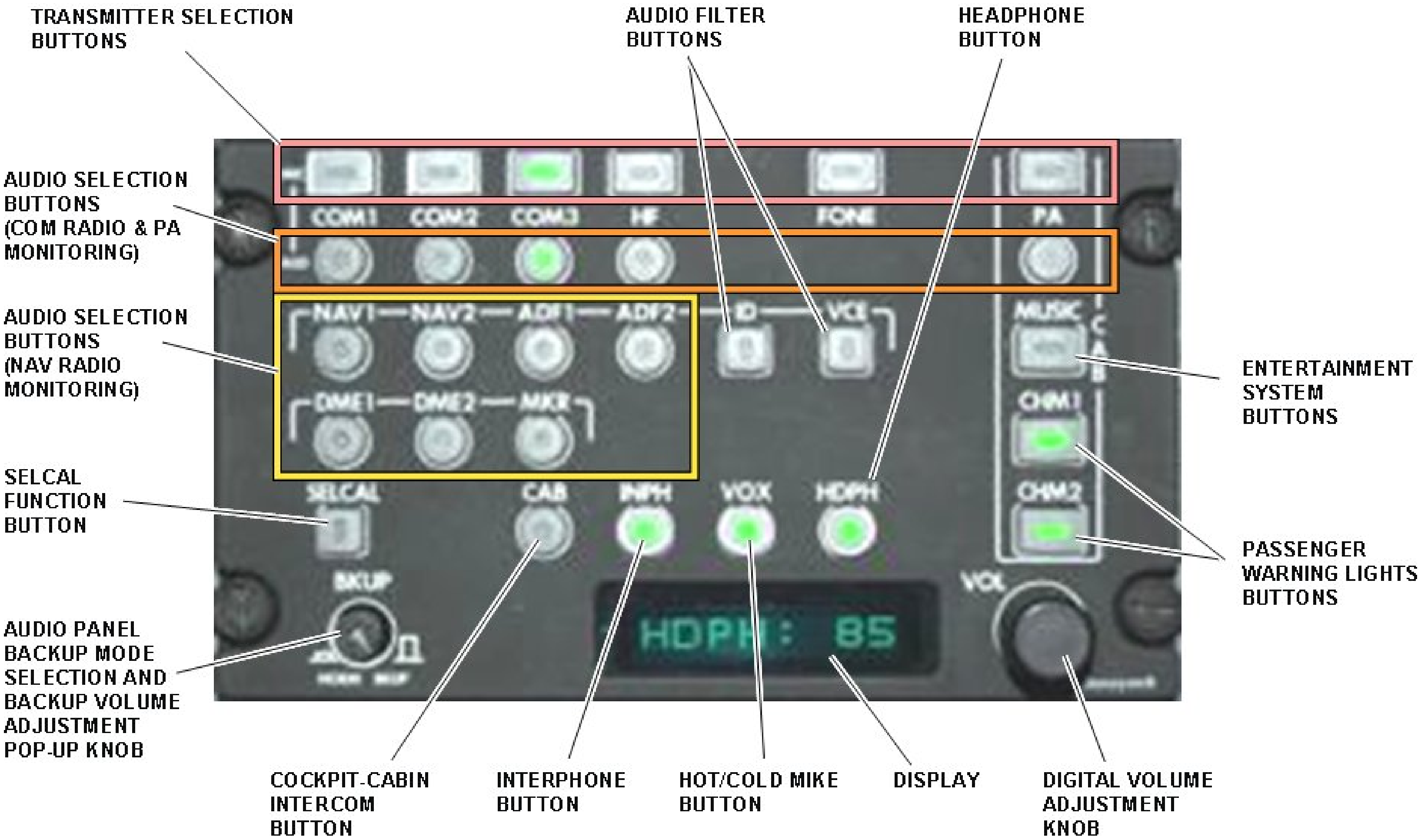
ON ………..   the headset is connected to the opposite side audio panel to give backup intercom system (ICS). Pushing the PTT button connects the pilot microphone to the on-side communications radio.

25. FREQUENCY window

………..……...   shows the selected radio and the current volume setting. It shows some messages.



AUDIO SYSTEM – CONTROLS AND INDICATORS



Radio Systems
Page 10-74

AUDIO CONTROL PANEL

CAS CAUTION MESSAGES

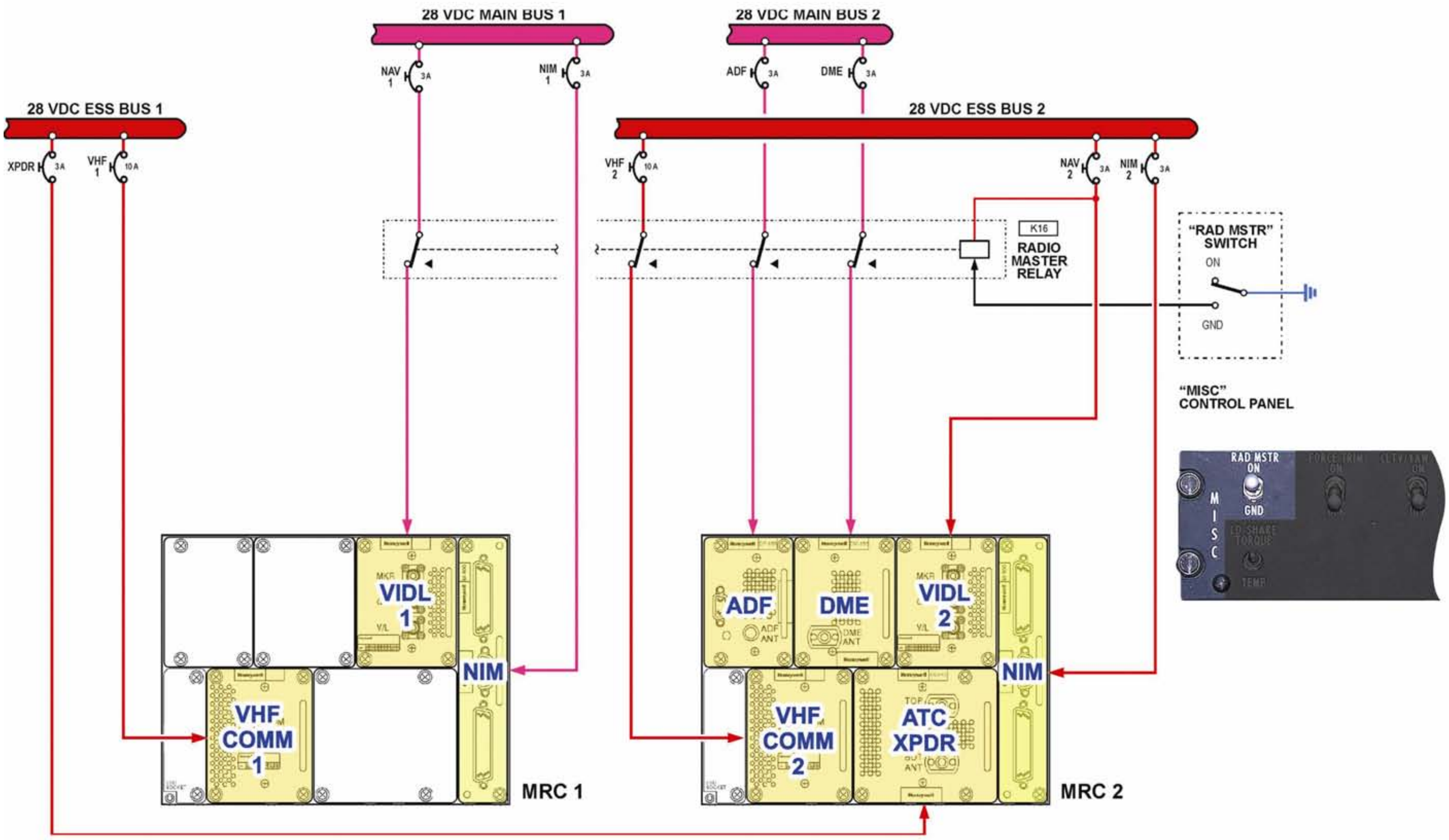
CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
1(2) VHFCOM OVHT	Associated radio transmitter overheat	VHF OVERHEAT	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES COMMUNICATION SYSTEM
1(2)(3)(4)(5)(6)(7)(8) AUDIO FAIL	Associated audio panel system failed CAUTION When Audio Panel 1/2 has been reverted to back-up mode audio tones and voice warnings cannot be heard by on side crew. NOTE Audio panel id: 1-Copilot, 2-Pilot, 3-Hoist Operator,4-Cabin Operator (if installed), 5-2nd Cabin Operator (if installed), 6-7-8-Reserved	AUDIO PANEL FAILURE	
1(2) MRC OVHT	Associated radio/nav modular system overheat	MRC OVERHEAT	

RADIO MASTER SYSTEM AND ELECTRICAL POWER

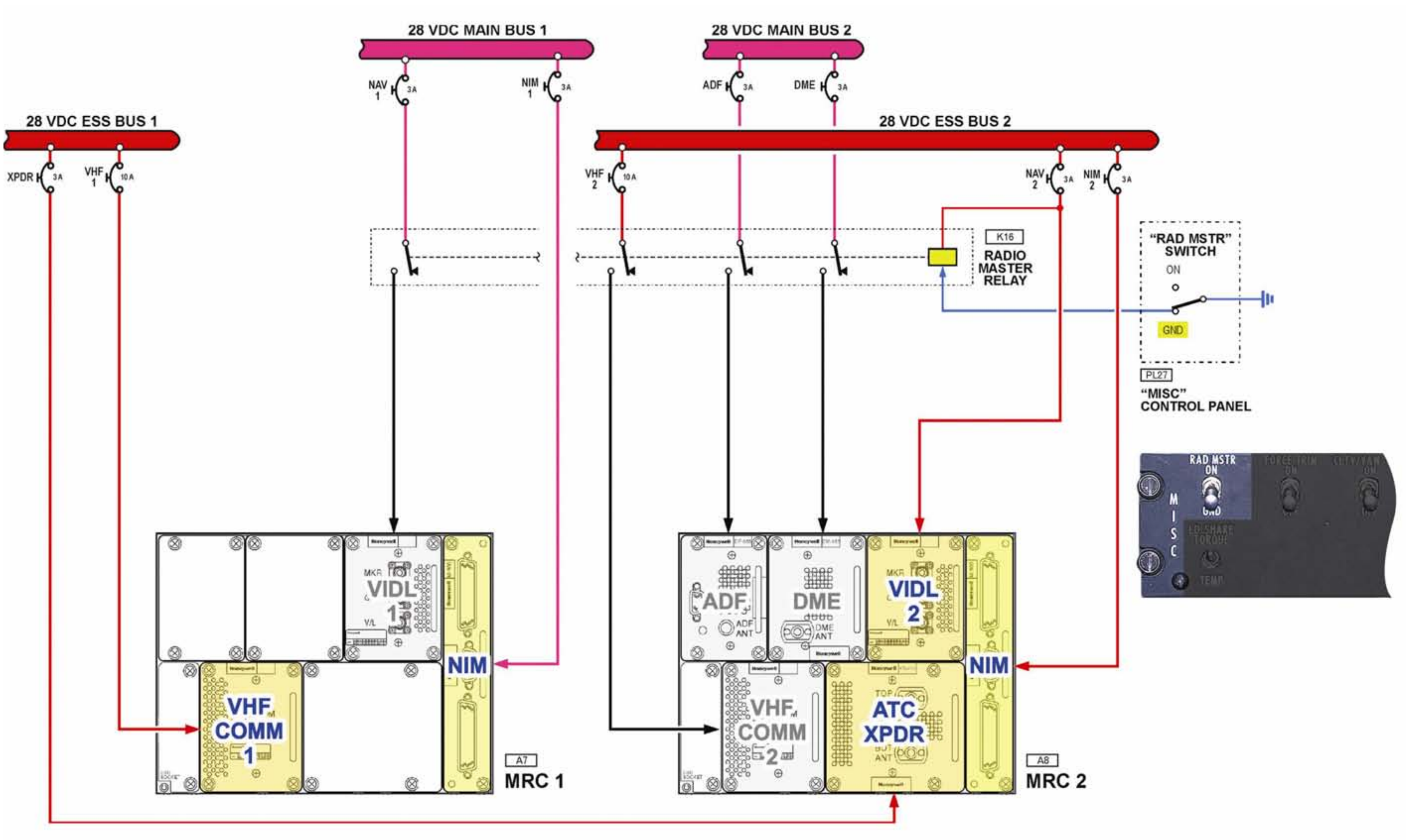
The following figures show the radio master system electrical connections. From Note that the RAD MSTR switch on the MISC panel set to ON does not give power to the radio system, but is used to remove power when set to GND.

When the helicopter is on the ground, the COMM2 antenna is close to the ground and so is not used for communications. This is the reason why the VHF COMM2 radio is not working opposite the VHF COMM1 radio.

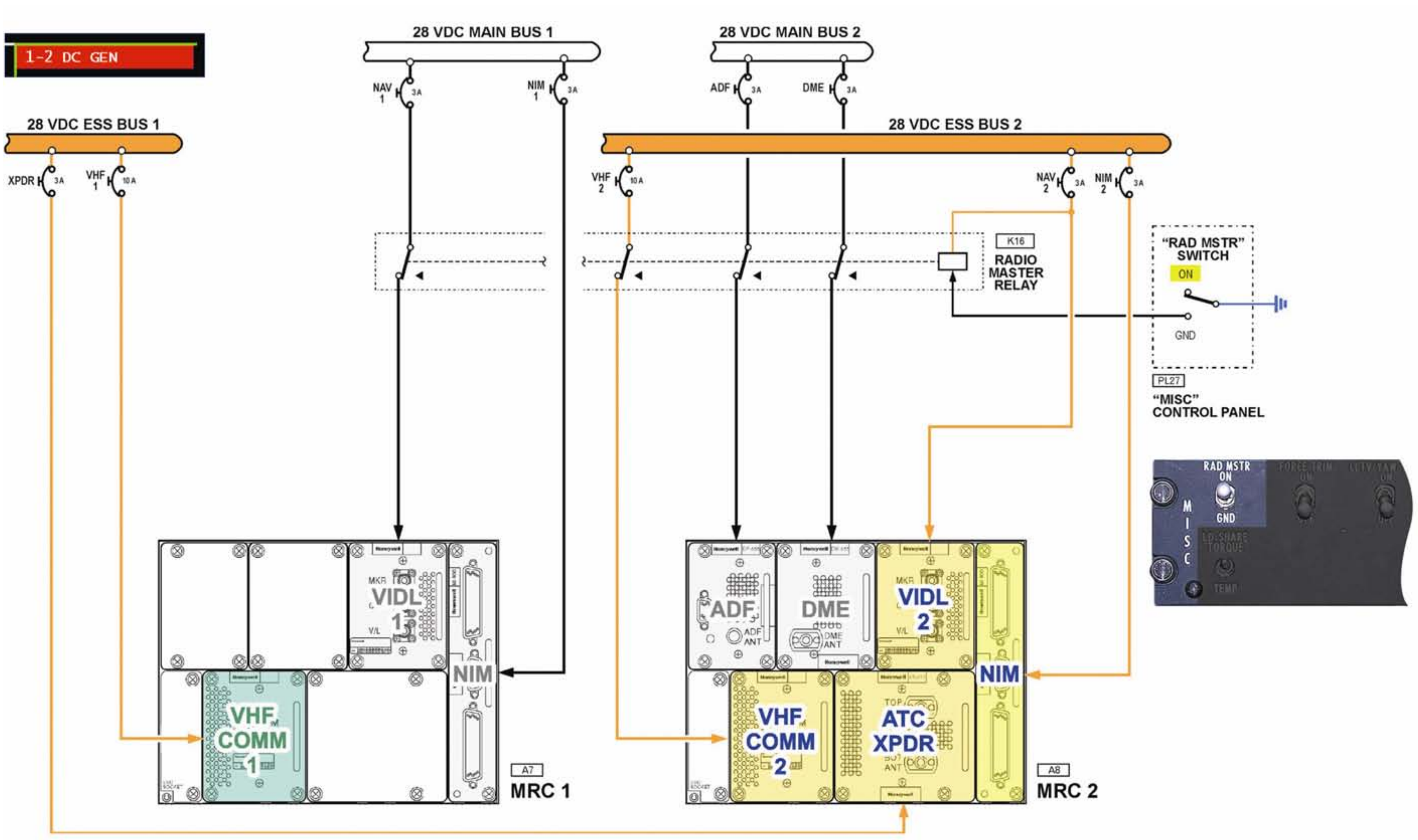
In case of dual generator failure, the VHF COMM1 radio is available only by the copilot in backup mode.



RADIO MASTER SWITCH SET TO ON



RADIO MASTER SWITCH SET TO GROUND



DUAL GENERATOR FAILURE – ESS BUS ONLY

CHAPTER 24
ELECTRICAL POWER

SECTION 00 – GENERAL

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ELECTRICAL POWER – GENERAL

The electrical power system is a 28 VDC system that has the purpose to:

- Generate and distribute DC electrical power
- Store electrical energy into batteries for:
  - In-flight emergency operations, and
  - Autonomous engine starting and ground operations
- Distribute DC electrical power provided by an external power source on the ground

The major components of the system are listed in the figure.

ELECTRICAL POWER – GENERAL ARRANGEMENT

The electrical power system consists of two similar subsystems, no.1 (LH) and no.2 (RH), each consisting of:

- A Starter-Generator and the relevant Generator Control Unit (GCU)
- A Battery (Main on LH; Auxiliary on RH)
- A set of bus bars ranked – from the most important – as:
  - Essential bus (ESS)
  - Main bus (MAIN)
  - Non-Essential bus (NON-ESS)

A Battery bus – directly powered by the Auxiliary Battery – is also available. The Battery bus is only to supply power to the on-board recorders (FDR/CVR and CMC) for proper shut-down when power is removed from the Essential busses.

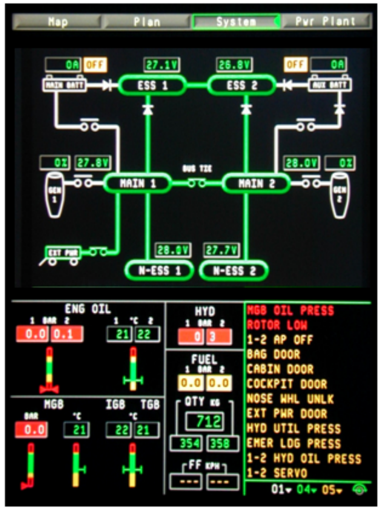
ESS BUS 1 and ESS BUS 2 are interconnected via a circuit breaker and protected by diodes so as they cannot feed any other bus bar.

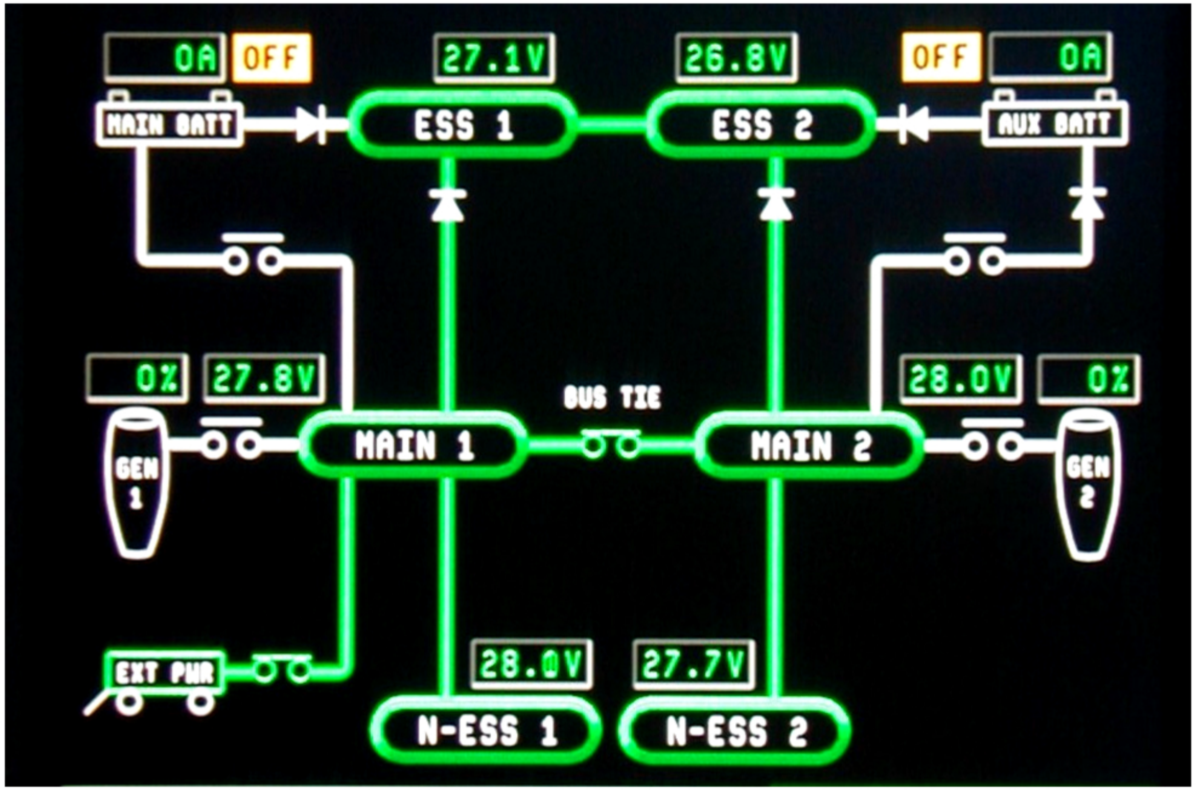
MAIN BUS 1 and MAIN BUS 2 are normally isolated; a BUS TIE contactor permits interconnection to allow transfer of power in case of generator failure or when generators are not operating.

The external power – when available – is connected to the MAIN BUS 1; however, external power is distributed to all bus bars.

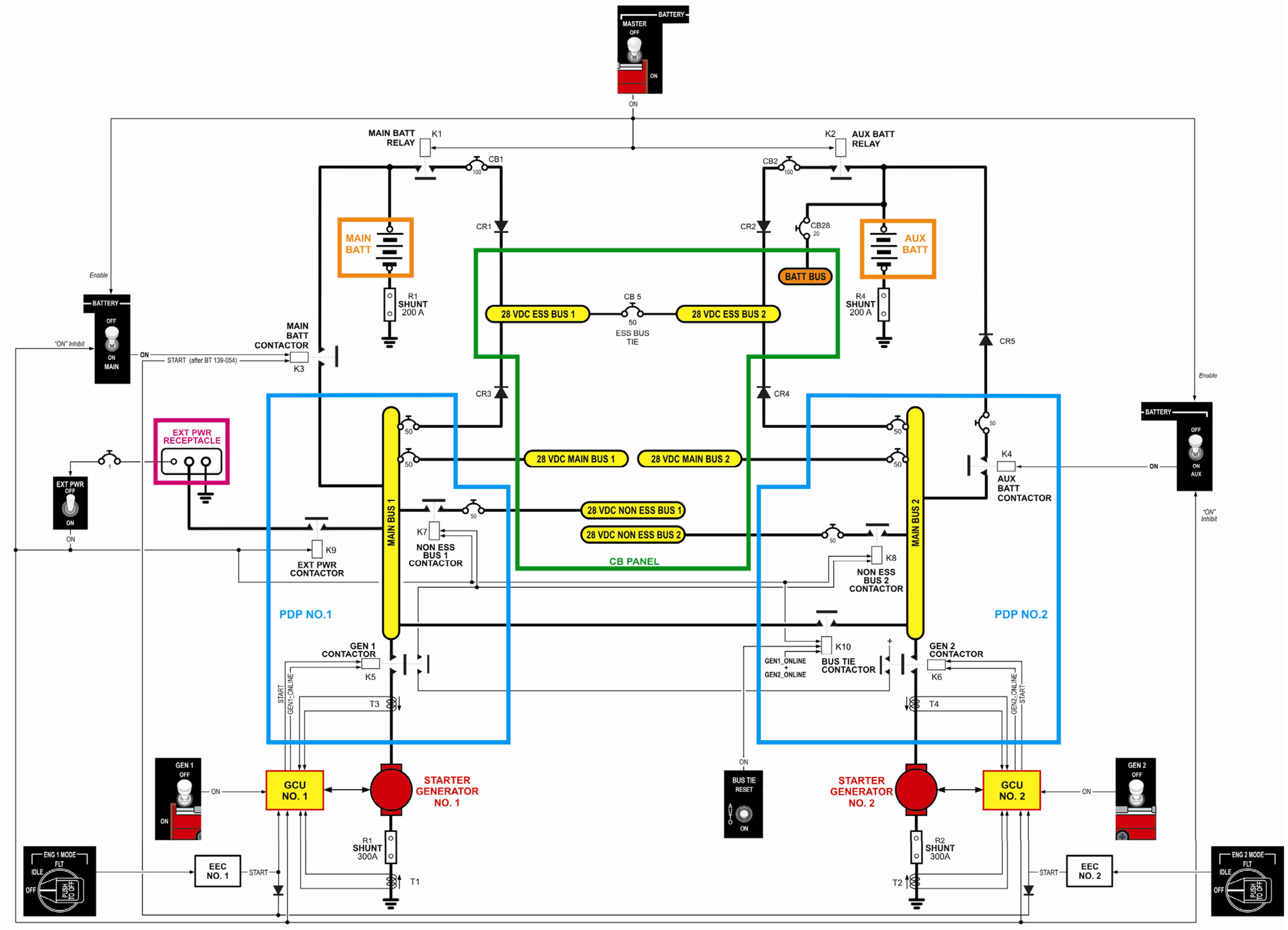
In the following pages two diagrams show the general arrangement of the electrical power system with the major components highlighted:

- the Synoptic Diagram, which can be displayed on the MFD, both on the ground and during flight
- the Simplified Diagram, which is used in this training manual to describe the system operation





ELECTRICAL POWER – MFD SYNOPTIC DIAGRAM



DC POWER – GENERAL ARRANGEMENT

ELECTRICAL POWER – MAJOR COMPONENTS

STARTER-GENERATOR

Two 9kW DC starter-generators are used to start the relevant engine and, when the engine is running, to produce DC power.

The starter-generator no. 1 is connected to the MAIN BUS 1; the starter-generator no. 2 is connected to the MAIN BUS 2.

Each starter-generator is installed on the accessory gearbox of the relevant engine, is brush-type and incorporates a speed sensor.

Each starter-generator is cooled by a mechanical fan: that implies a maximum load versus pressure altitude (as reported by the placard on the instrument panel).

GENERATOR CONTROL UNIT (GCU)

Two GCU (GCU1 and GCU2) are used to control, protect and monitor the relevant starter-generator.

- Control functions include voltage regulation (generator mode), field weakening (starter mode) and line contactor control
- Protection functions include overvoltage, undervoltage, reverse current, overcurrent (short circuit)
- Monitoring functions include starter-generator speed and external power presence

The GCU's are installed in the aft Avionic bays.

Each GCU also terminates the Manual start cycle of the relevant engine when the starter speed has exceeded the equivalent of 50% Ng.

BATTERIES

Two Nickel-Cadmium batteries are used to store electrical energy:

- Main Battery (40 Ah, standard; 44 Ah, optional)
- Auxiliary Battery (13 Ah, standard; 27 Ah or 28 Ah, optional)

The MAIN Battery is used to:

- Feed the starter during engine starting
- Feed the Essential and Main loads during in-flight emergency (dual generator failure) or on the ground (engines not running and external power not available)

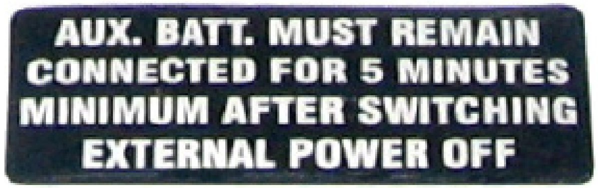
The Auxiliary (AUX) Battery is used to:

- Feed the Essential loads during engine starting, in-flight emergency (dual generator failure) or on the ground (engines not running and external power not available)
- Feed the Battery bus

Each battery is provided with a temperature sensor used to trigger a caution message in the CAS window when the battery overheats.

The batteries cannot be recharged by the external power: they are automatically disconnected when external power is applied.

The AUX battery must not be immediately unplugged after switching aircraft power off, in order to allow the proper shutdown of the on-board recorders (FDR/CVR and CMC) via the Battery bus, as stated by the placard in the nose compartment.



EXTERNAL POWER RECEPTACLE

A standard 28 VDC receptacle is provided in the lower RH side of the nose to permit connection of an external power source.

The external power is used to:

- supply power to all DC bus bars
- Feed the starter during engine starting

The receptacle is protected by a door; a microswitch detects if the external power door is not latched closed and triggers the EXT PWR DOOR caution message in the CAS window.

POWER DISTRIBUTION PANEL (PDP)

Two PDP (PDP1 and PDP2) are installed in the forward area of the cabin roof.

PDP's are cabinets that contain the power contactors and the control circuits used to:

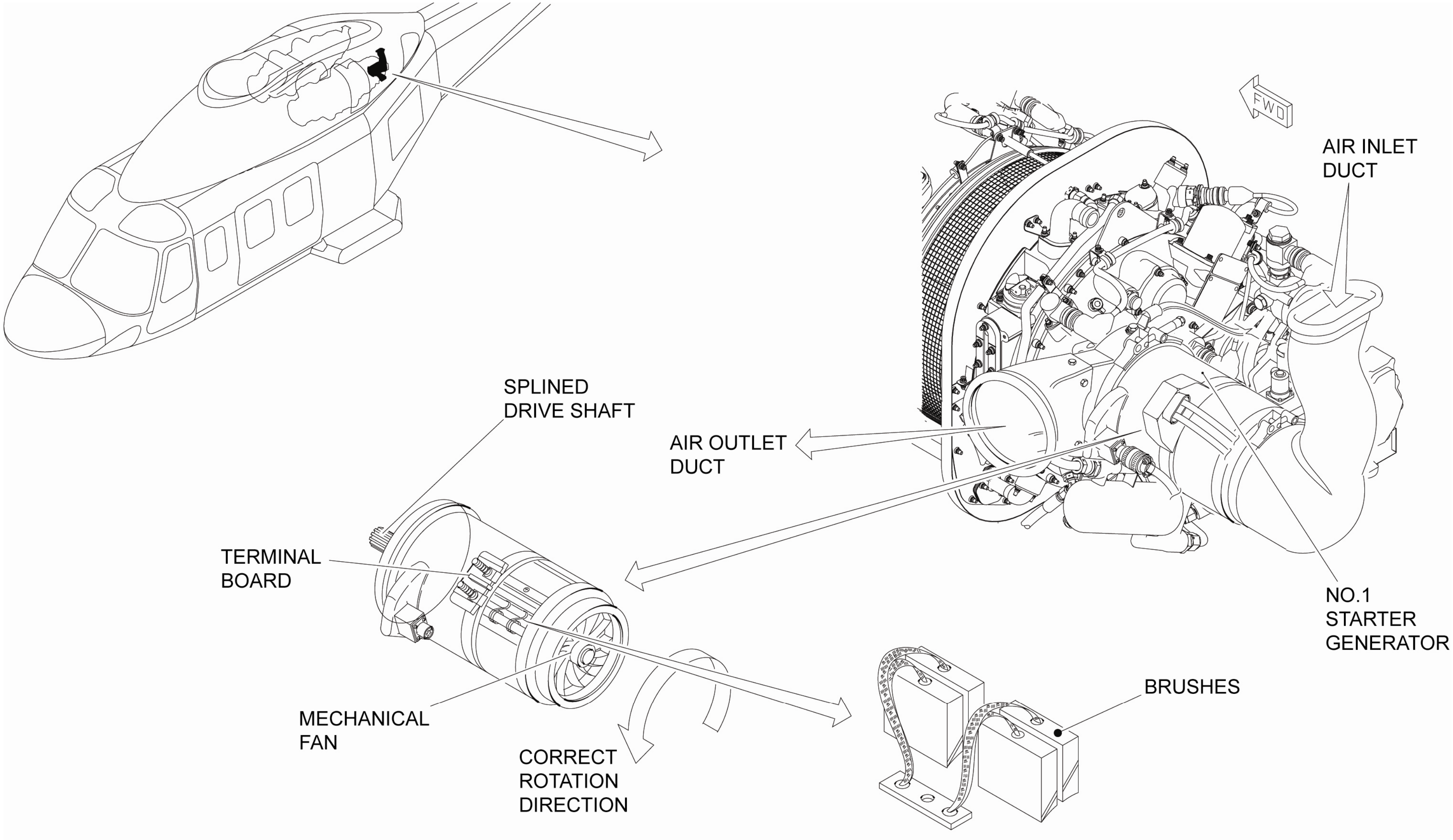
- connect the aircraft electrical power sources (generators and batteries) to the LH and RH distribution bus bars via the MAIN LOAD BUS 1 and MAIN LOAD BUS 2, respectively
- check and connect the external power source to the aircraft distribution bus bars
- control the Main bus tying (BUS TIE Contactor)

SHUNT

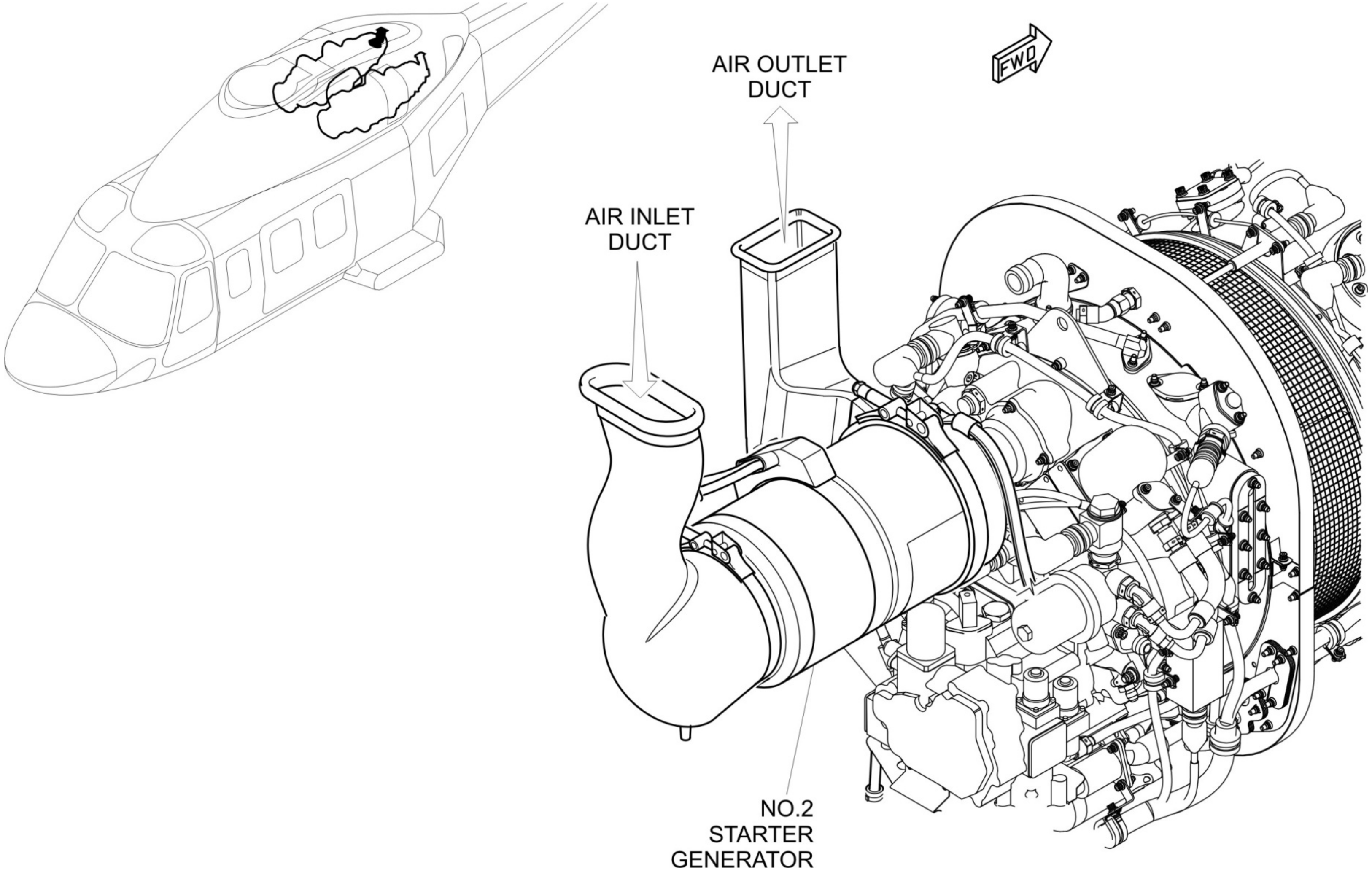
Four shunts are used to pick up a signal proportional to the current flowing to/from electrical power sources (main and auxiliary batteries, starter-generators 1 and 2) to display the relevant electrical load on indicators.

DC CURRENT TRANSFORMER (CT)

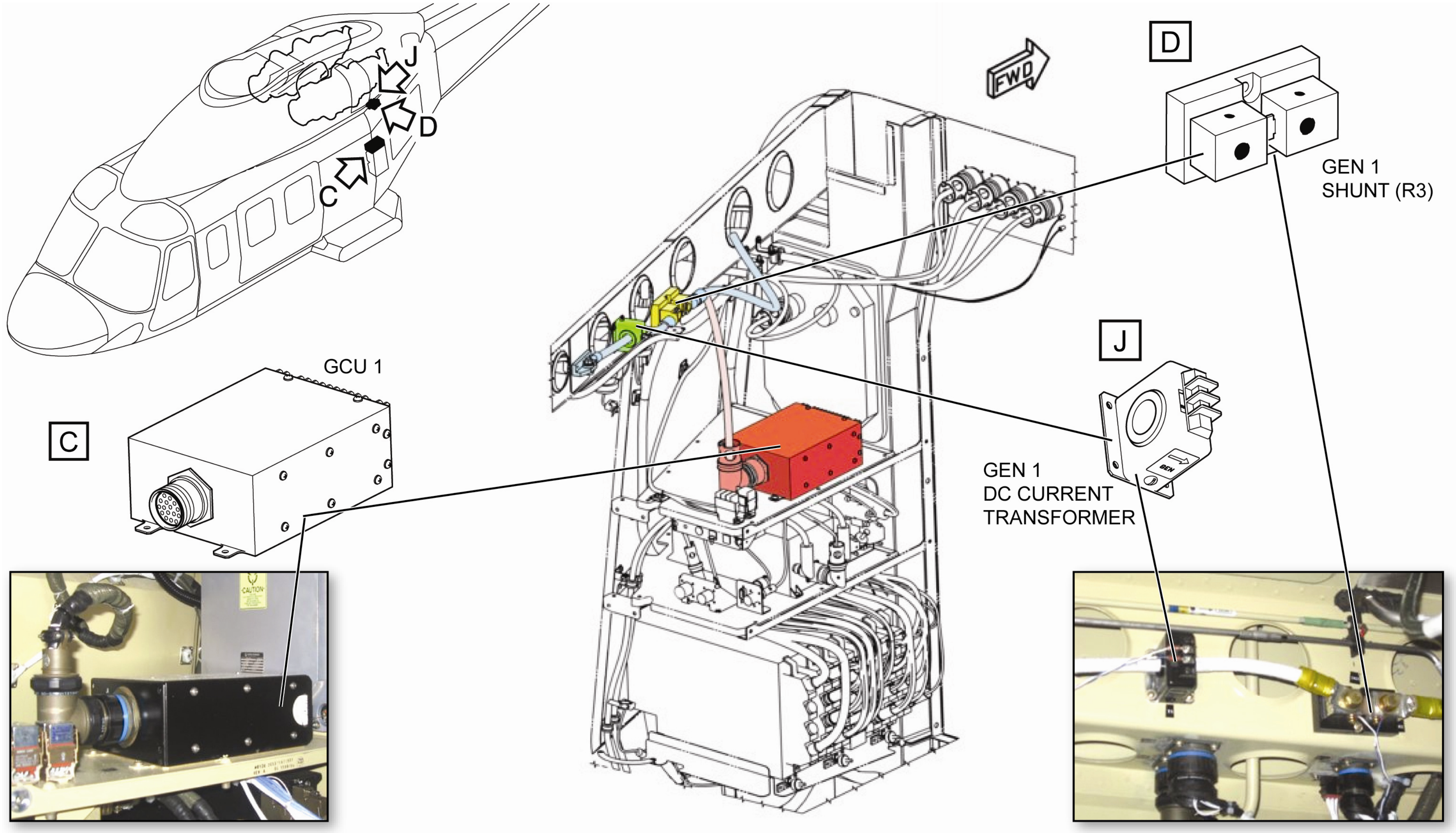
Four current transformers are used to detect the differential changes in DC current feeding lines, routing a signal to the GCU which de-energizes the line contactor when an overcurrent or a short-circuit is detected.



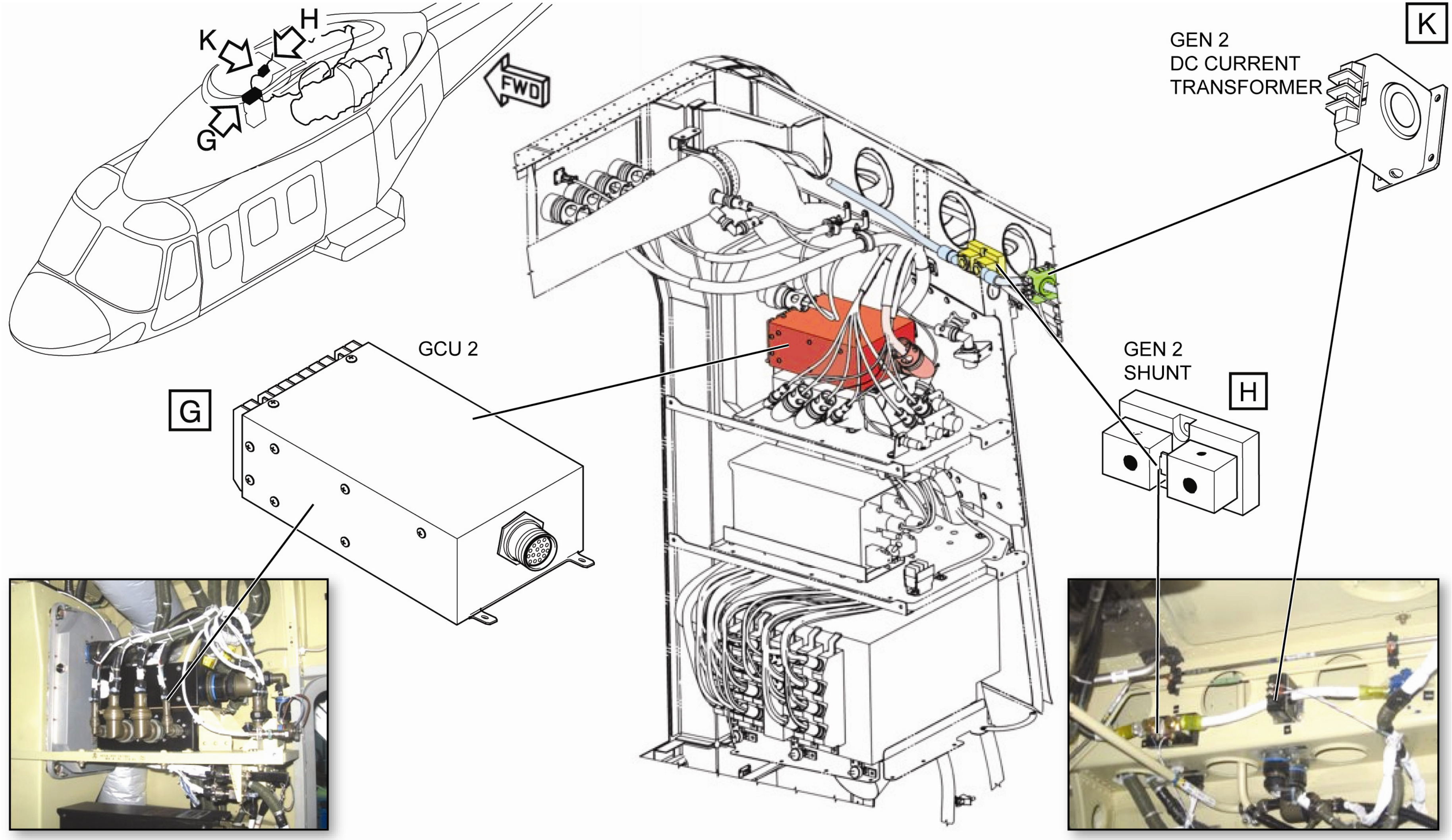
DC STARTER / GENERATOR NO.1



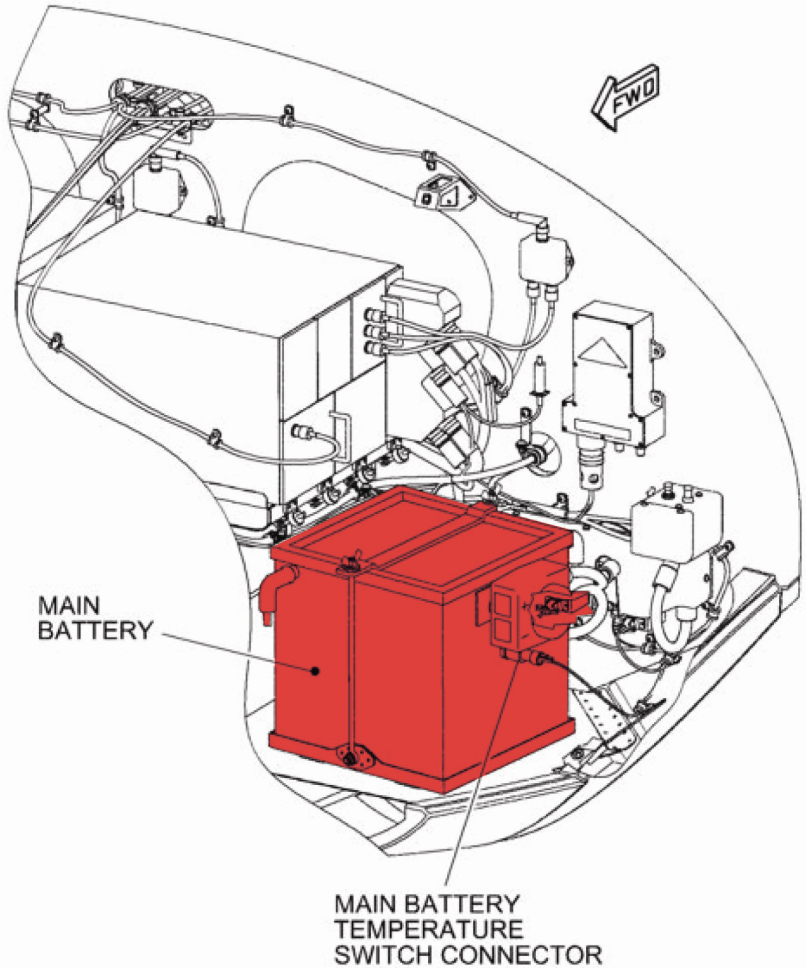
DC STARTER / GENERATOR NO.2



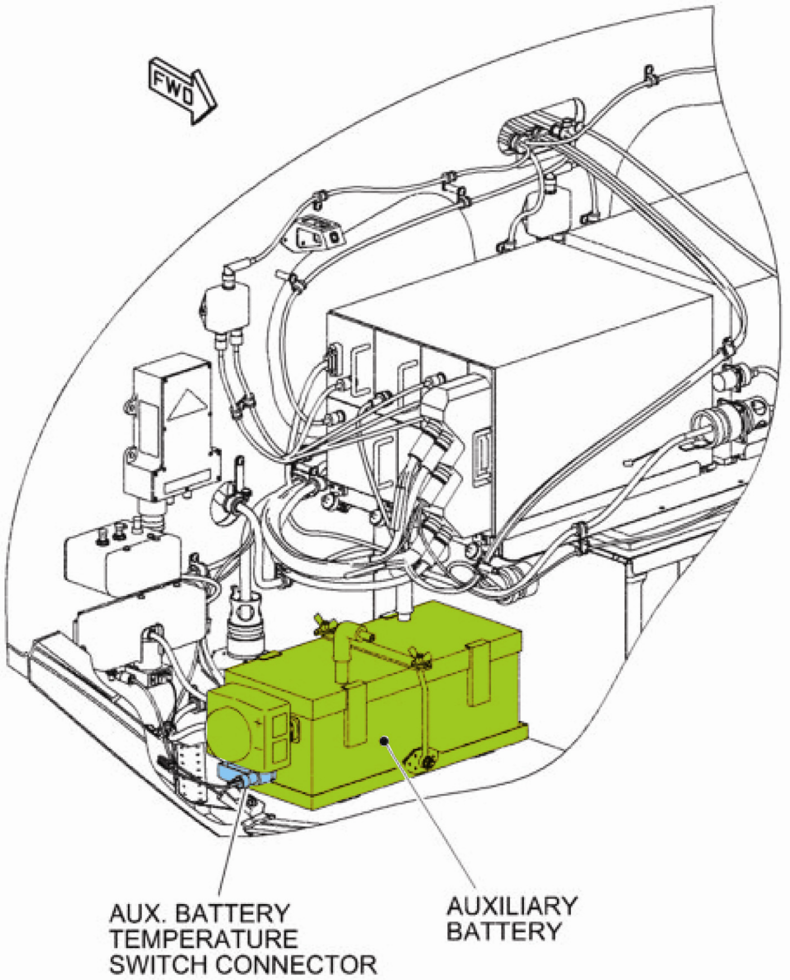
REAR AVIONIC BAYS – LH AVIONICS RACK



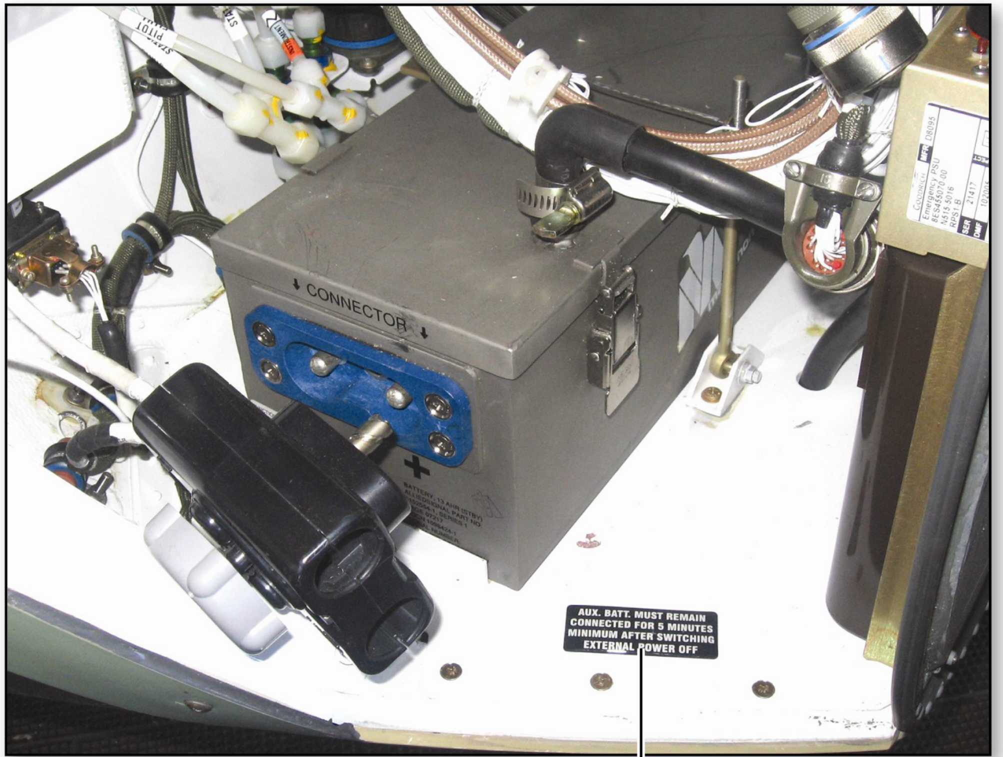
REAR AVIONIC BAYS – RH AVIONICS RACK

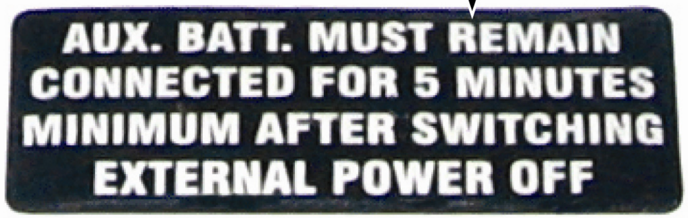


NOSE COMPARTMENT – LH SIDE

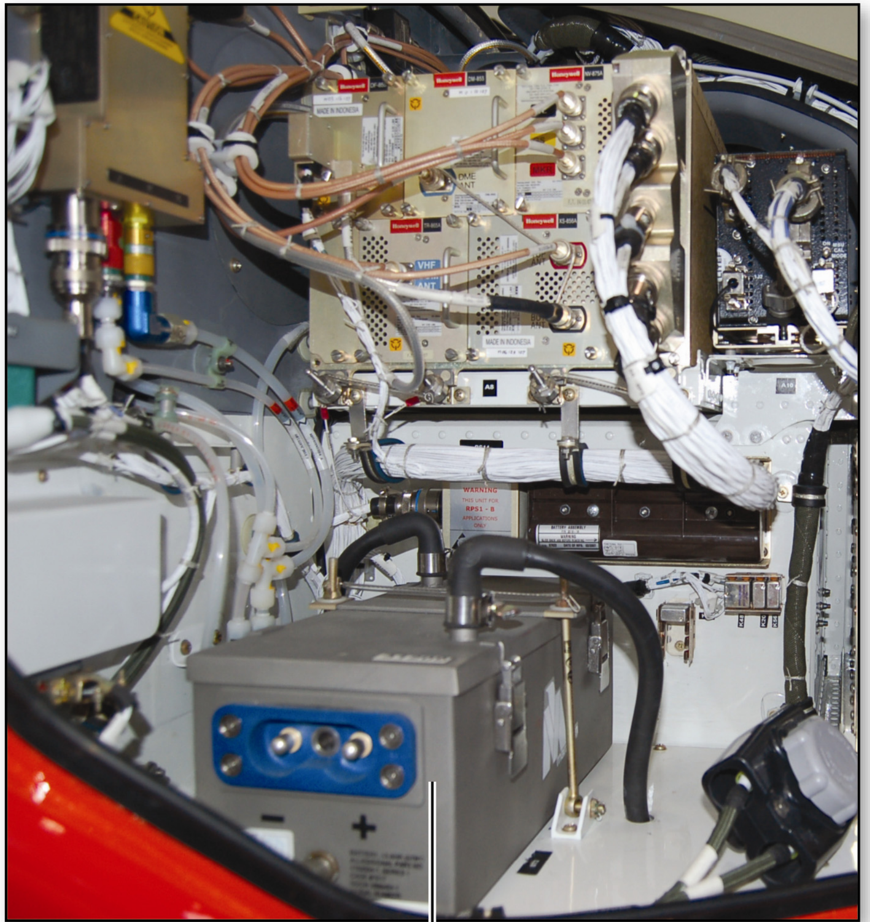


NOSE COMPARTMENT – RH SIDE





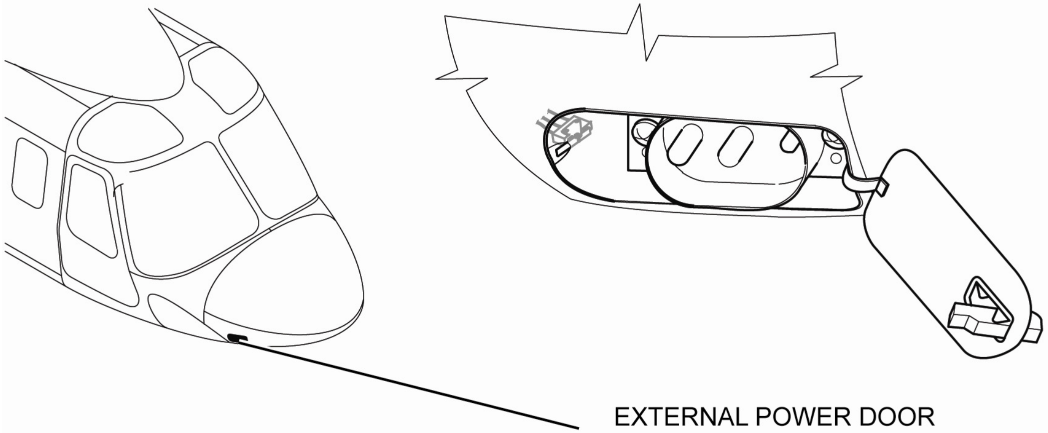
AUXILIARY BATTERY – PLACARD



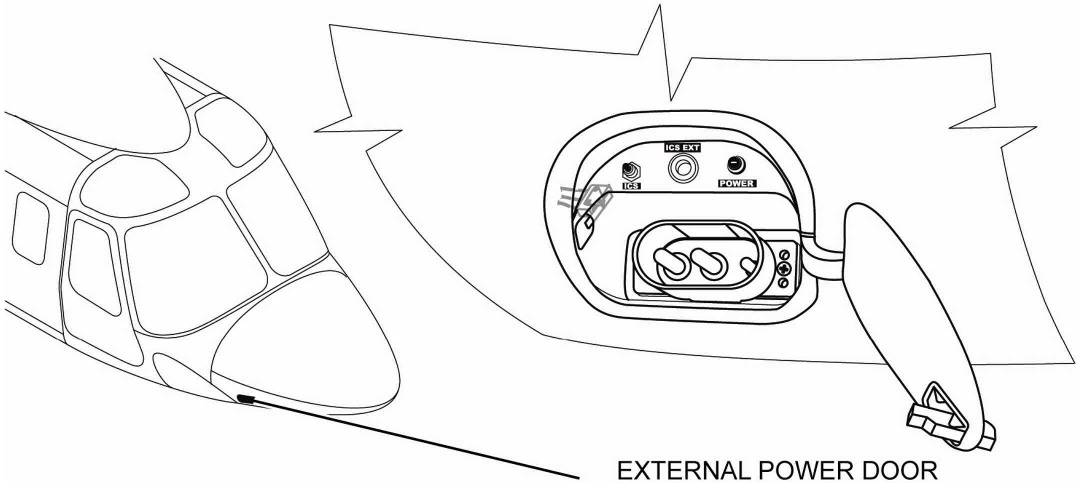
AUXILIARY BATTERY (RH side)

MAIN BATTERY (LH side)

MAIN BATTERY AND AUXILIARY BATTERY – LONG NOSE CONFIGURATION

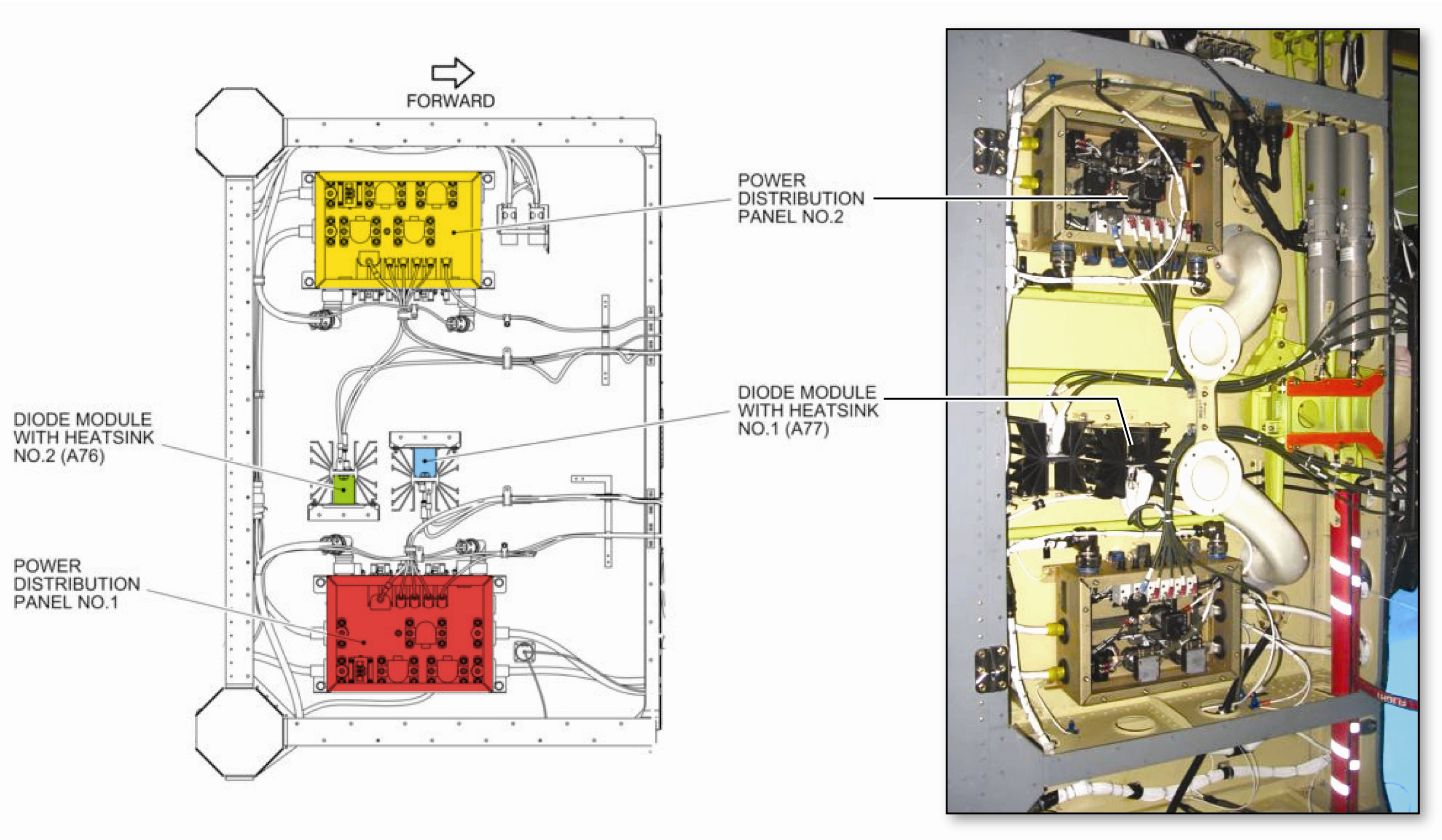


EXTERNAL POWER DOOR
SHORT NOSE CONFIGURATION



EXTERNAL POWER DOOR
LONG NOSE CONFIGURATION

DC EXTERNAL POWER RECEPTACLE


FORWARD
POWER DISTRIBUTION PANEL NO.2
DIODE MODULE WITH HEATSINK NO.1 (A77)
DIODE MODULE WITH HEATSINK NO.2 (A76)
POWER DISTRIBUTION PANEL NO.1

CABIN ROOF

CIRCUIT BREAKER (CB) PANELS

A circuit breaker panel is installed in the overhead console: it houses all the circuit breakers available in the cockpit for standard and most common optional systems connected to the aircraft distribution bus bars.

An optional additional circuit breaker panel is installed when dedicated optional systems are installed.

CIRCUIT BREAKER (CB) PANEL ARRANGEMENT

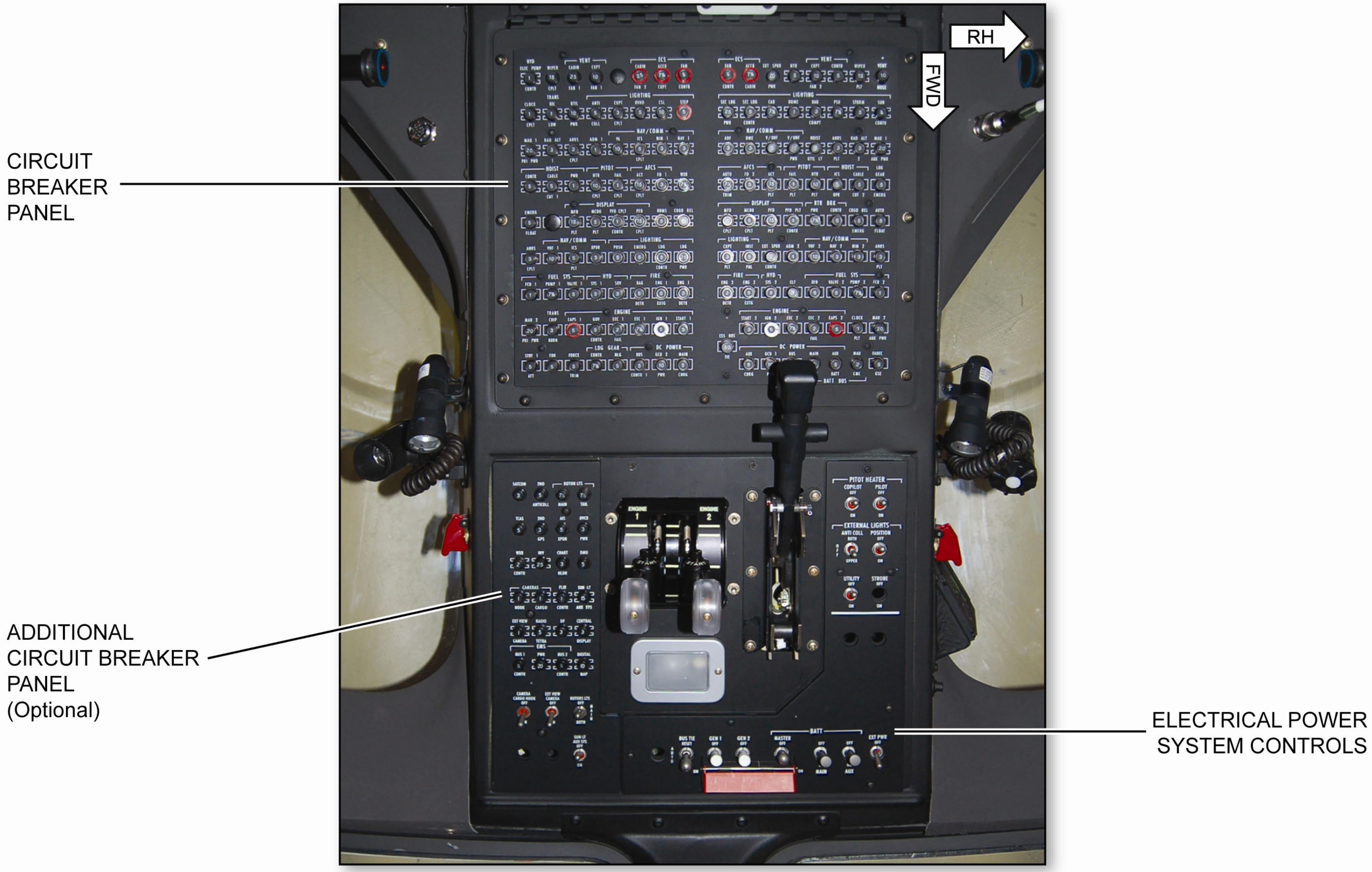
The CB panel is divided in two groups of circuit breakers:

- LH group includes CB connected to Essential, Main and Non-Essential bus bars no.1
- RH group includes the CB connected to Essential, Main and Non-Essential bus bars no.2 and Battery Bus

The ESS BUS TIE circuit breaker — rated 50 A — keeps ESS BUS 1 and ESS BUS 2 interconnected.

ELECTRICAL SYSTEM CONTROL PANEL

All the switches that control the electrical power system are installed in the forward area of the overhead console.



OVERHEAD CONSOLE

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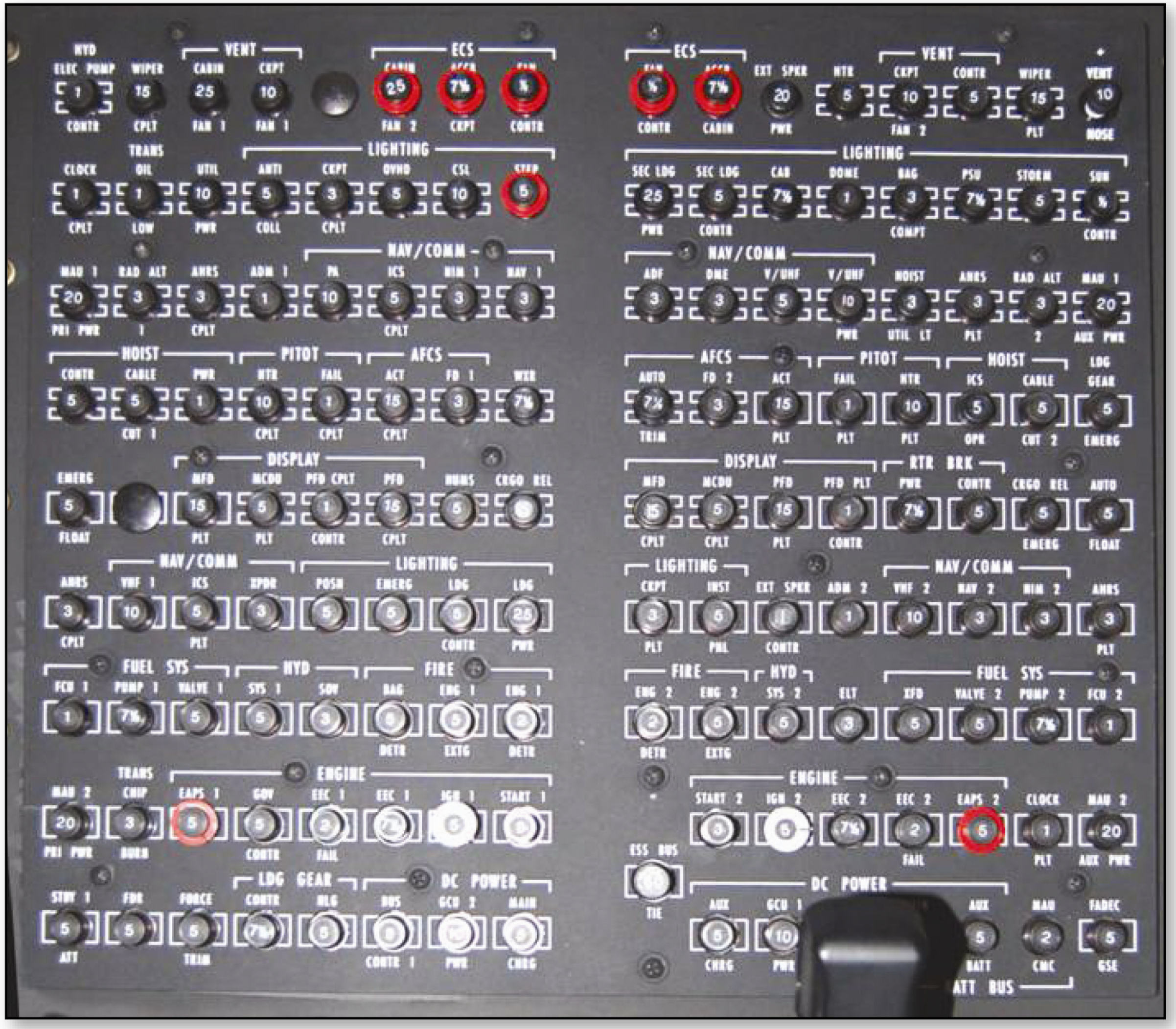
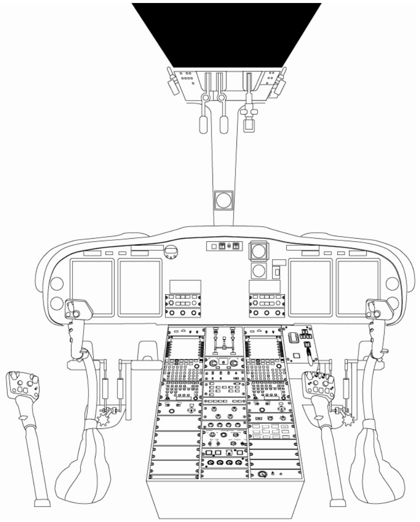




ESS BUS

MAIN BUS

NON-ESS BUS

CIRCUIT BREAKER PANEL

ELECTRICAL POWER – CONTROLS

1. BUS TIE switch

   ON ……..……. BUS TIE contactor is forced to close: MAIN BUS 1 and MAIN BUS 2 are interconnected

   AUTO …….. (normal position) BUS TIE contactor operates automatically according to the control logic

   RESET ……… (spring-loaded momentary position) resets the latching protection logic that inhibits BUS TIE contactor operation when overcurrent is detected by the GCU

2. GEN 1 switch

   OFF ……..……. disables or shuts down Generator 1 and resets latched functions in the GCU 1

   ON ....……….. requests GCU 1 to control Generator 1 to be on-line

3. GEN 2 switch

   OFF ……..……. disables or shuts down the Generator 2 and resets latched functions in the GCU 2

   ON ..……….. requests GCU 2 to control Generator 2 to be on-line

4. BATTERY MASTER switch

   OFF ……..……. disconnects batteries (MAIN and AUX) from all bus bars

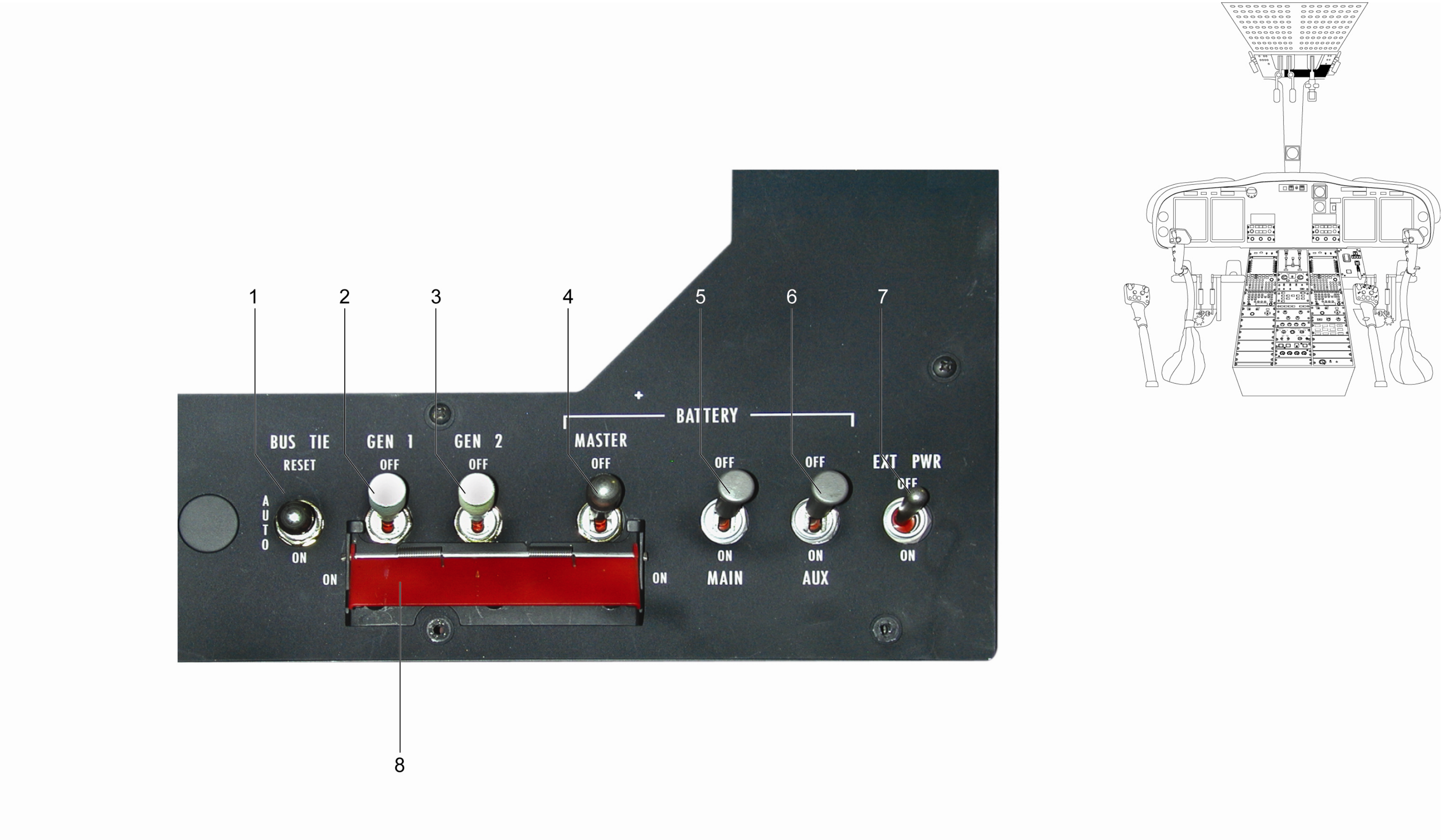
   ON …………… - connects MAIN and AUX batteries to ESS buses

   - enables the BATTERY MAIN and BATTERY AUX switches

5. BATTERY MAIN switch

   OFF ..……….. disconnects the MAIN battery from MAIN BUS 1

   ON ..……..….. if BATTERY MASTER switch is at ON and not otherwise inhibited, connects the MAIN battery to the MAIN BUS 1



ELECTRICAL POWER CONTROL

6. BATTERY AUX switch

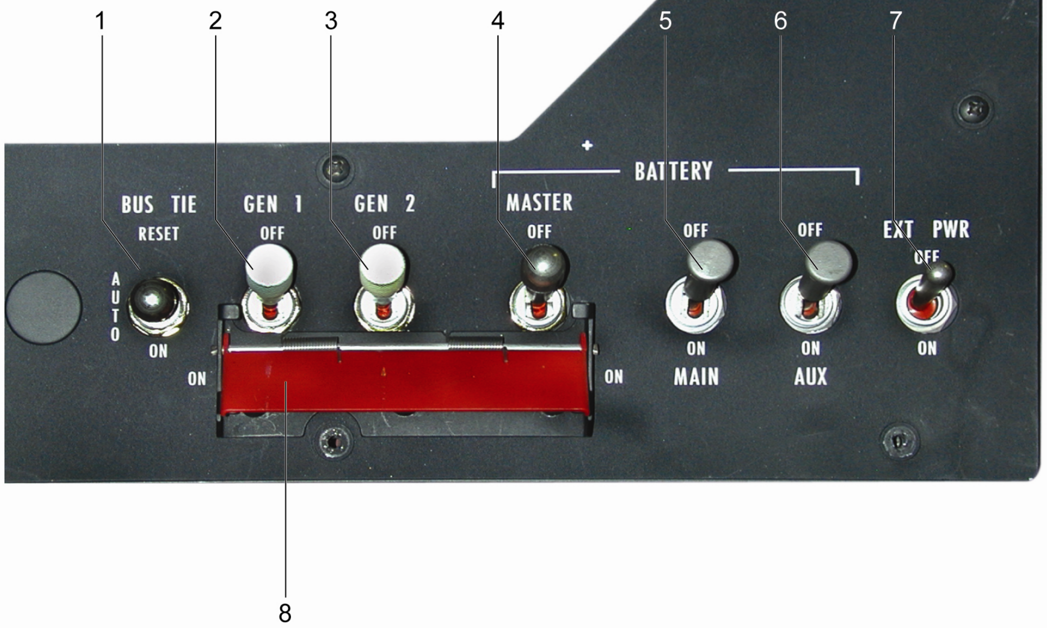
OFF .…….…...	disconnects the AUX battery from MAIN BUS 2
ON ..…….…....	if BATTERY MASTER switch is at ON and not otherwise inhibited, connects the AUX battery to the MAIN BUS 2

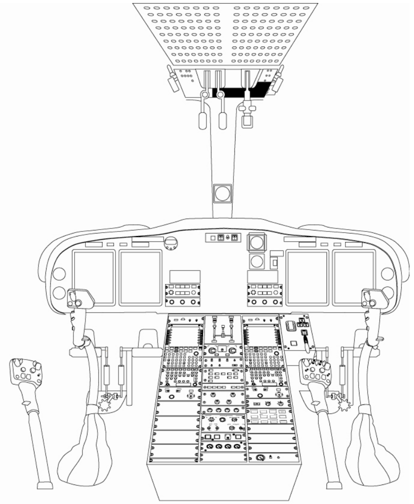
7. EXT PWR switch (allows the pilot to control the application of external power source)

OFF .…….…...	disconnects the external power from aircraft distribution bus bars
ON ...…….…..	connects the external power to the aircraft distribution bus bars

8. Red gang bar

Moved backward	cuts off all aircraft electrical power sources at the same time (GEN 1, GEN 2 and BATTERY MASTER switches are moved to OFF all together)





ELECTRICAL POWER CONTROLS – OVERHEAD SWITCH PANEL

ELECTRICAL POWER – INDICATORS

1. VOLTMETERS

Digital readout of the voltage on MAIN BUS 1, MAIN BUS 2, ESS BUS 1 and ESS BUS 2

When voltage is below 22.0 V values are displayed in amber reverse video; if it occurs during flight following a dual generator failure it means that the battery is about to discharge completely and a total black-out of electrical power may be expected soon

When voltage is below 15.0 V values are set to 0.0

NOTE. ESS BUS 1 and 2 voltage is normally less than MAIN BUS 1 and 2 voltage by about 0.8 VDC because of the drop of voltage across the diodes that protect the ESS Busses.

2. BATTERY AMPEREMETERS

Green band of the analogue vertical scale of the MAIN and AUX battery amperemeters represents a battery charge condition and is associated to positive digital readout values in Amperes.

Amber band represents a battery discharge condition and is associated to amber reverse video negative digital readout values in Amperes.

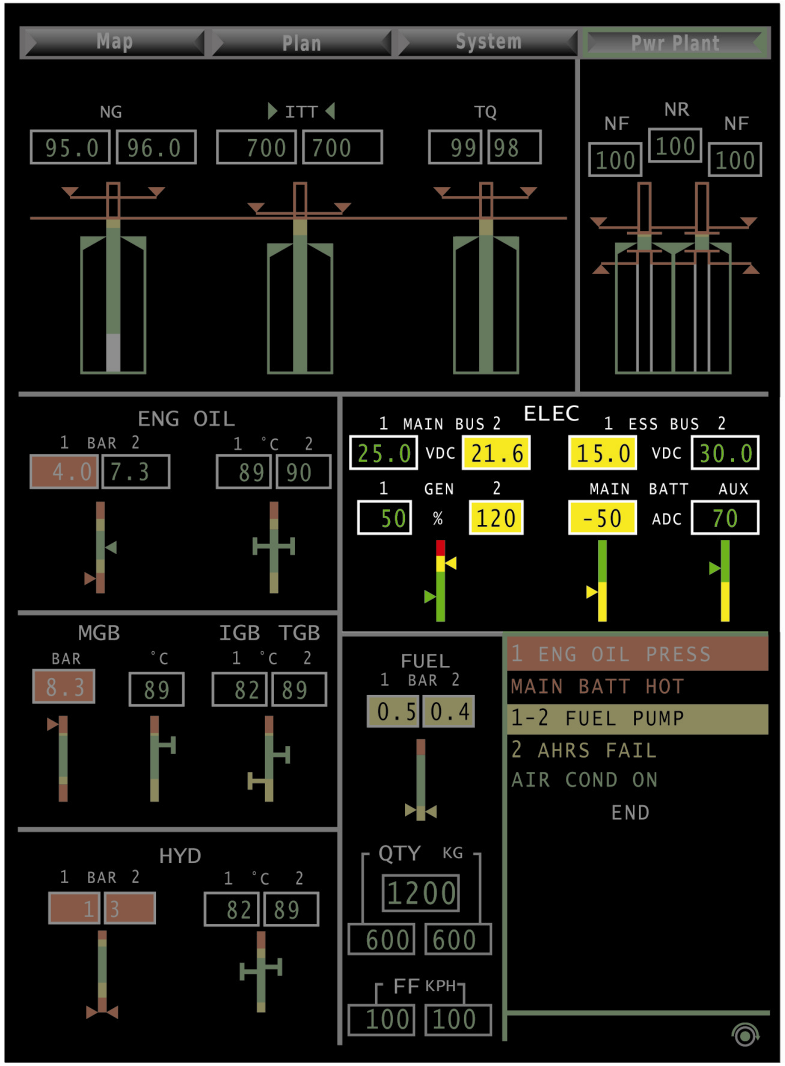
3. GENERATOR LOADMETERS

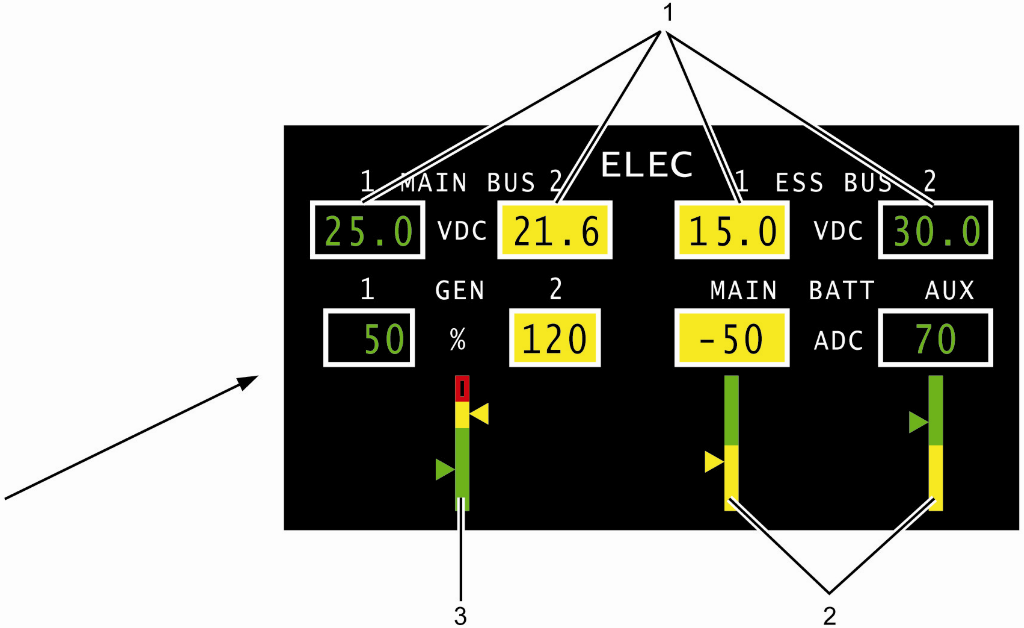
The generator load is displayed in % of maximum continuous generator output power (100% ≡ 300 A).

Overload in the amber cautionary range is only admitted for short time periods since a generator over temperature may be expected.

Overload in the red warning range must be avoided.

NOTE. When starting the second engine on batteries (Assisted Start) the generator that is already on-line is allowed to operate in the red range overload condition.





Maximum load for each generator vs. pressure altitude

ft Hp	below 15000	16000	17000	18000	19000	20000
GEN LOAD %	**100**	**87**	**74**	**60**	**47**	**33**

Note. This placard is located on the instrument panel

ELECTRICAL POWER – INDICATORS

ELECTRICAL POWER – SYNOPTIC DIAGRAM

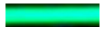
On each Multi-Function Display (MFD) a synoptic diagram representing the status of the DC power system operation can be displayed via the “System / DC Electrical” menu.

The DC Electrical synoptic diagram shows the status of:

- MAIN Battery and AUX Battery connection status, current flow (amperemeters)
- GEN 1 and GEN 2 status, connection to MAIN buses and electrical load (loadmeter in % of max continuous power)
- ESSENTIAL, MAIN and NON-ESSENTIAL busses status and voltage (voltmeters)
- BUS TIE contactor status
- EXTERNAL POWER status

CAUTION: In case of MAU 1(2) failure, do not use synoptic pages information.

DESCRIPTION OF SYMBOLS

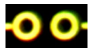

green lines stand for active connections


white lines stand for non-active connections


yellow dashed lines stand for undetermined connections


stands for a CLOSED contactor

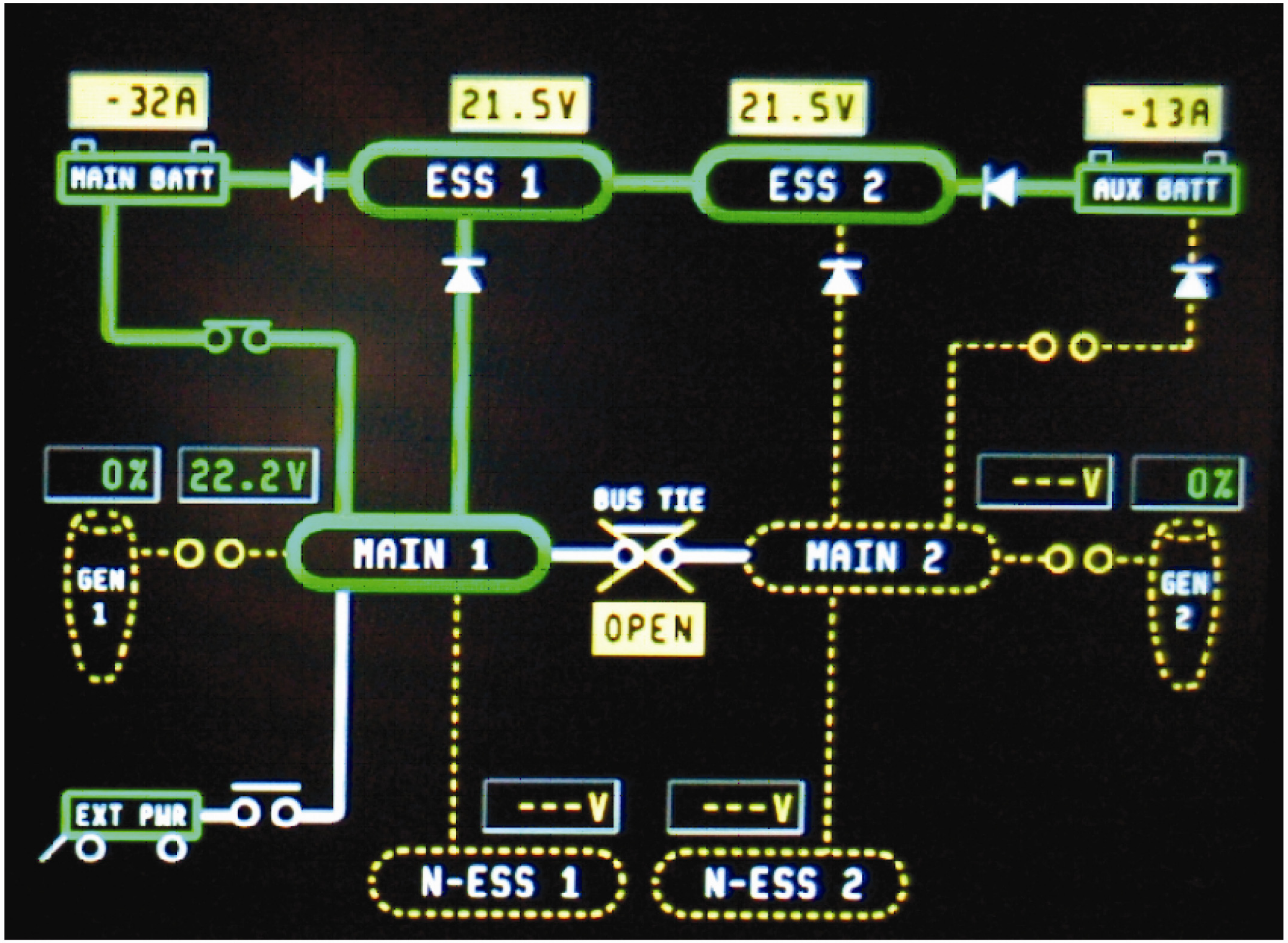

stands for an OPEN contactor

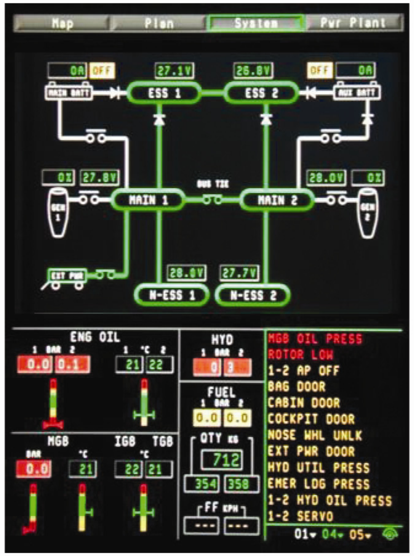

stands for an undetermined contactor


stands for a diode

NOTE Battery amperemeters show yellow reverse-video negative values when battery is discharging.

Voltmeters show yellow reverse-video values when voltage is less than 22 VDC.





MFD - DC ELECTRICAL SYSTEM SYNOPTIC DIAGRAM

ELECTRICAL POWER SYNOPTIC DIAGRAM

ELECTRICAL POWER – SIMPLIFIED SCHEMATIC DIAGRAM

A simplified schematic diagram is used to describe the operation of the electrical power as follows:

- Normal procedures:
  - Engine starting on batteries (BATTERY Starting)
  - Engine starting on external power (EXT PWR Starting)
  - In-flight normal operation
- Emergency procedures:
  - Single Generator Failure
  - Single DC Bus Failure
  - Dual Generator Failure

BATTERY STARTING

The following diagrams represent the sequence of actions for a normal battery starting of the engines (steps 1 to 8 on the simplified schematic diagram). In this example engine no. 2 is started first.

Refer to AW139-RFM-4D – Section 2 for Normal Procedures.

STEP 1

Helicopter is parked and ready for flight. All switches are in the safe position.

STEPS 2 TO 3

When BATTERY MASTER switch is set to ON, Main and Aux Batteries are connected to ESS BUS 1 and ESS BUS 2 thus providing power to the rotorcraft essential loads.

When BATTERY MAIN switch is set to ON, the Main Battery is connected to MAIN BUS 1 (Main Battery contactor K3 closes) which is then also powered.

When BATTERY AUX switch is set to ON, the Auxiliary Battery is connected to MAIN BUS 2 (Aux Battery contactor K4 closes), but MAIN BUS 2 is not powered because of the reverse biased diode (CR5) which only permits recharging of the Auxiliary battery from MAIN BUS 2.

When GEN 1 and GEN 2 switches are set to ON they give an input to the relevant GCU so that the GCU will put the relevant generator on-line as soon as conditions permit.

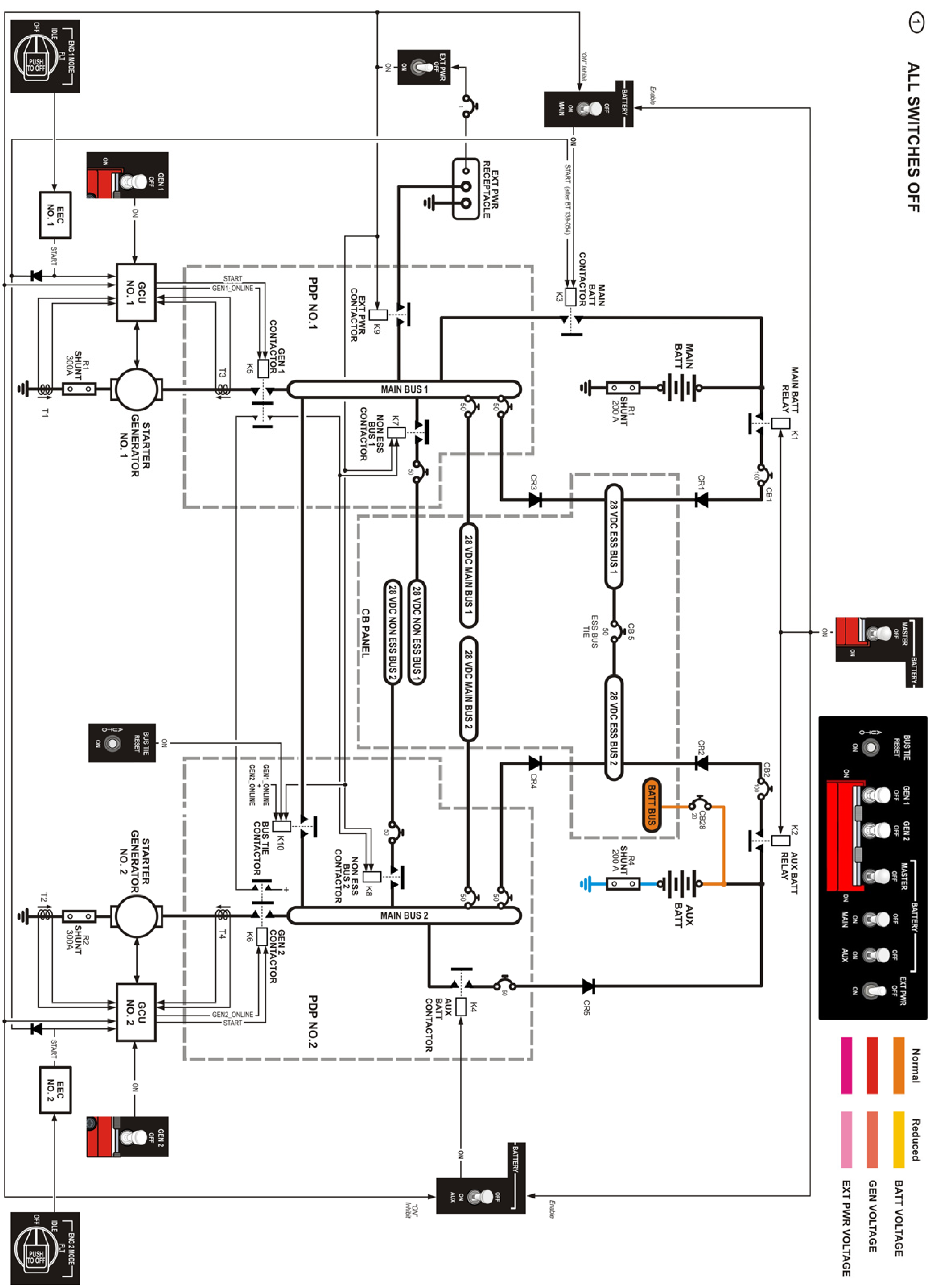
23 VOLT CHECK

Before attempting starting the engine on batteries, the pilot has to check that the involved MAIN BUS voltage is not less than 23 V.

STEP 4

As in this example the engine no 2 is started first, the BUS TIE switch must be set to ON to power MAIN BUS 2.

BUS TIE Contactor closes thus connecting MAIN BUS 1 and MAIN BUS 2.



STEP 1 – ALL SWITCHES OFF

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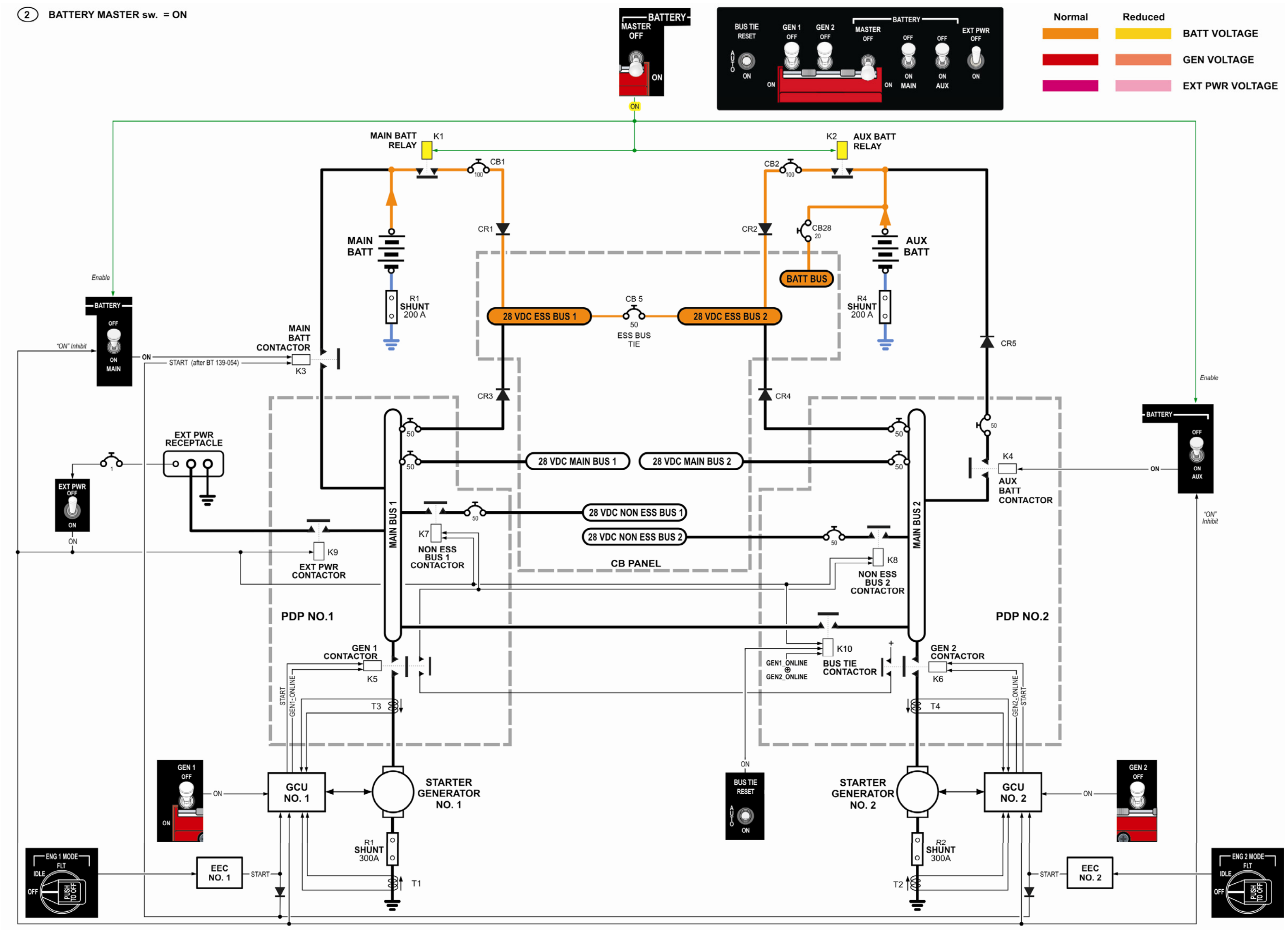
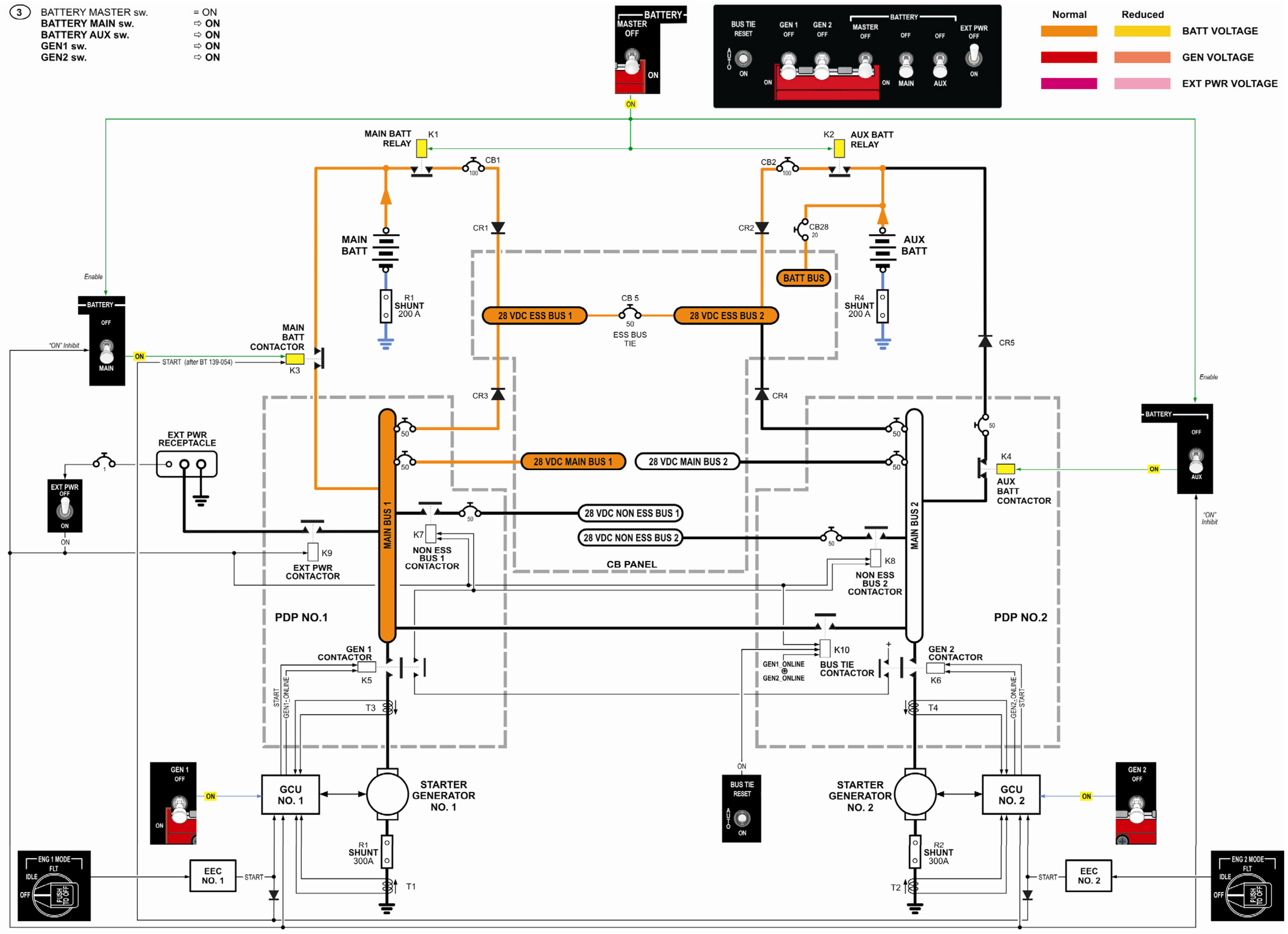
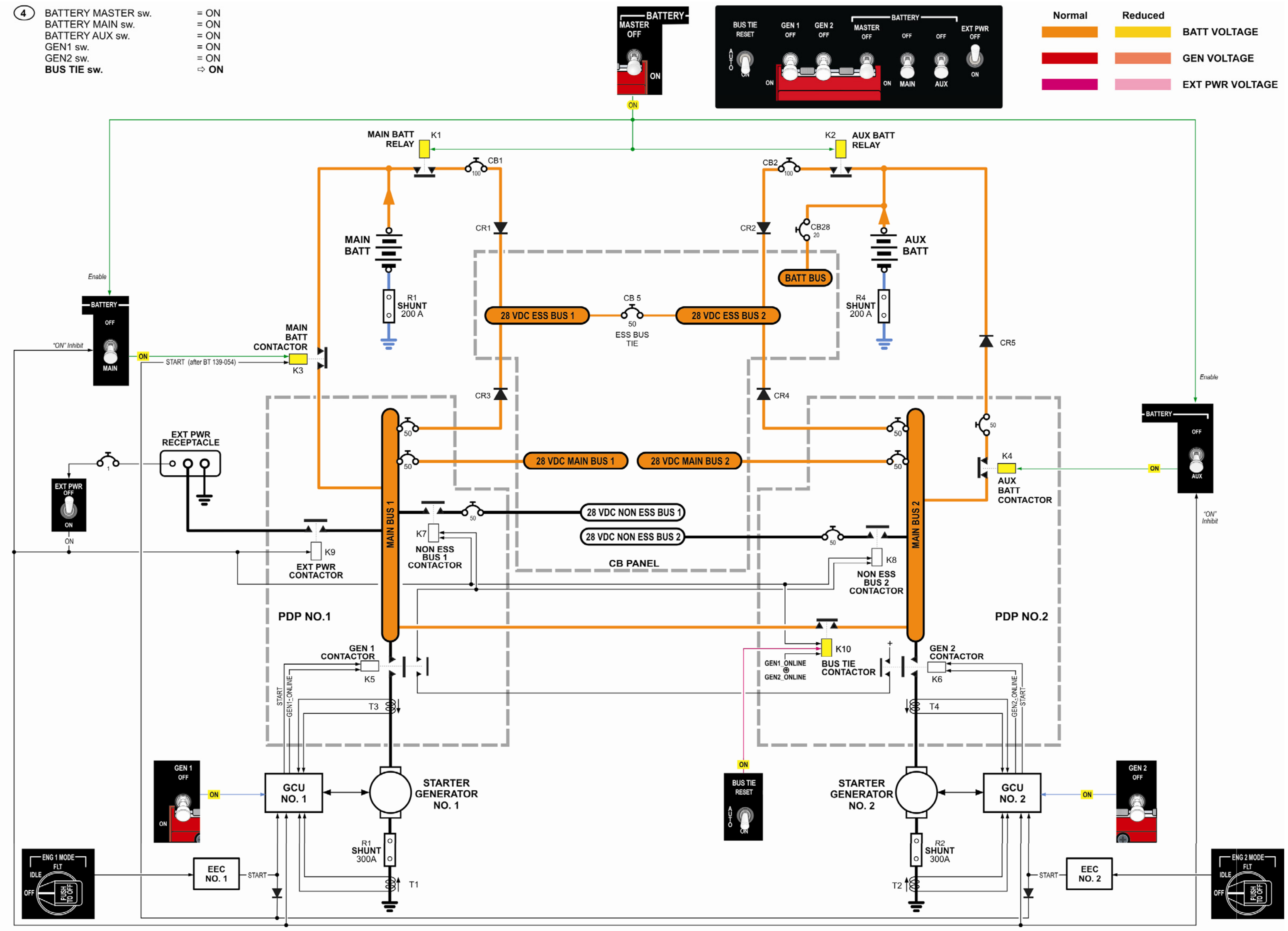


STEP 2 – BATTERY MASTER SWITCH ON



STEP 3 –MAIN AND AUX BATTERY SWITCHES ON



STEP 4 – BUS TIE SWITCH ON

BUS TIE LOGIC

BUS TIE contactor closes if any of the following conditions is met:

- BUS TIE switch = ON
- One GEN = ON-LINE and the other GEN = OFF-LINE
- EXT PWR is connected to helicopter bus bars

STEP 5 – ENGINE 2 STARTING ON BATTERIES

An Engine normal start (AUTO Mode) is initiated by the Engine Electronic Unit (EEC) when pilot moves the relevant ENG MODE switch on the ENG control panel to IDLE or FLIGHT.

This causes the GCU to close the relevant Generator contactor so the starter is fed by the MAIN Battery power via MAIN BUS 2 and cranks the engine.

Due to the great amount of power required to start the engine, MAIN battery voltage drops very much (represented with a paler shade of color).

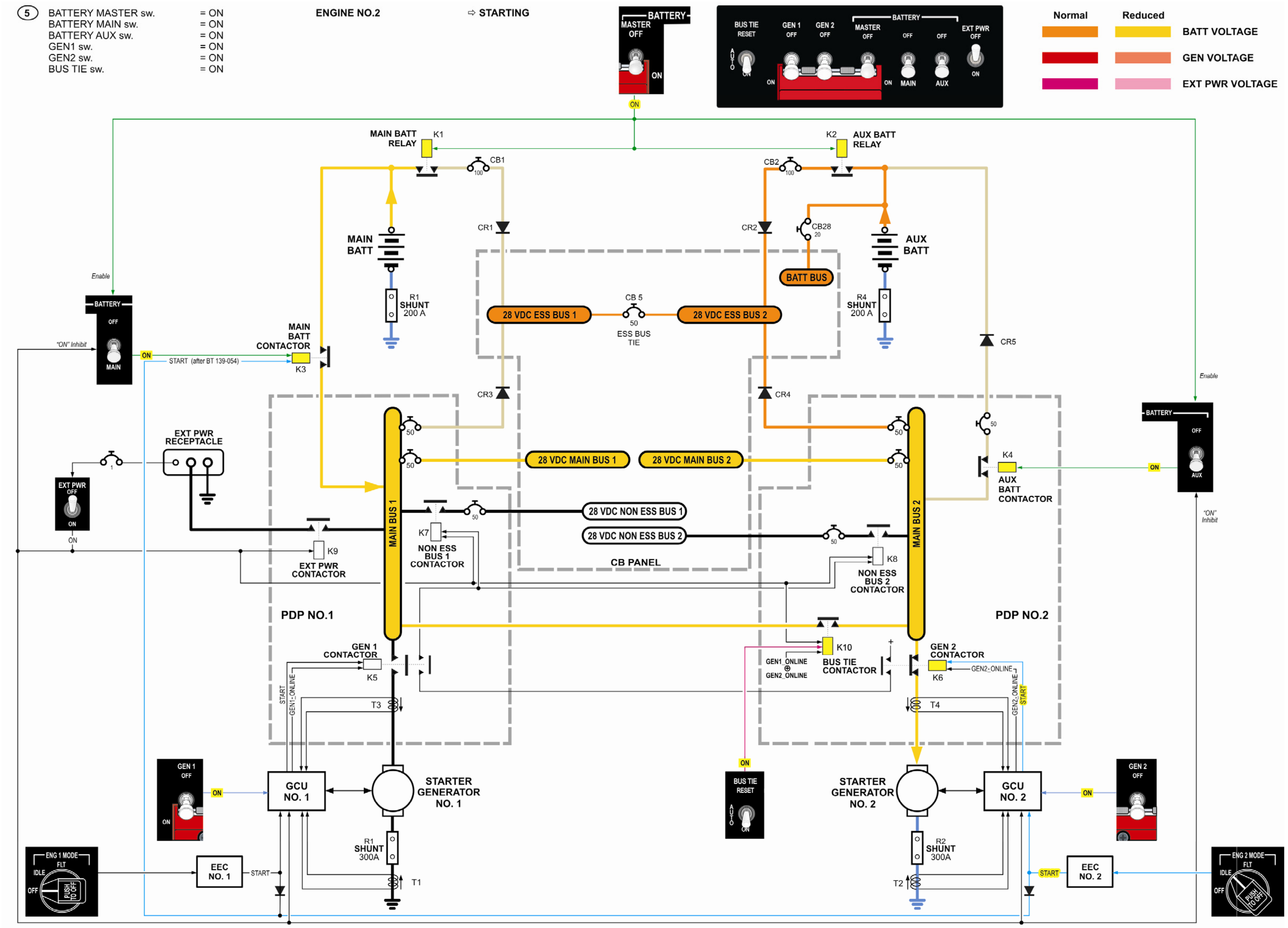
The AUX battery only supplies the ESS BUS 1 and 2, in order to keep a sufficient high voltage for the pilot Display Units (PFD and MFD): this ensures pilots can monitor engine parameters during starting.

Engine start sequence is terminated by the EEC when it detects Ng = 49%: the START signal is removed and the GCU releases the Generator contactor.

The EEC then accelerates the engine to Nf = 65% (IDLE) or Nf = 100% (FLIGHT).

NOTE. An Engine Manual Start (MANUAL Mode) is initiated by pressing the START push-button on the relevant Engine Control Lever (ECL) which directly sends a control signal to the relevant GCU.

In this case the EEC is not involved and the start sequence is terminated by the GCU thanks to the Generator speed sensing signal.



STEP 5 – ENGINE NO.2 STARTING

STEP 6 – GEN 2 ON-LINE

When the starting sequence is terminated, since the GEN 2 switch is at ON, the GCU closes the Generator 2 contactor when 50% Ng is exceeded: GEN 2 is on-line.

GEN 2 feeds all MAIN and ESS busses and recharges the MAIN and AUX batteries.

The BUS TIE contactor is closed because of the following two conditions:

- BUS TIE switch = ON
- One GEN = ON-LINE and the other GEN = OFF-LINE

STEP 7 – ENGINE 1 STARTING: ASSISTED START

When the second engine is started on batteries (in the example engine no. 1), the generator that is already on-line assists the MAIN battery in feeding the starter: this condition is called assisted start.

STEP 8 – BOTH GENERATORS ON-LINE

After the second engine is started, also the second generator is automatically connected on-line by the relevant GCU.

BUS TIE switch is then returned to AUTO for normal operation

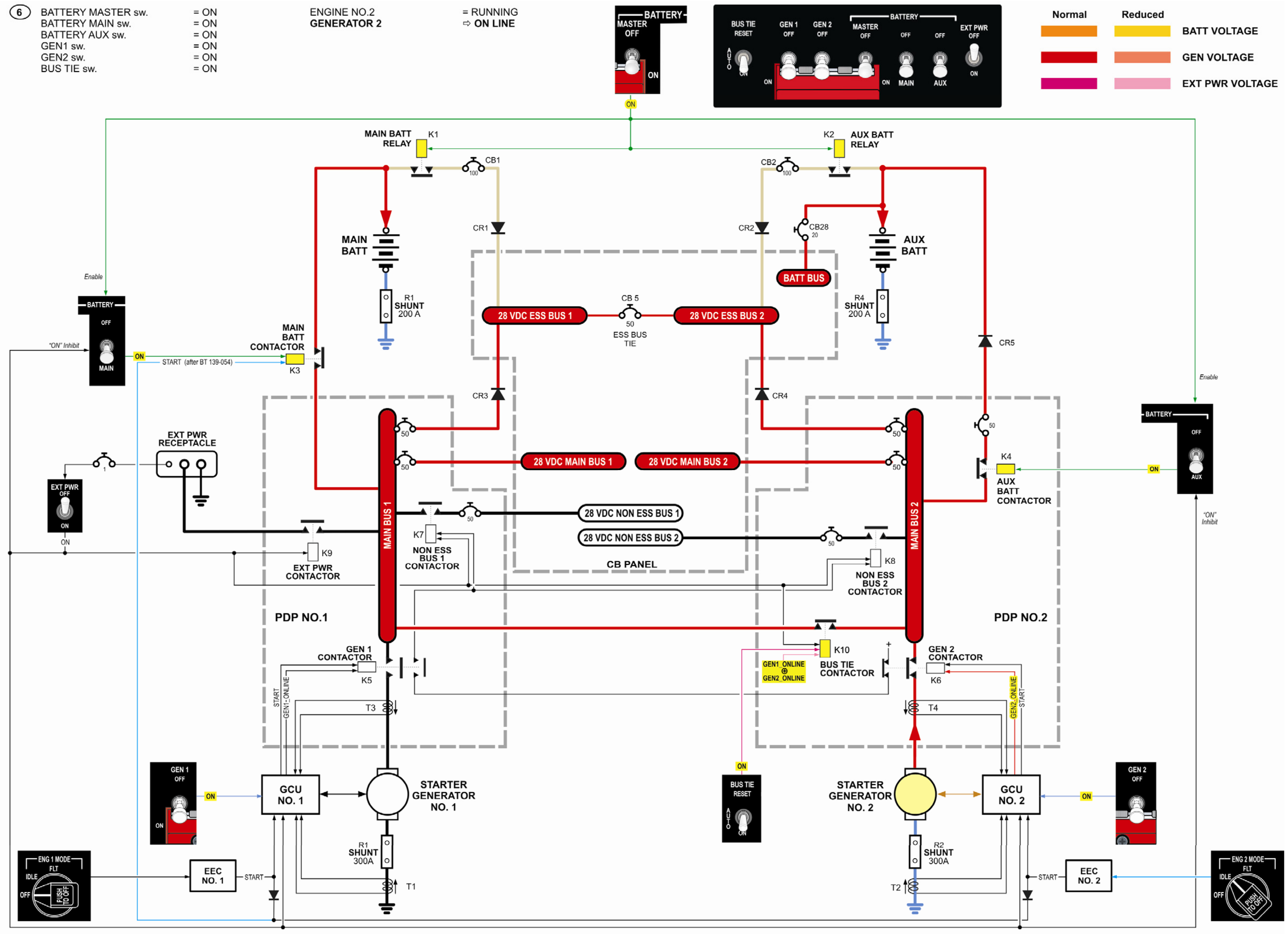
With both generators on-line, either on the ground or during flight all DC busses are powered by the two generators separately:

- BUS TIE contactor is open
- NON-ESS BUS 1 and NON-ESS BUS 2 are energized
- The two batteries are charged by GEN 1 (MAIN battery) and GEN 2 (AUX battery)

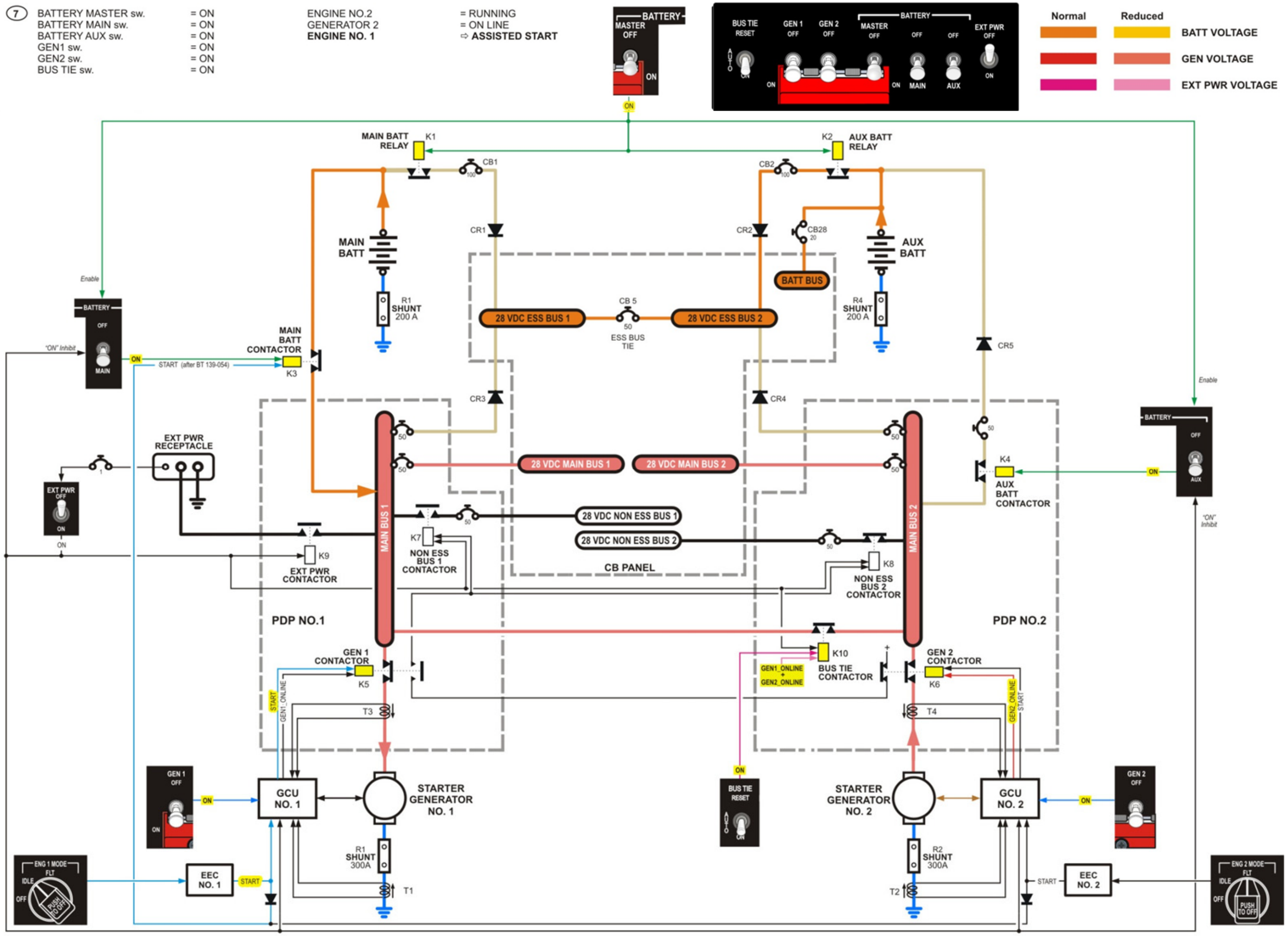
NON-ESS BUS LOGIC

NON-ESS BUS 1 and NON-ESS BUS 2 are energized if any of the following conditions is met:

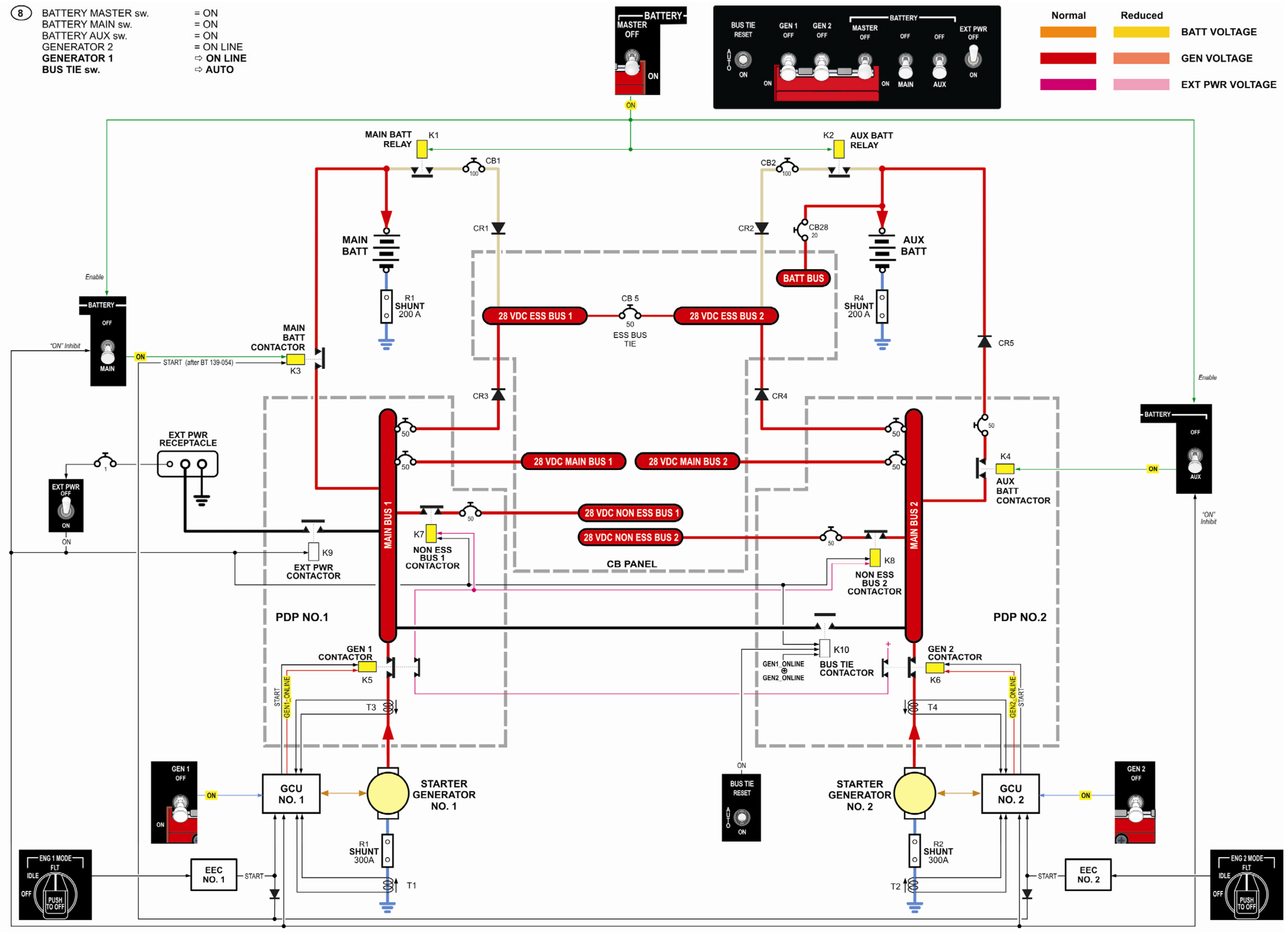
- Both GEN = ON-LINE
- EXT PWR is connected to helicopter bus bars



STEP 6 – GEN 2 ON LINE



STEP 7 – ENGINE NO.1 ASSISTED START



STEP 8 – GEN 1 AND GEN 2 ON–LINE AND BUS TIE SWITCH AUTO

EXTERNAL POWER STARTING

STEP E1 – EXTERNAL POWER AVAILABLE AT RECEPTACLE

When an external power voltage is available at the external power receptacle, it is checked for reverse polarity and overvoltage but it is not automatically connected to the aircraft distribution bus bars.

STEP E2 – CONNECTION OF THE EXTERNAL POWER

The EXT PWR switch on the Electrical control panel allows the pilot to control the connection of the external power source to the distribution bus bars. When a valid external power is available at the receptacle and the EXT PWR switch is moved to ON:

- EXT PWR contactor closes
- BUS TIE contactor automatically closes
- NON-ESS BUS 1 & NON-ESS BUS 2 contactors close
- MAIN and AUX battery connection circuits to MAIN BUS 1 and 2 are inhibited to prevent recharging of the batteries from external power
- Both GCU's are fed with a control signal that prevents generators to be put on-line while external power is feeding the aircraft loads

The external power supplies all distribution bus bars (ESS, MAIN and NON-ESS).

NOTE. Battery bus is supplied by the AUX battery.

STEP E3 – BATTERIES AND EXT PWR

The following diagrams represent the sequence of actions for a normal battery starting of the engines using the external power. In this example engine no. 1 is started first. Refer to RFM – Section 2 for Normal procedures.

Steps 1 thru 3 discussed for Battery Starting are to be performed also in case of External Power Starting. When EXT PWR switch is then moved to ON:

- MAIN BATT contactor and AUX BATT contactor open to prevent recharging of the batteries from external power

NOTE: Diodes CR1 and CR2 prevent the current flow from the ESS BUS 1 and 2 to the MAIN and AUX batteries

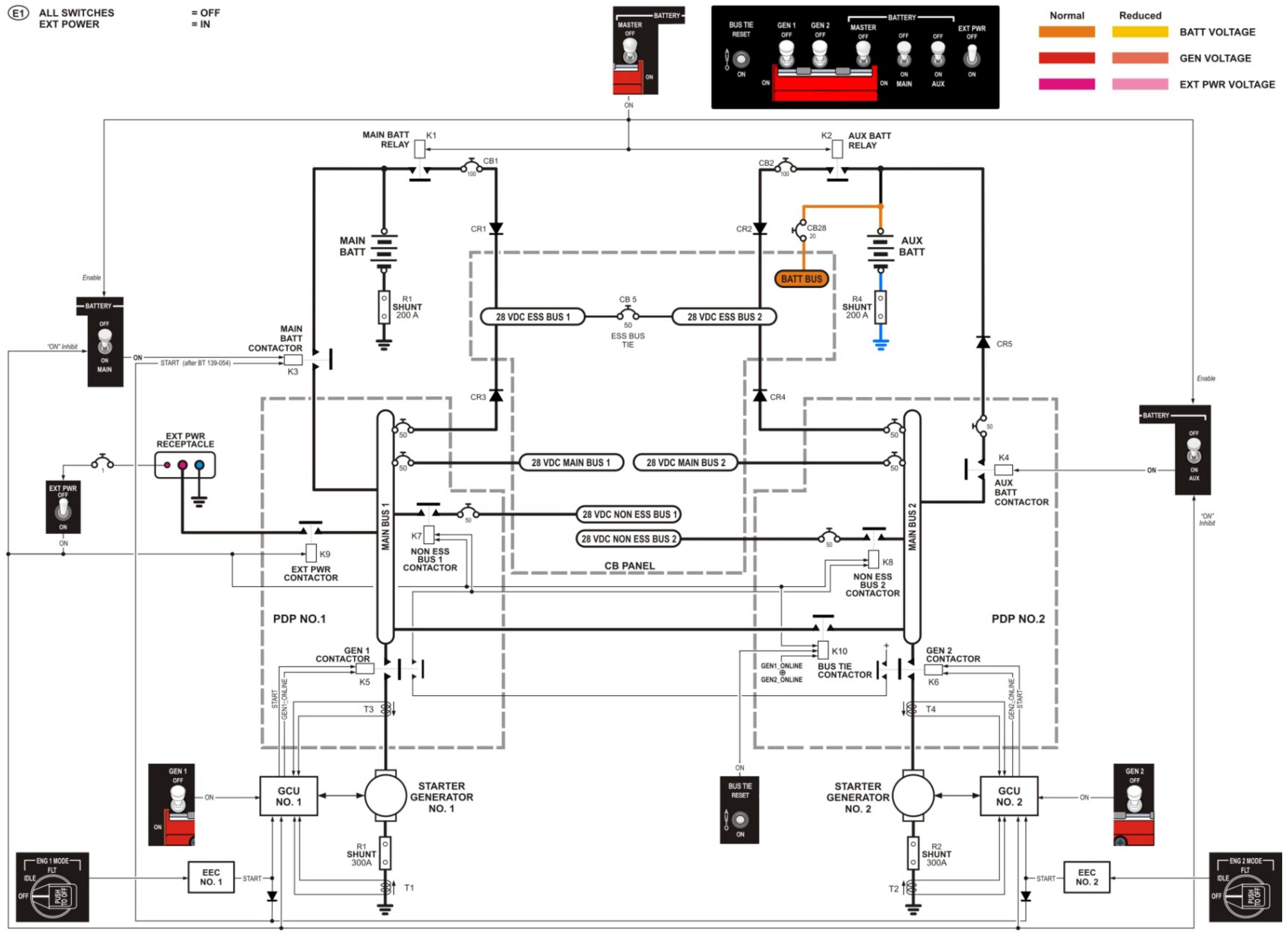
STEP E4 – ENGINE 1 STARTING

The Bollettino Tecnico 139-054 (Technical Bulletin 139-054) has introduced the concept of assisted start also when starting on External Power. During the start sequence (START control signal is active) the MAIN battery is connected to MAIN BUS 1 to allow a soft change-over of the starter power source in case the external power fails.

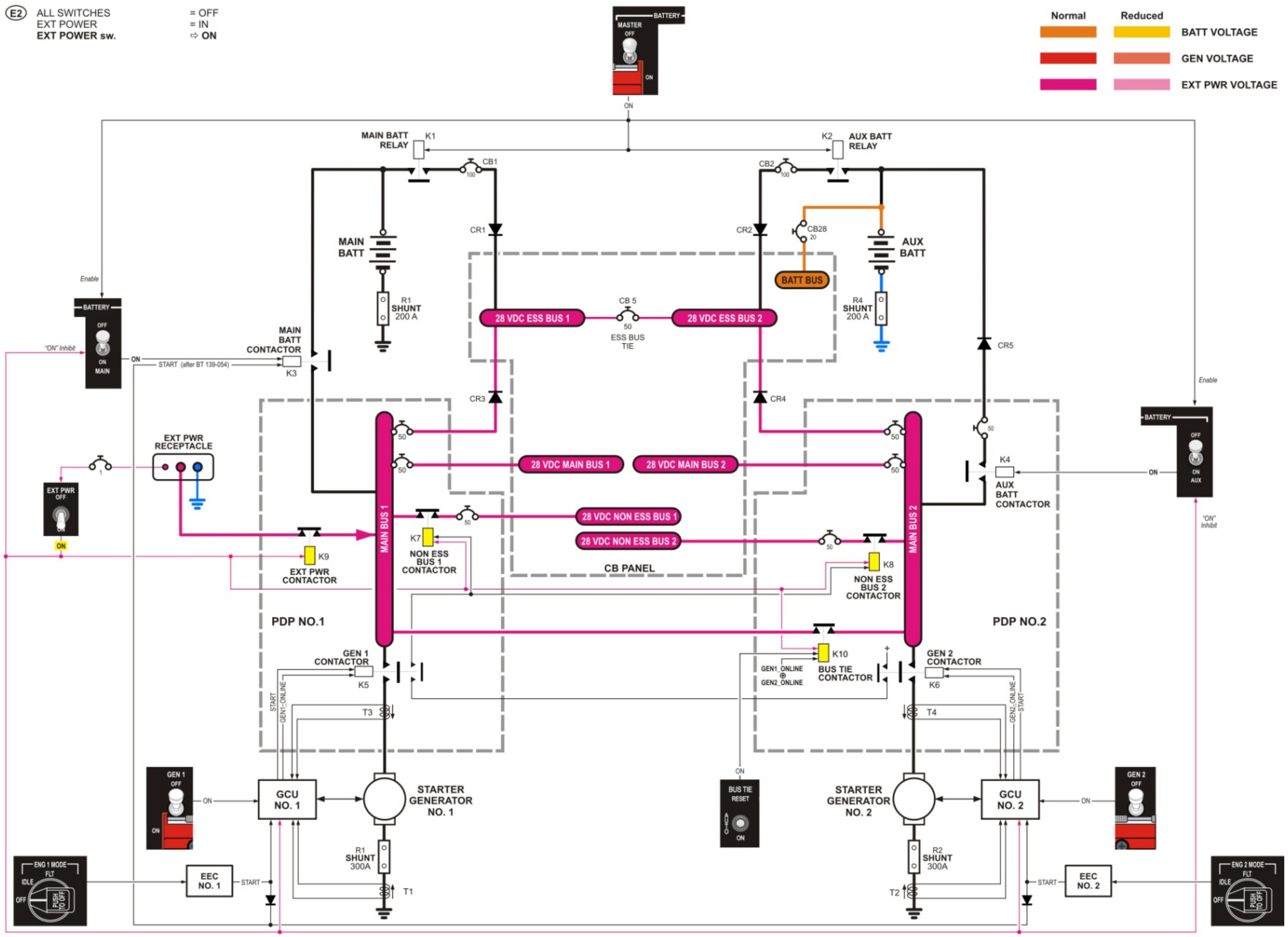
NOTE. BUS TIE switch is to be moved to ON if engine no. 2 is started first.

STEP E5 – ENGINE RUNNING

After engine starting, Generators remain off-line until the EXT PWR switch is returned to OFF.



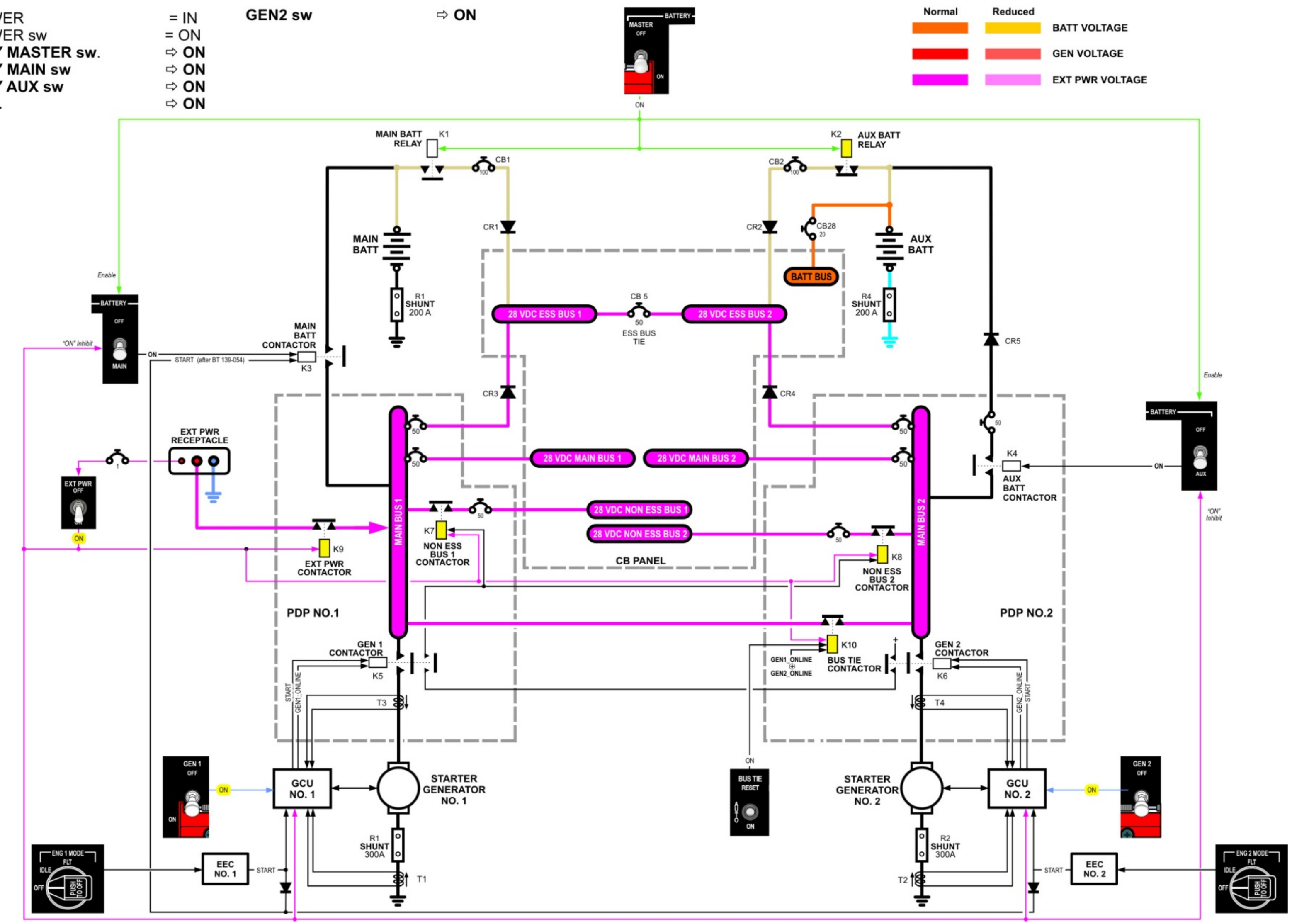
STEP E1 – EXTERNAL POWER IN



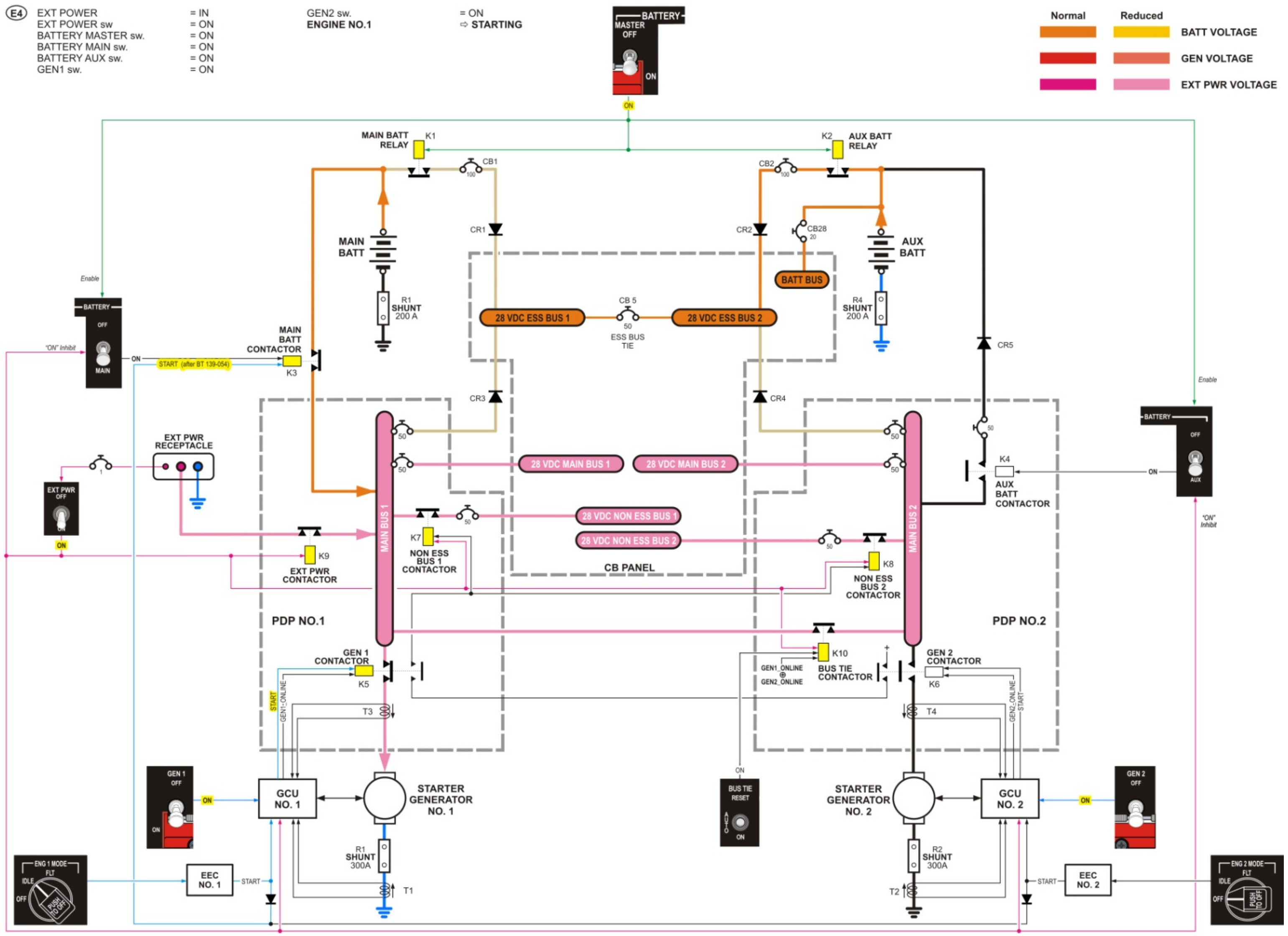
STEP E2 – EXTERNAL POWER SWITCH ON

E3

EXT POWER	= IN	**GEN2 sw**	⇨ **ON**
EXT POWER sw	= ON		
BATTERY MASTER sw.	⇨ **ON**		
BATTERY MAIN sw	⇨ **ON**		
BATTERY AUX sw	⇨ **ON**		
GEN1 sw.	⇨ **ON**		

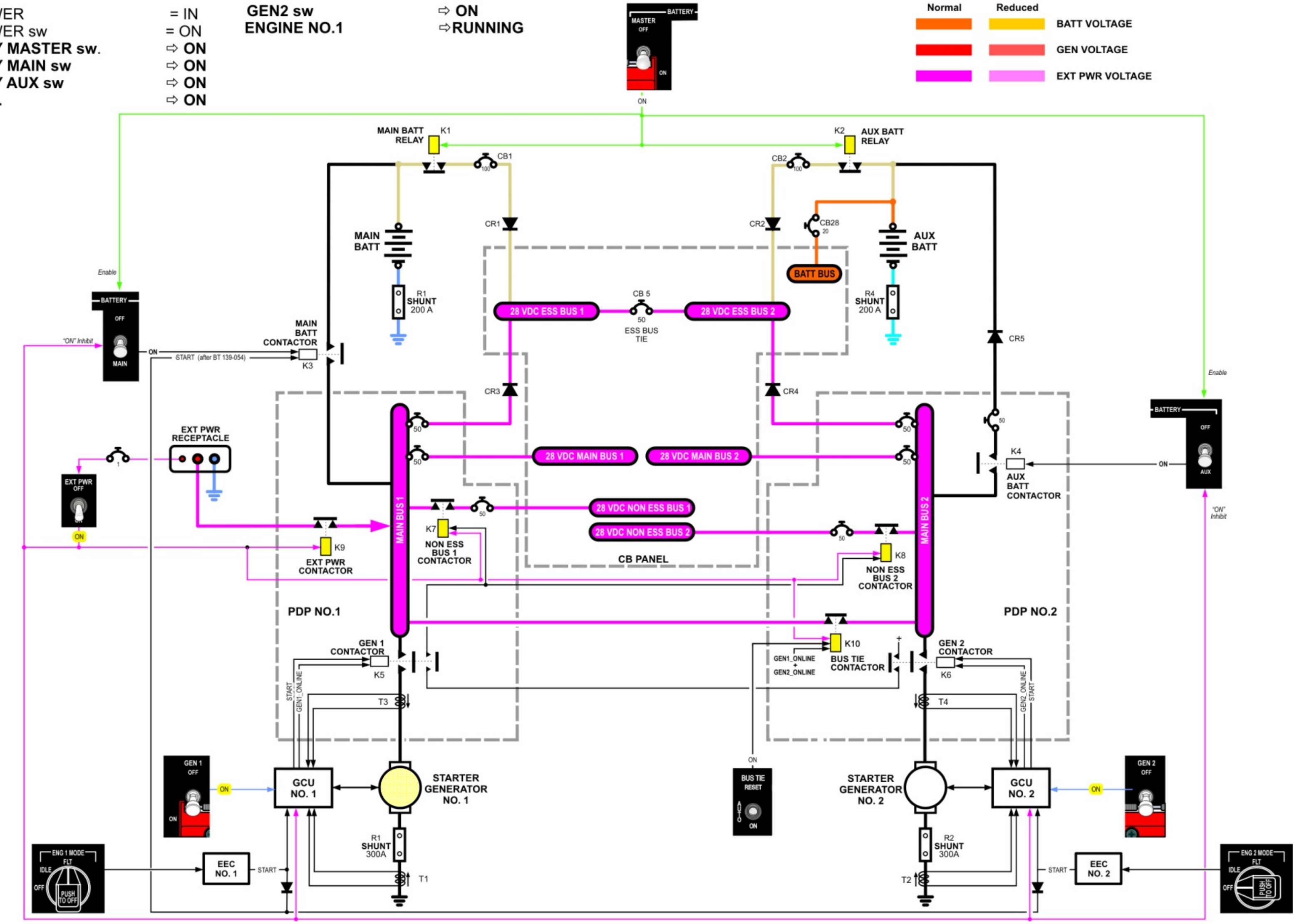


STEP E3 – BATTERY AND GENERATOR SWITCHES ON



STEP E4 – ENGINE NO.1 STARTING

(E5) EXT POWER = IN
EXT POWER sw = ON
BATTERY MASTER sw. ⇨ ON
BATTERY MAIN sw ⇨ ON
BATTERY AUX sw ⇨ ON
GEN1 sw. ⇨ ON
GEN2 sw ⇨ ON
ENGINE NO.1 ⇨RUNNING



STEP E5 – ENGINE NO.1 RUNNING

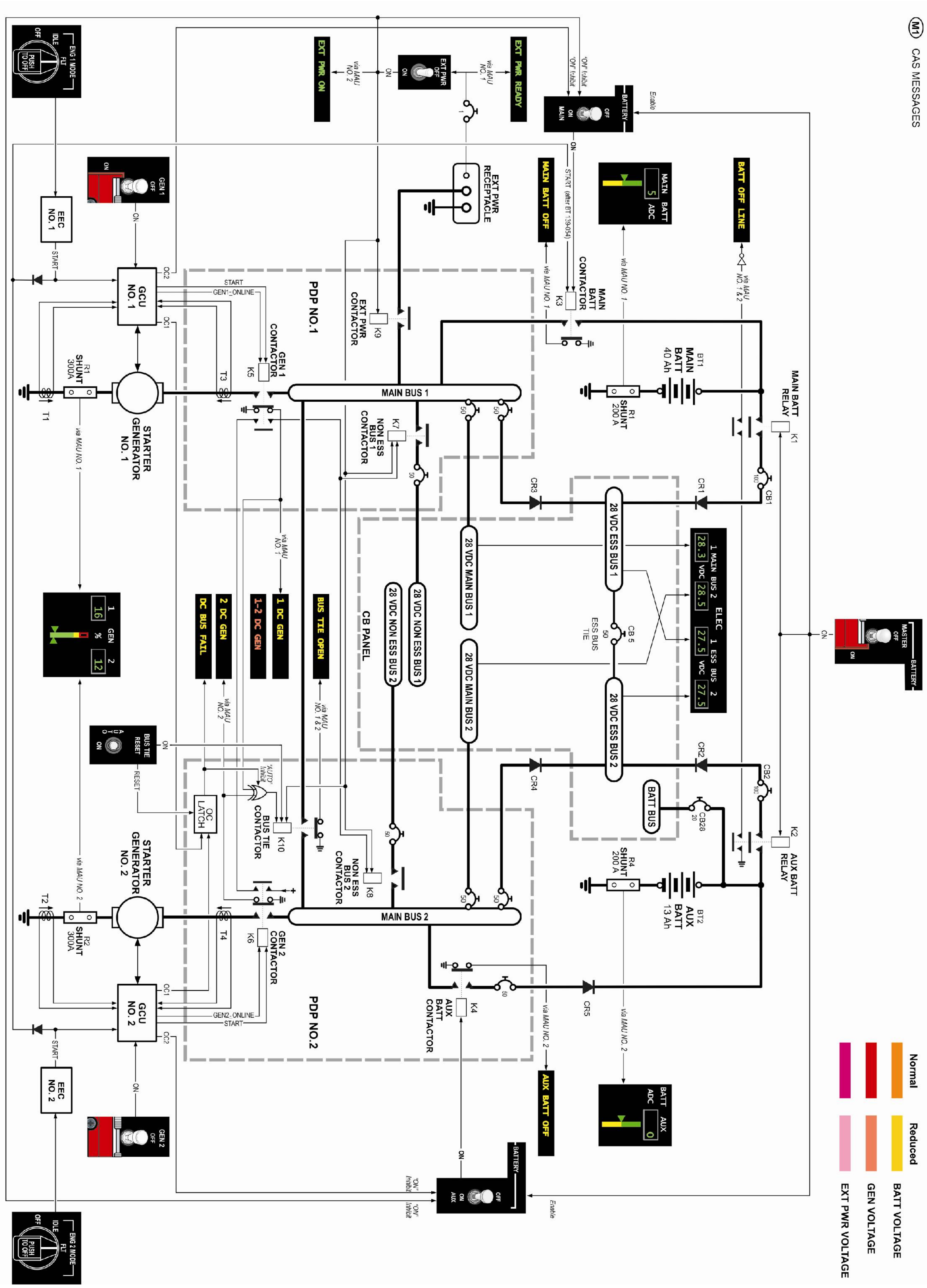
INDICATIONS AND IN – FLIGHT NORMAL OPERATION

The following simplified diagrams (M1, M2) integrate the previous diagram information with the interfacing of the Electrical power system indicators and the major CAS messages.

M1 is meant to just show the indications.

M2 shows the normal system operation and indications during flight.

NOTE. Values are only given as examples.



M1 – CAS MESSAGES

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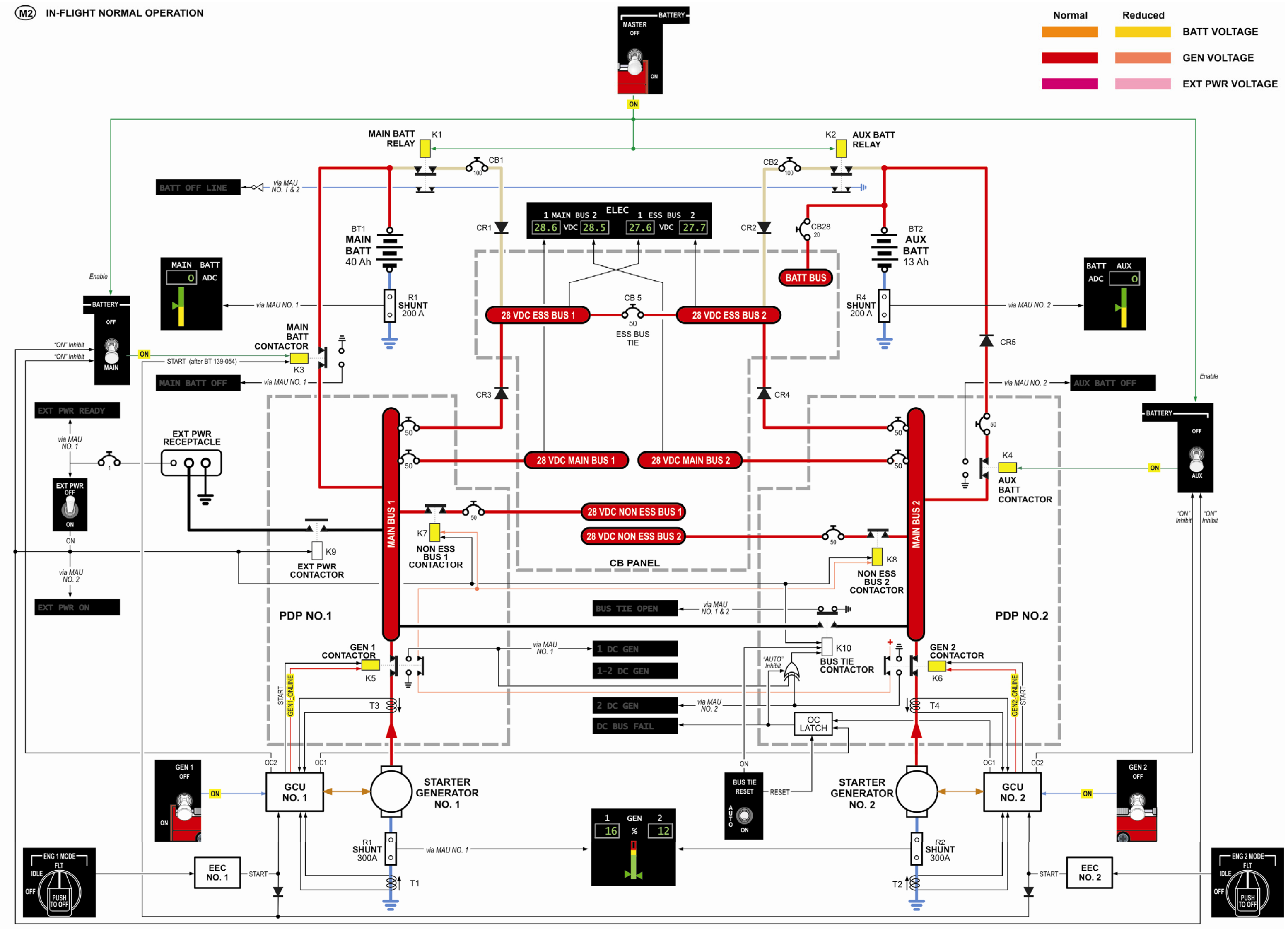


M2 – IN-FLIGHT NORMAL OPERATION

IN-FLIGHT EMERGENCIES AND MALFUNCTIONS

In the following pages the most significant emergencies and malfunctions are detailed using the simplified schematic diagram (F1 to F5). Refer to AW139-RFM-4D – Section 3 for Emergency and Malfunction procedures.

F1 – SINGLE GENERATOR FAILURE (1 DC GEN)

When a GCU detects that the relevant Generator has failed, it automatically open the Generator contactor.

The same CAS message is displayed when the GEN switch is moved to OFF while the relevant engine is running.

- BUS TIE contactor automatically closes
- NON-ESS BUS 1 and NON-ESS BUS 2 are disconnected

F2 – MOMENTARY OVERCURRENT CONDITION (DC BUS FAIL)

When a GCU detects an overcurrent exceeding a first threshold (OC1), it triggers a latching logic that:

- Drives the DC BUS FAIL message in the CAS
- Inhibits the BUS TIE contactor automatic operation (remains open)

Once triggered, the OC latching logic can only be reset by moving the BUS TIE switch to RESET.

F3 – SHORT CIRCUIT (DC BUS FAIL + 1 DC GEN)

If the overcurrent condition persists a second threshold is then exceeded (OC2): that is considered a short circuit.

In this case the relevant GCU also:

- disconnects the relevant Generator, causing the 1 DC GEN or the 2 DC GEN message to be displayed
- disconnects the relevant Battery to prevent discharging on the short circuit, causing the MAIN BATT OFF or the AUX BATT OFF message to be displayed

BUS TIE contactor remains open to prevent the failed bus from being connected to the only available generator.

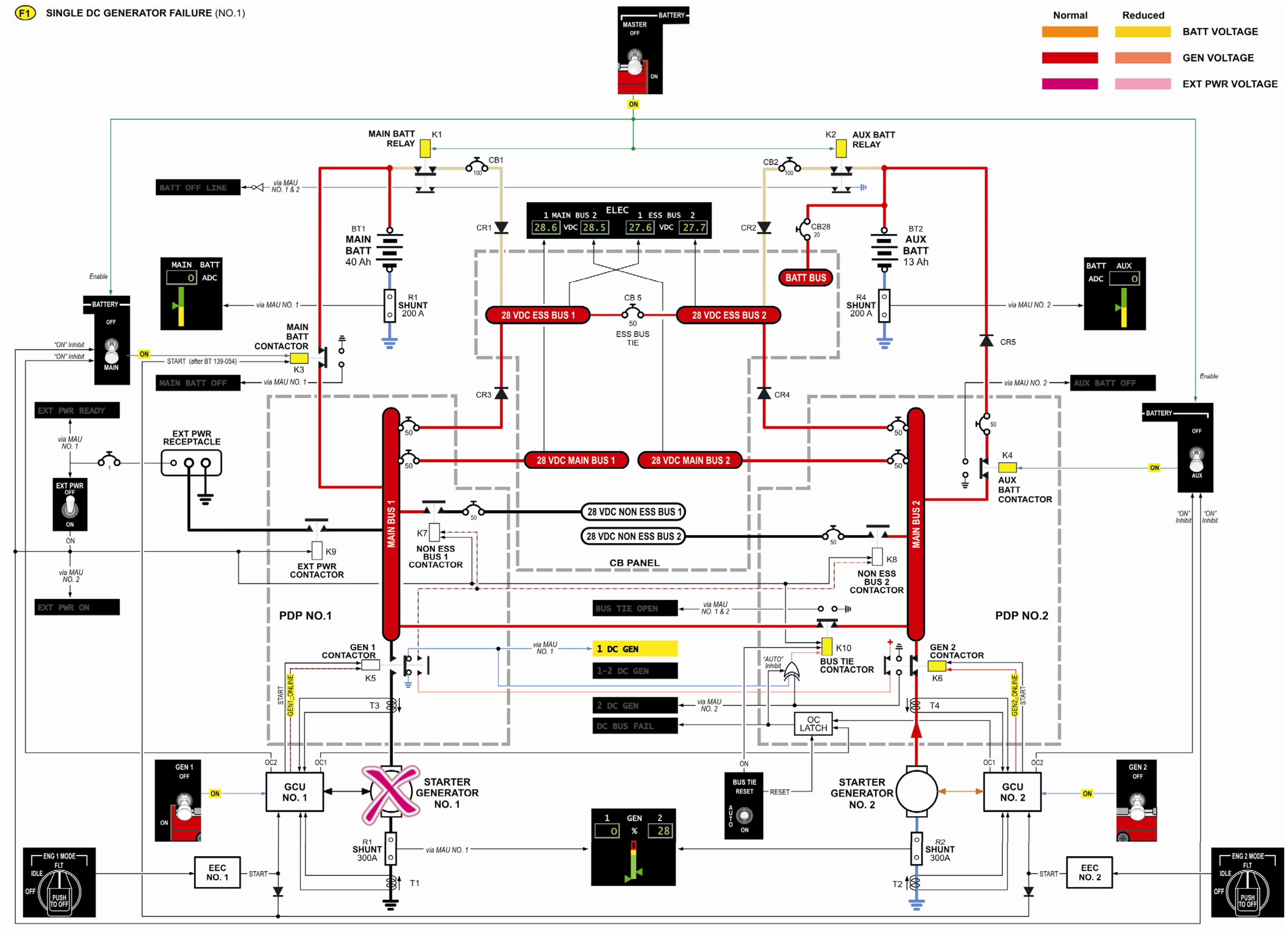
NOTE. Pilot must do nothing.

F4, F5 – DUAL GENERATOR FAILURE (1-2 DC GEN)

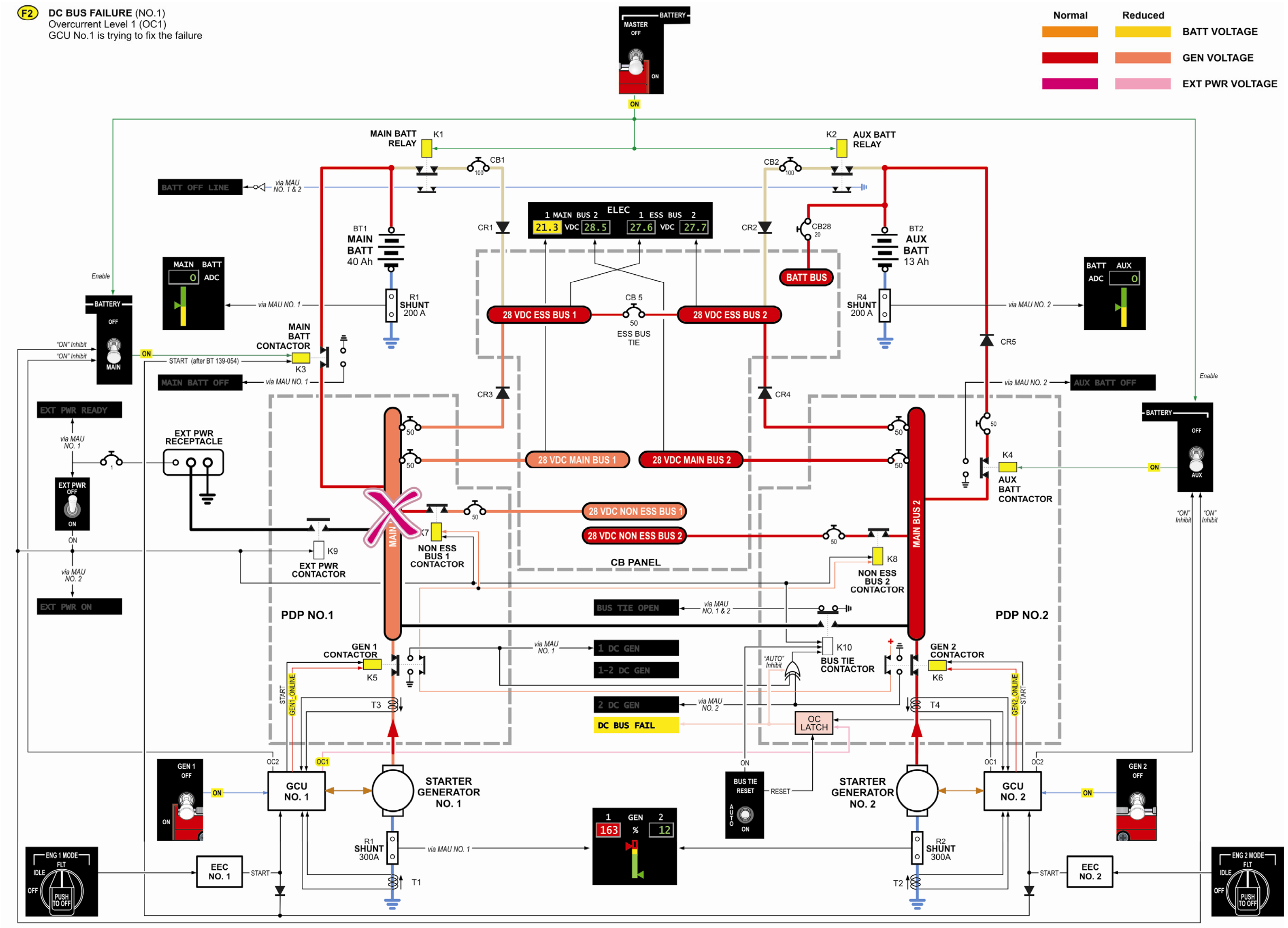
When both Generators have failed only MAIN and AUX Batteries supply the aircraft loads. Manual shedding of the MAIN BUS 1 (BATTERY MAIN switch to OFF) is required in order to only supply the Essential (Emergency) loads.

F6 - ELECTRICAL BLACK OUT

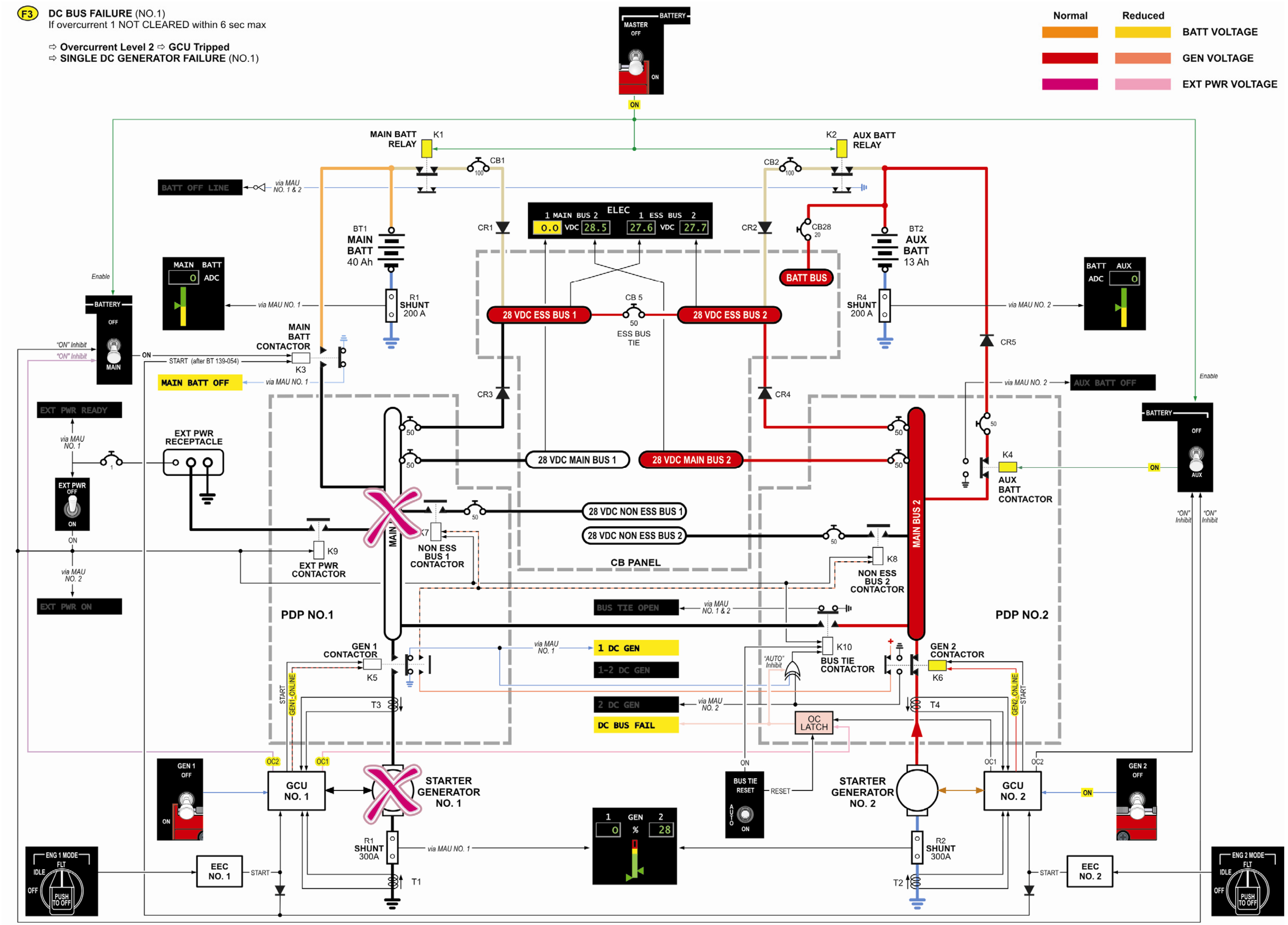
Following a dual generator failure, if batteries discharge completely (electrical black out) the engines continue to run in AUTO Mode: they are controlled by the EEC which is then electrically powered by the relevant Permanent Magnet Alternator (PMA). Since the landing gear is electrically actuated, it is necessary to extend it before losing electrical power completely; an increase in fuel consumption is then to be expected.



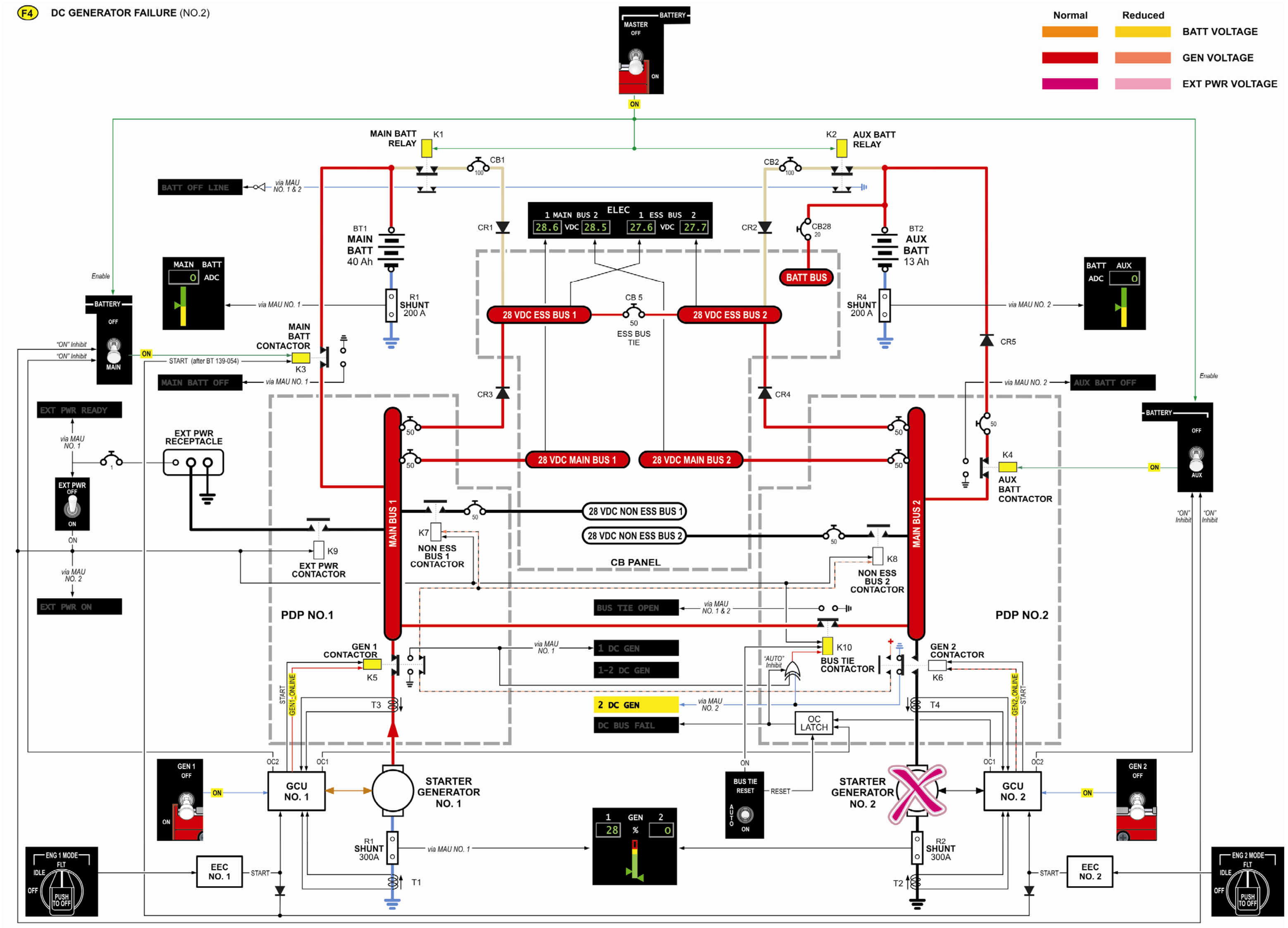
F1 – SINGLE DC GENERATOR FAILURE (NO.1)



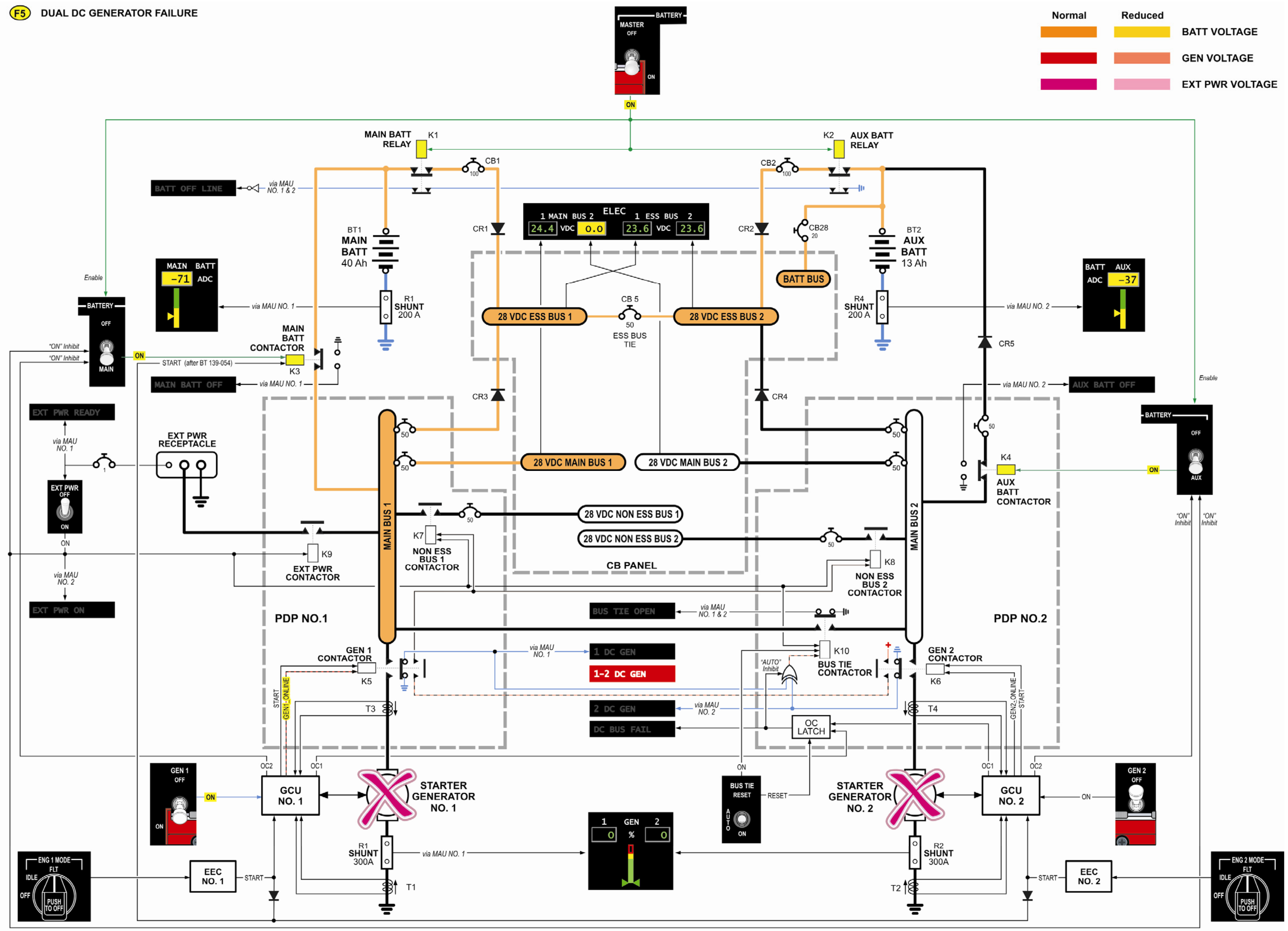
F2 – DC BUS FAILURE (NO.1) – OVERCURRENT LEVEL 1



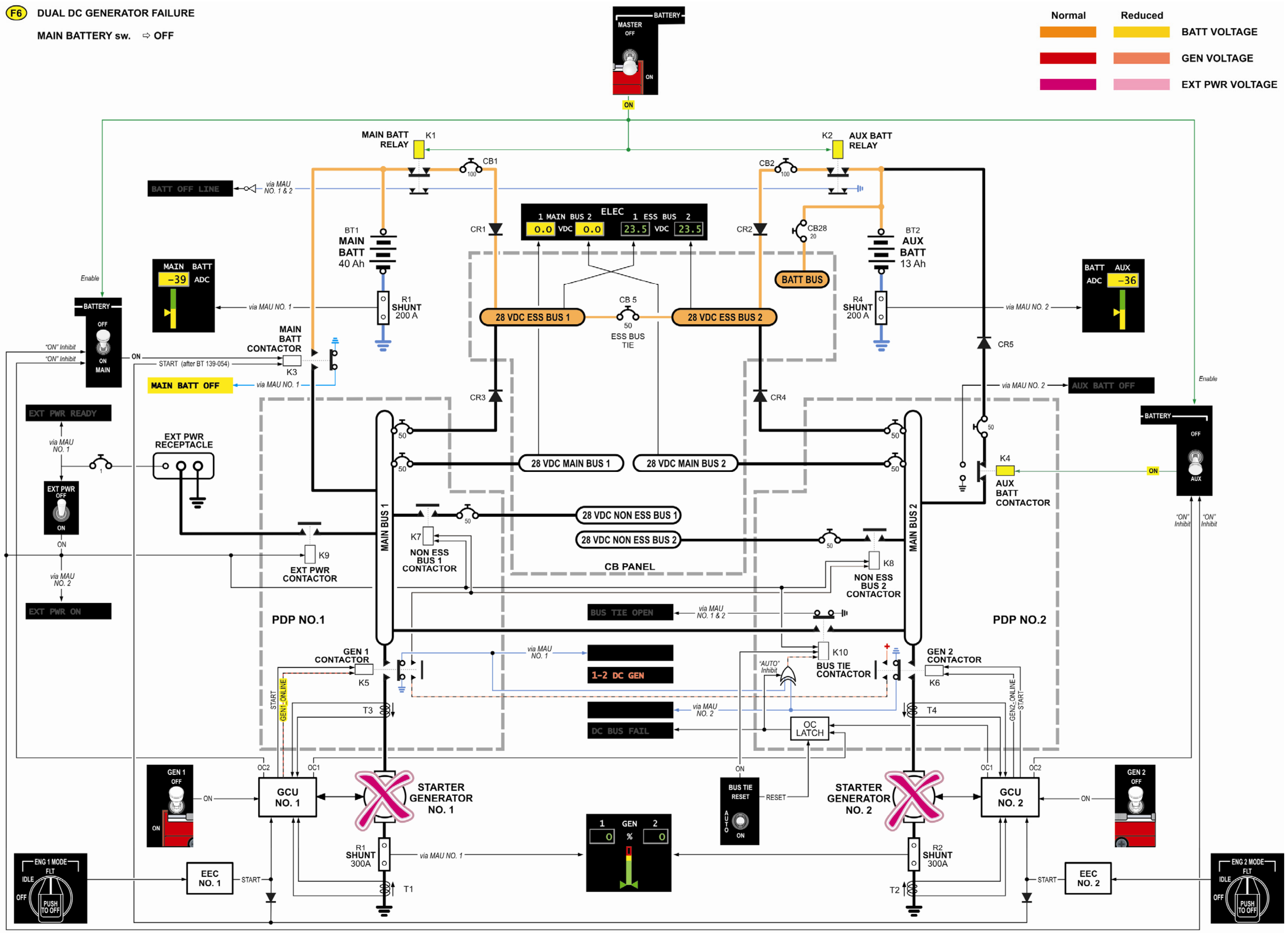
F3 – DC BUS FAILURE (NO.1) - IF OVERCURRENT 1 NOT CLEARED WITHIN 6 SEC MAX



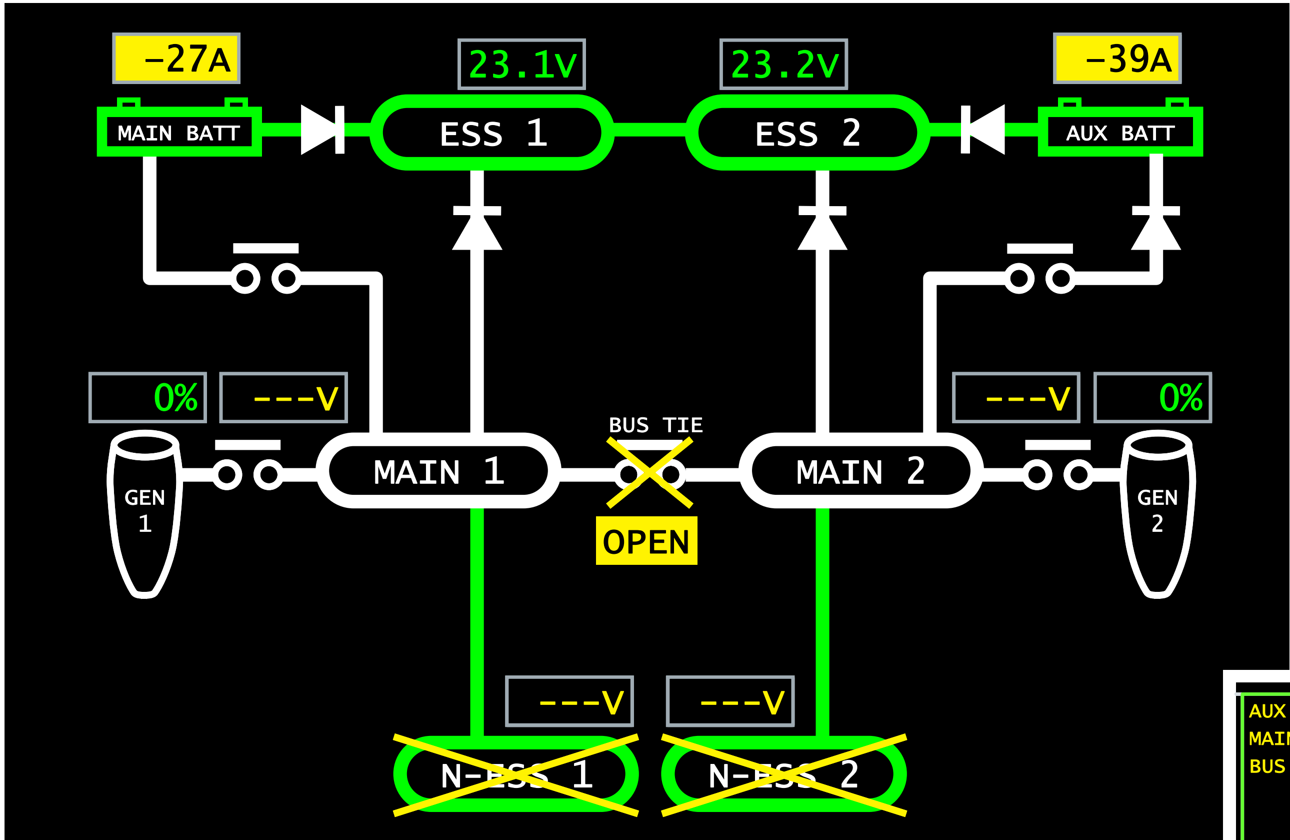
F4 – DC GENERATOR FAILURE (NO.2)

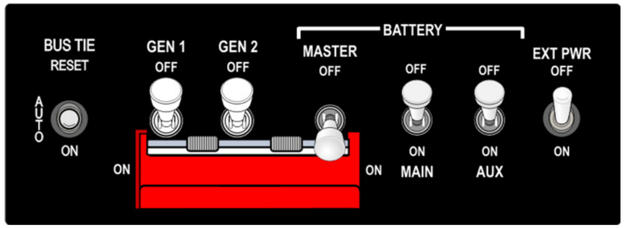


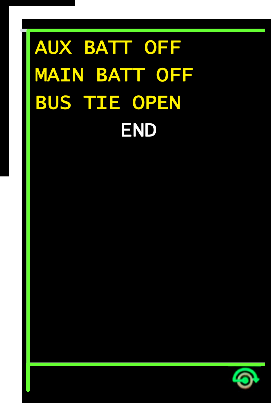
F5 – DUAL DC GENERATOR FAILURE



F6 – DUAL DC GENERATOR FAILURE – MAIN BATTERY SWITCH OFF

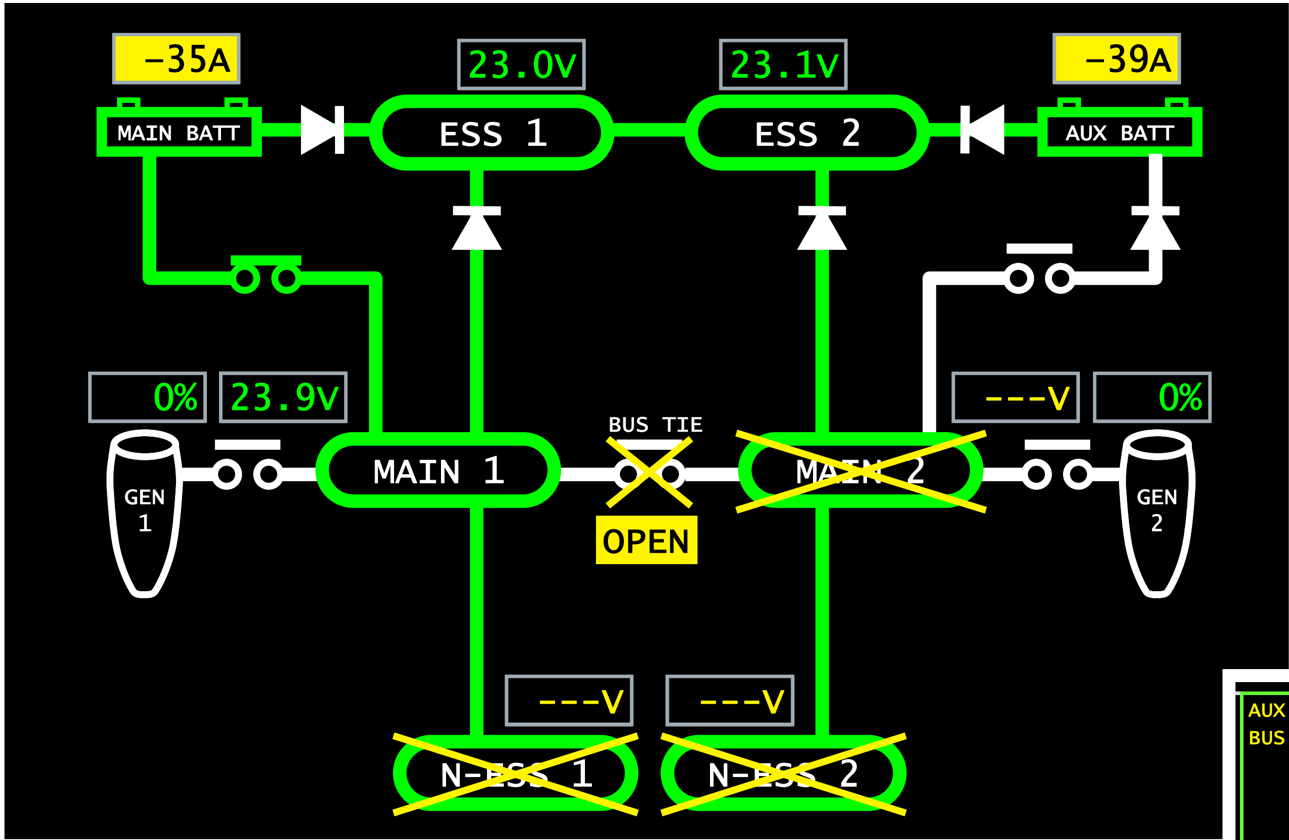


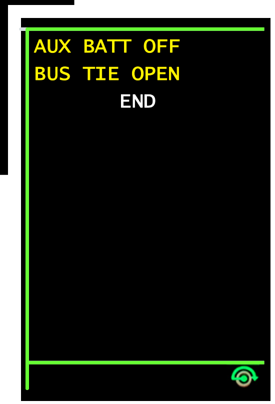




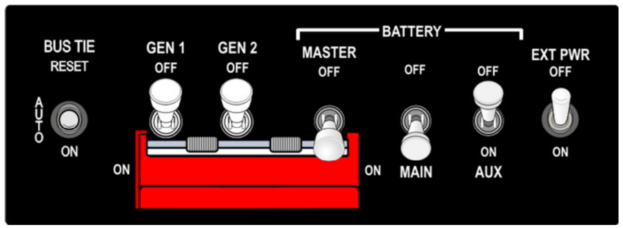
Note: CAS Messages shown are only
those related to DC Power system

DC POWER SYNOPTIC – BATT MASTER = ON

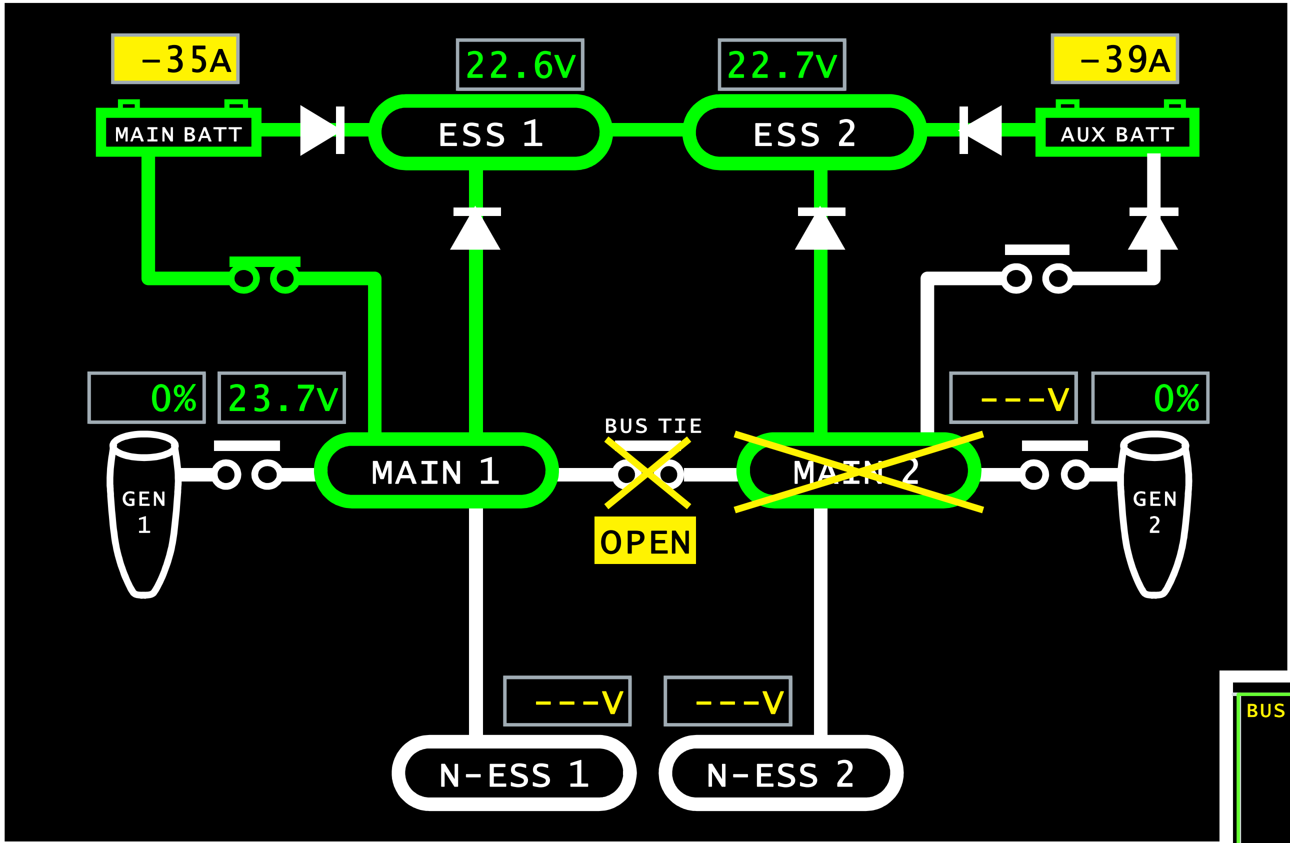


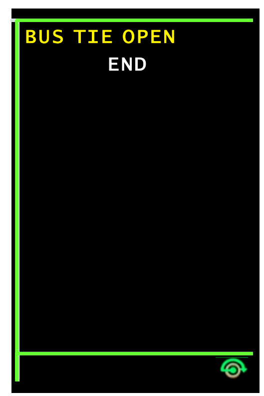


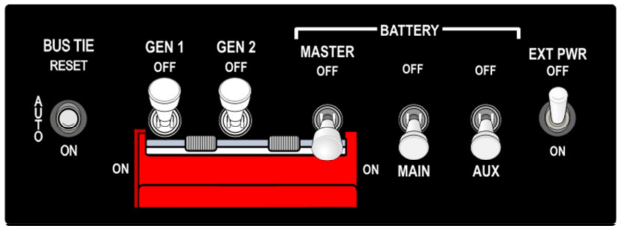
Note: CAS Messages shown are only
those related to DC Power system



DC POWER SYNOPTIC – BATT MASTER + MAIN = ON

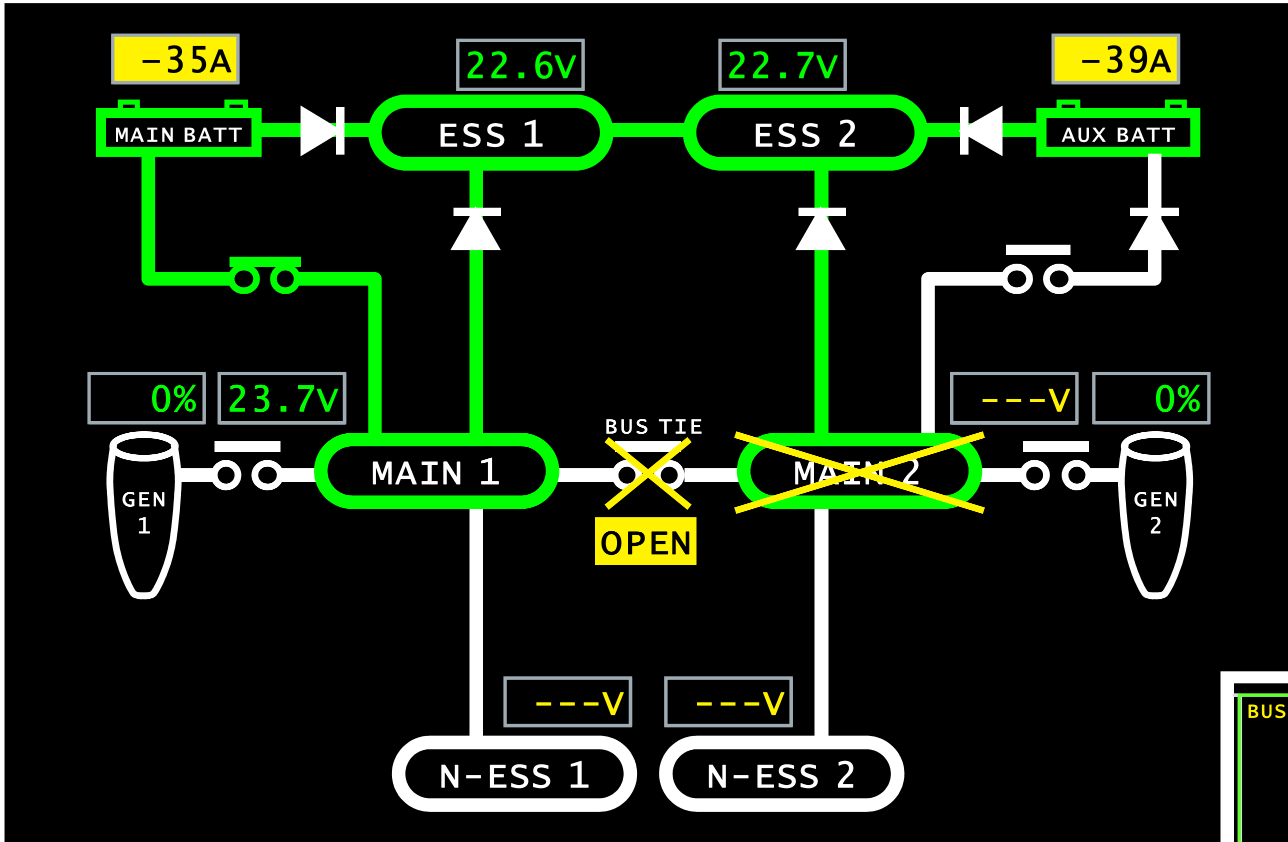


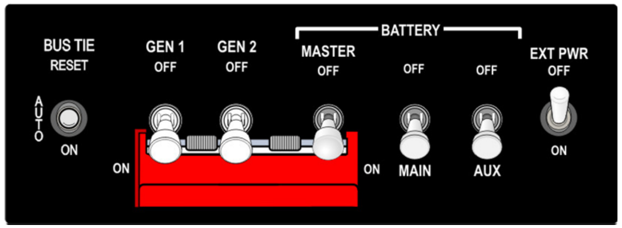


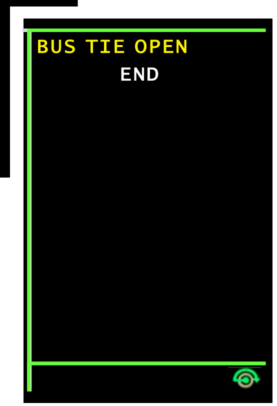


Note: CAS Messages shown are only those related to DC Power system

DC POWER SYNOPTIC – BATT MASTER + MAIN + AUX = ON

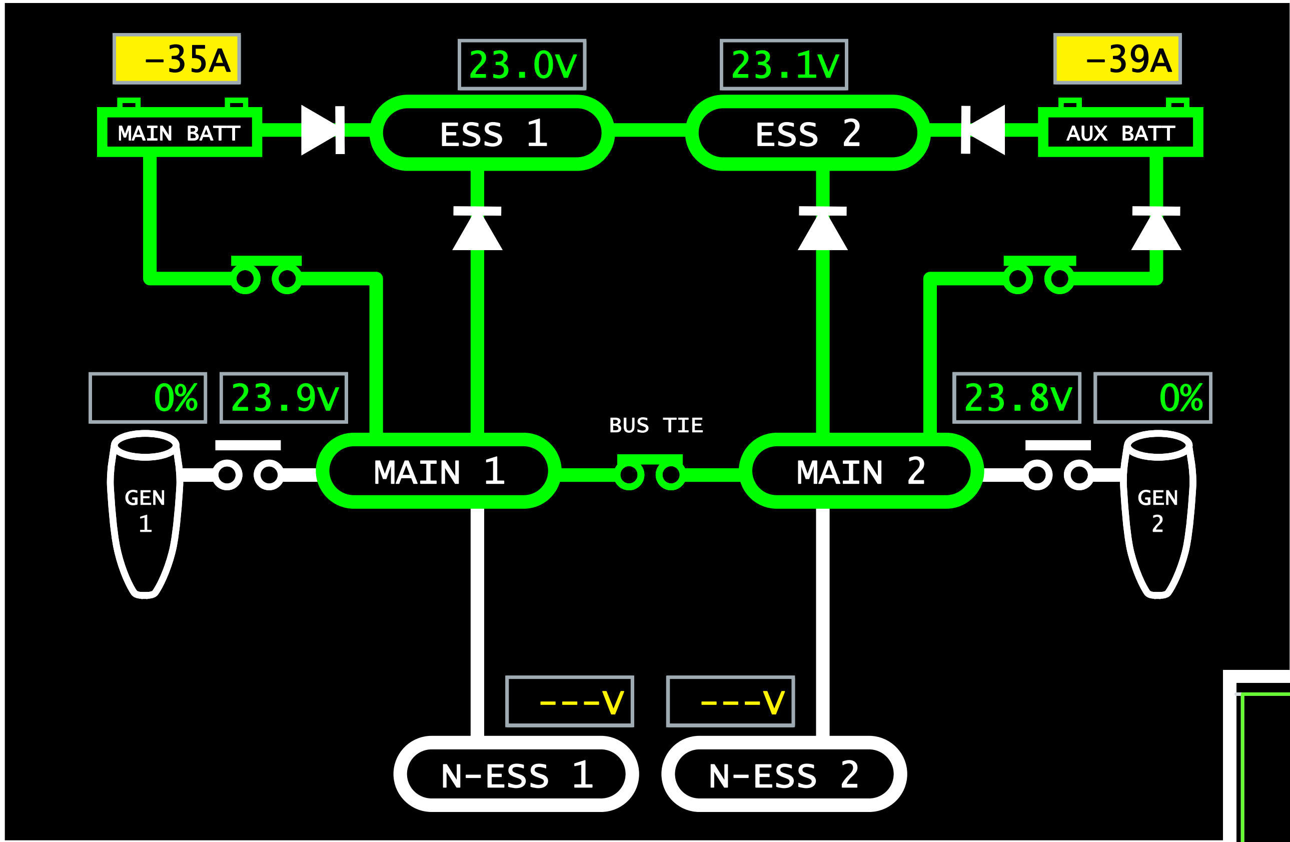


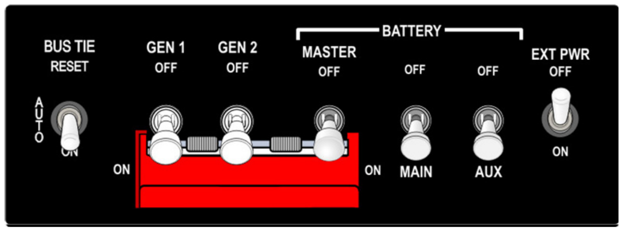


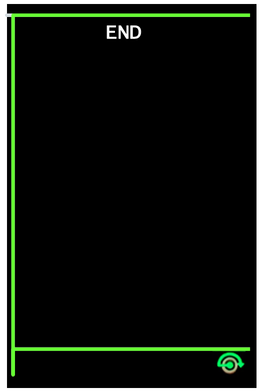


Note: CAS Messages shown are only those related to DC Power system

DC POWER SYNOPTIC – BATT + GEN = ON

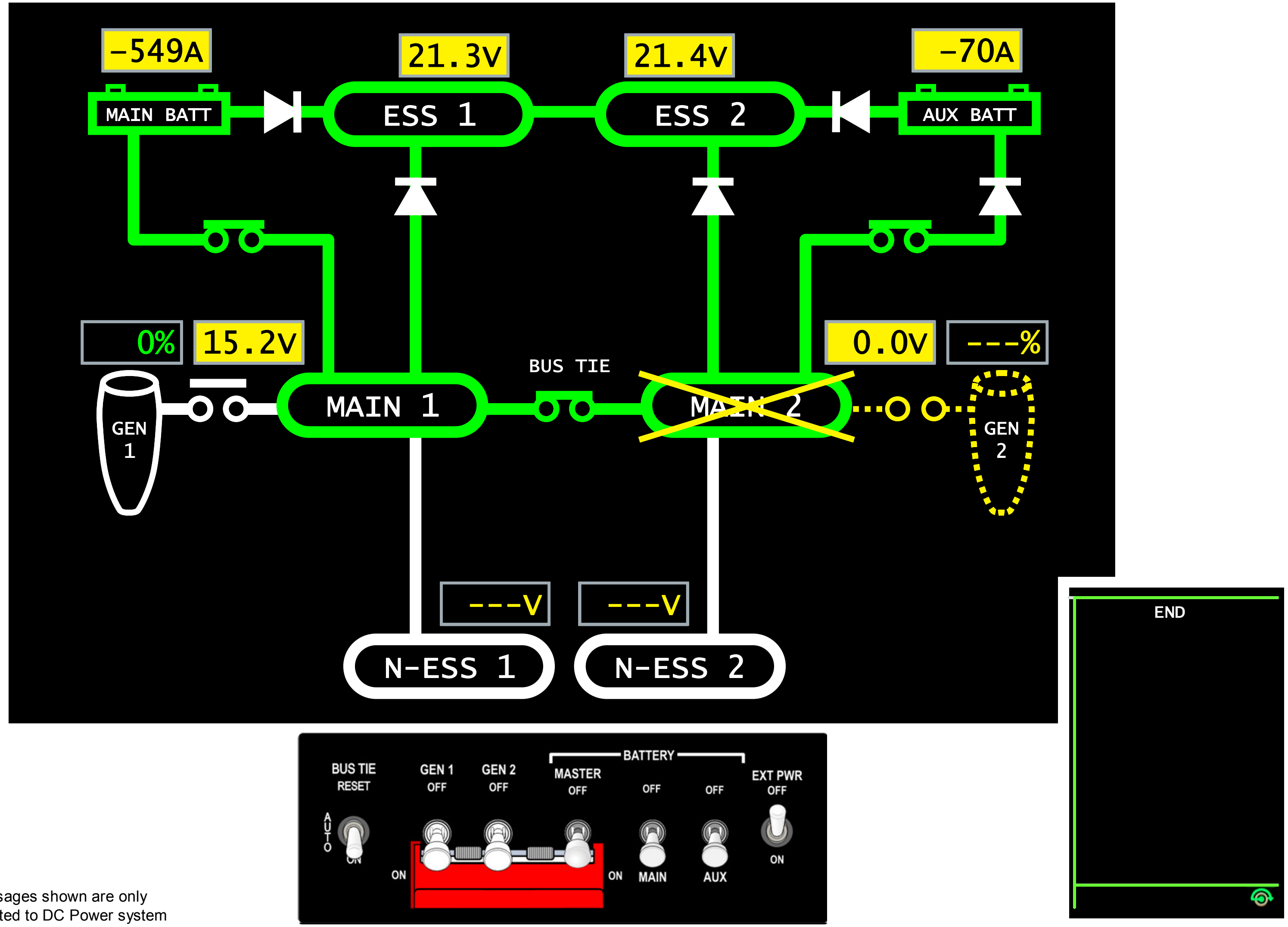






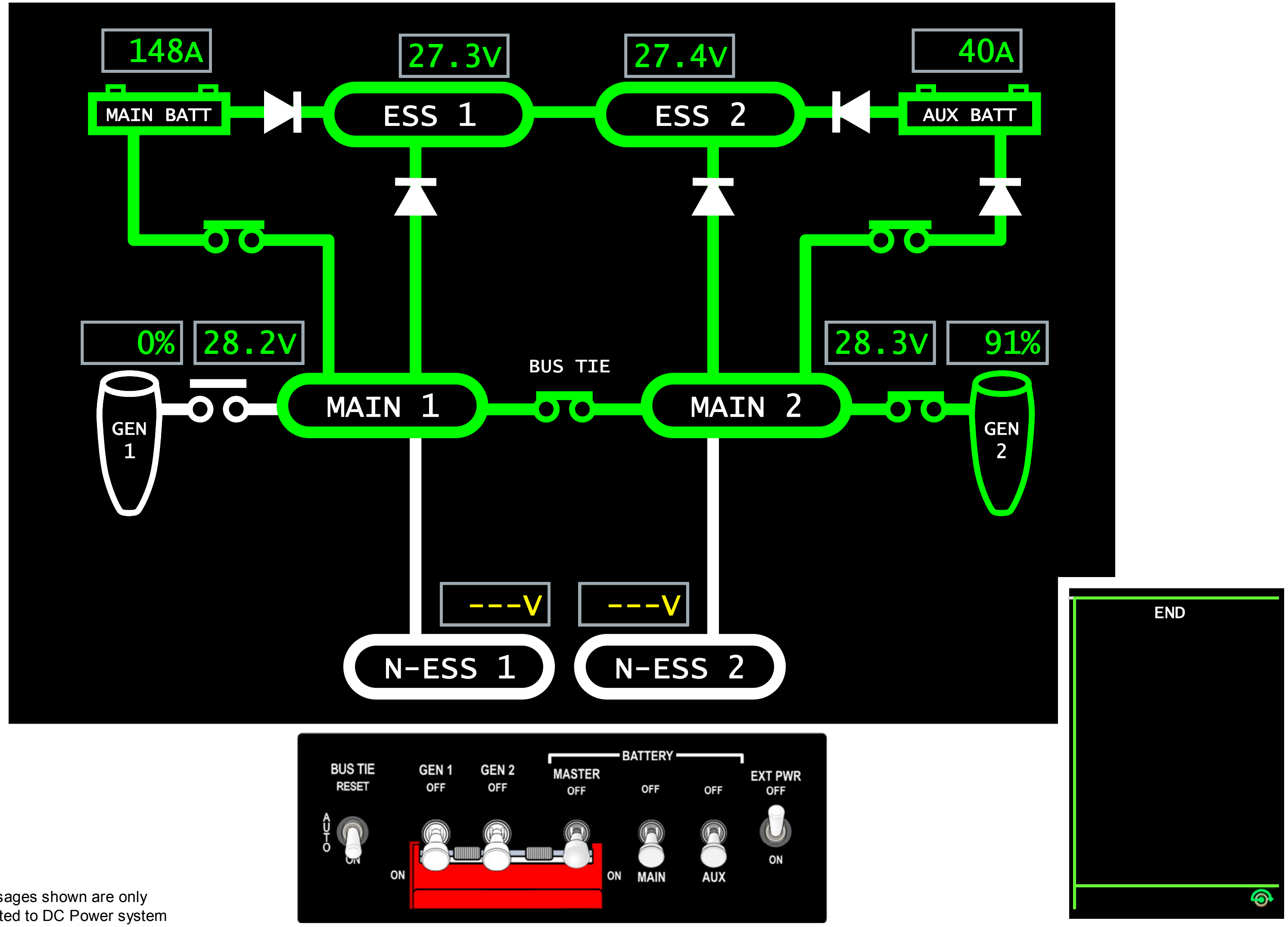
Note: CAS Messages shown are only
those related to DC Power system

DC POWER SYNOPTIC – BATT + GEN + BUS TIE = ON



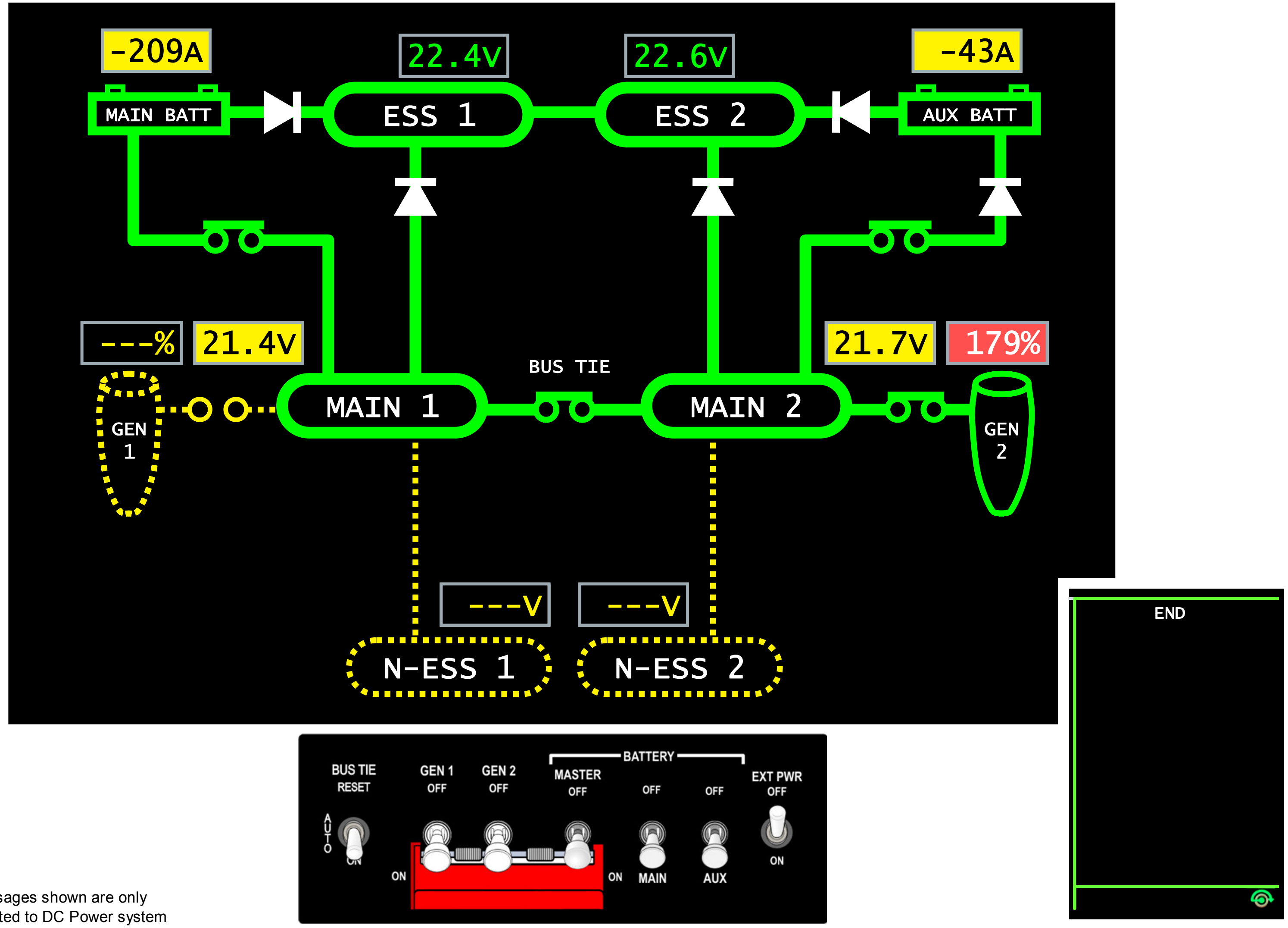
Note: CAS Messages shown are only those related to DC Power system

DC POWER SYNOPTIC – BATT STARTING (ENGINE NO.2)



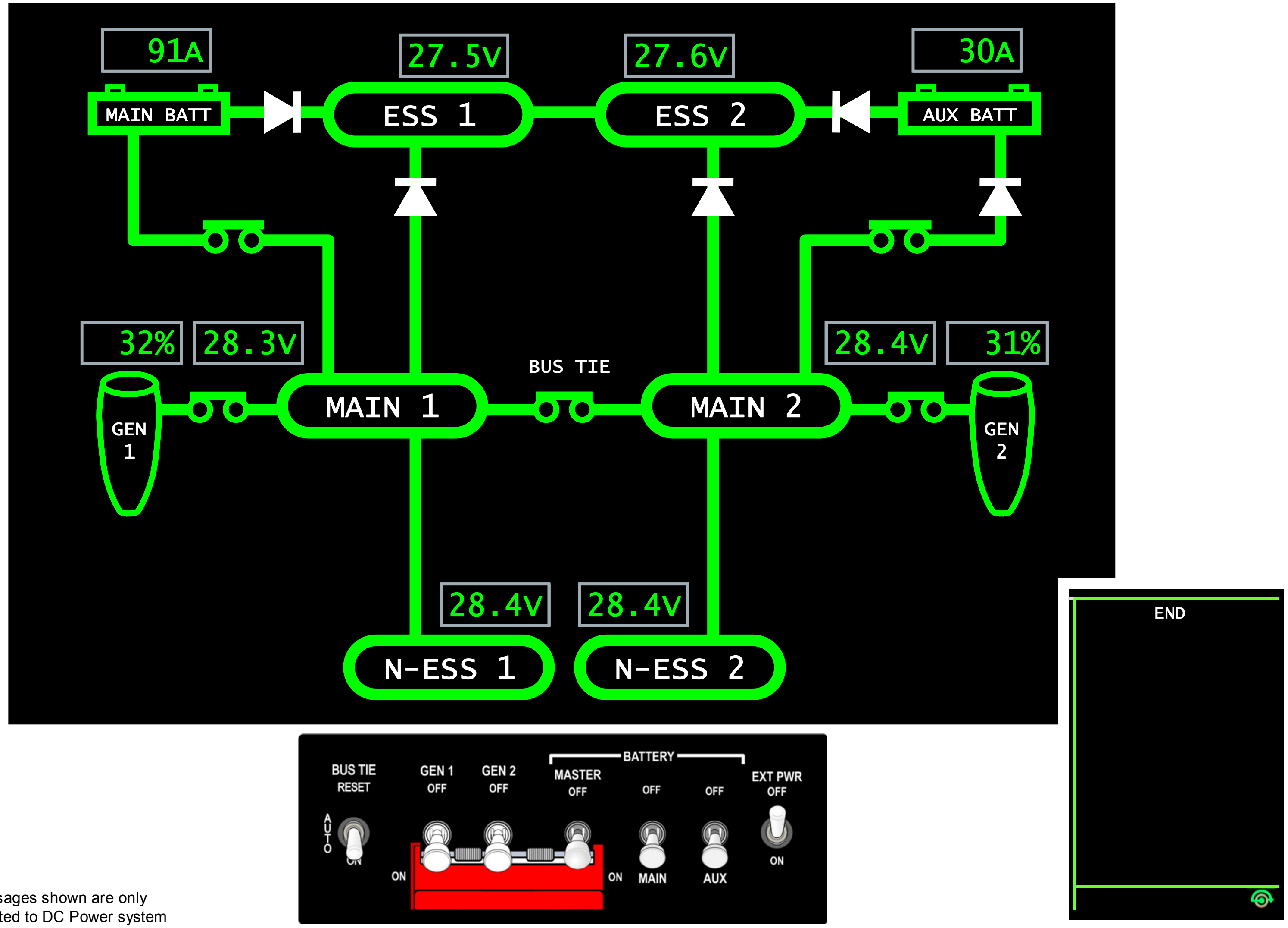
Note: CAS Messages shown are only those related to DC Power system

DC POWER SYNOPTIC – ENGINE NO.2 RUNNING



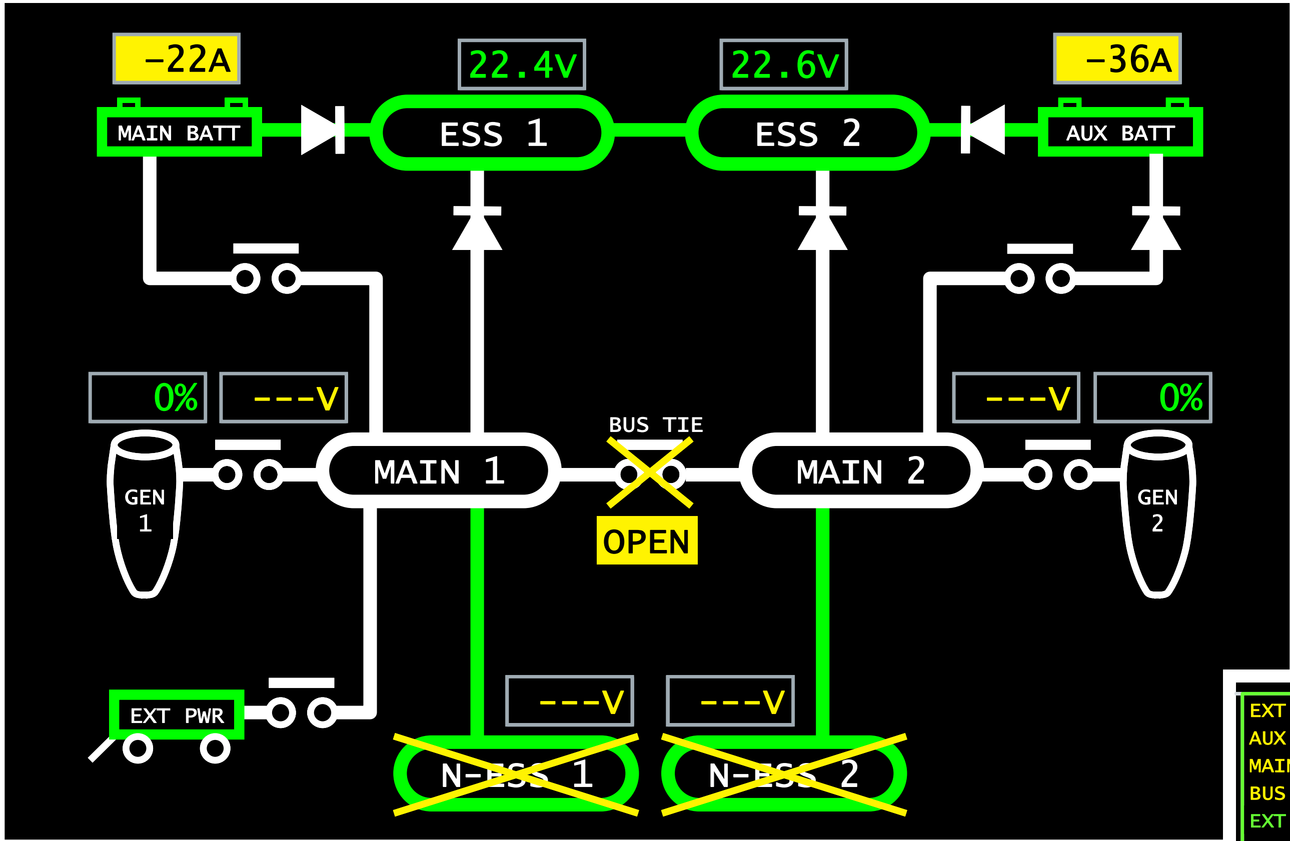
Note: CAS Messages shown are only
those related to DC Power system

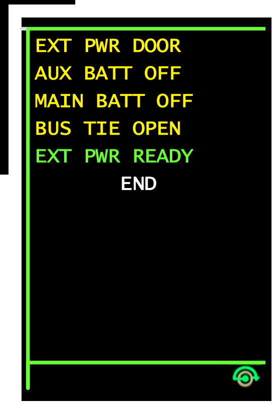
DC POWER SYNOPTIC – ENGINE NO.2 RUNNING

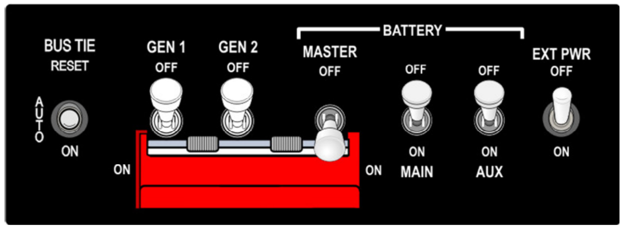


Note: CAS Messages shown are only
those related to DC Power system

DC POWER SYNOPTIC – BOTH ENGINES RUNNING (BUS TIE = ON)

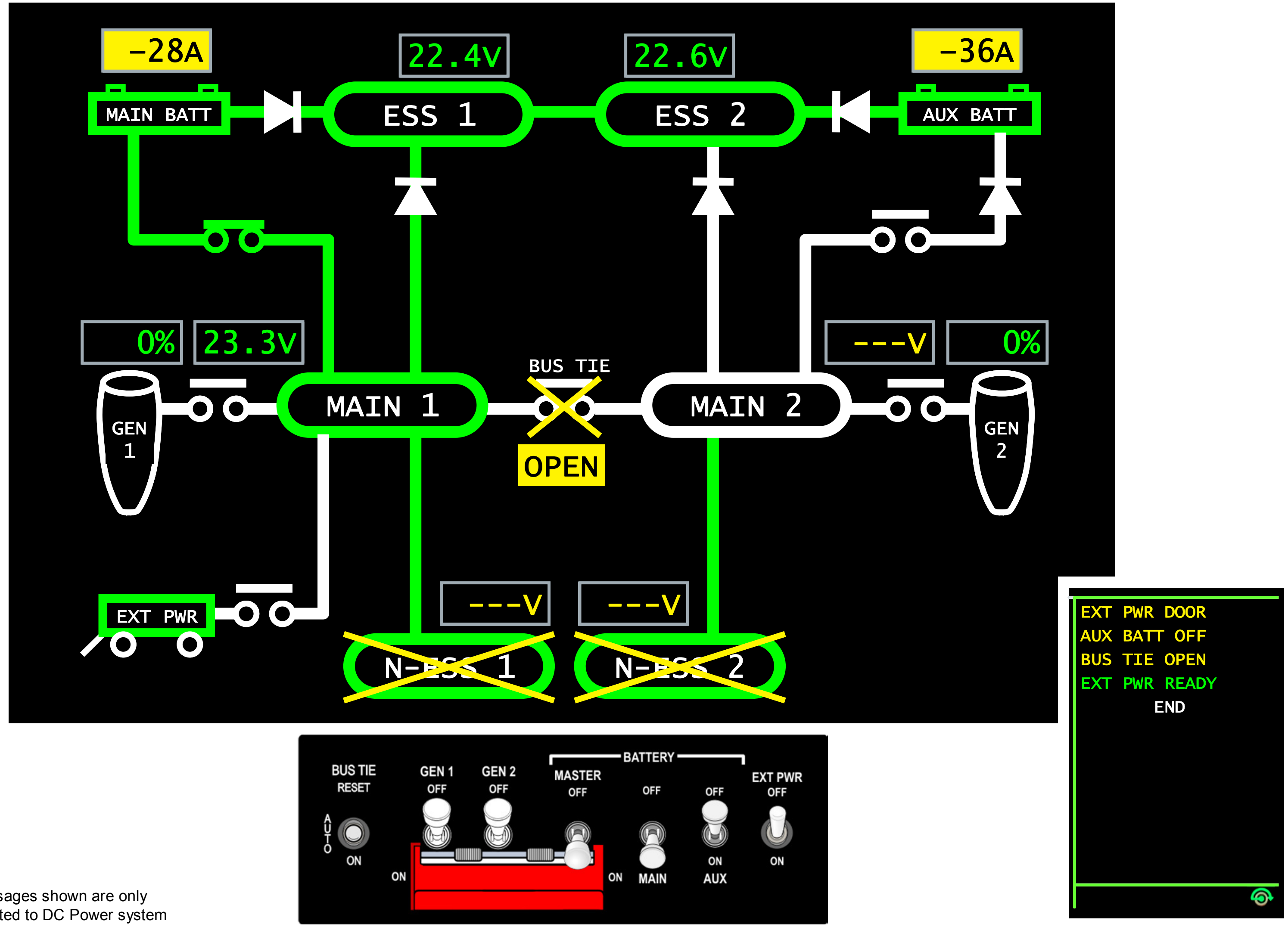






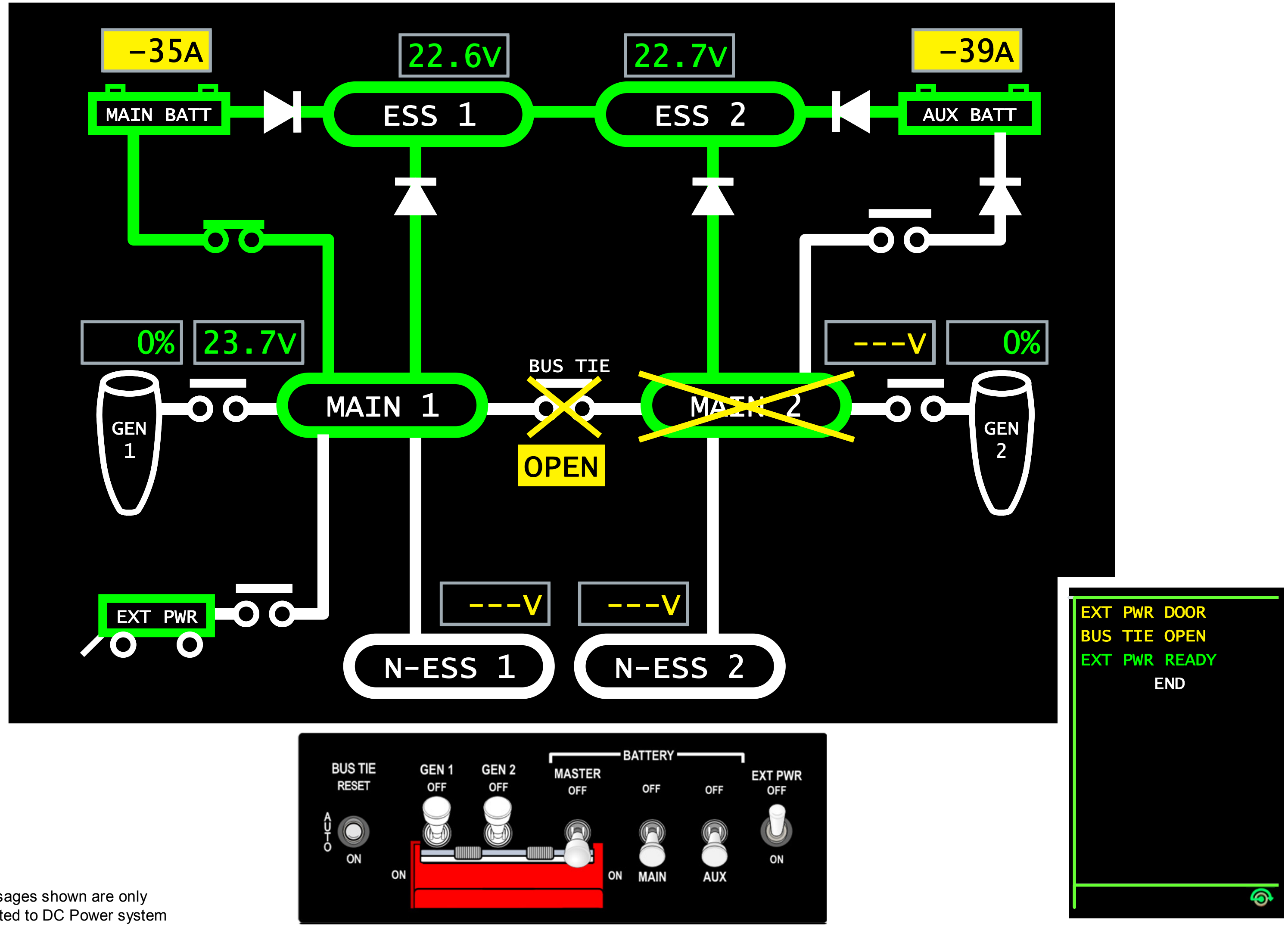
Note: CAS Messages shown are only
those related to DC Power system

DC POWER SYNOPTIC – BATT MASTER = ON, EXT PWR = READY



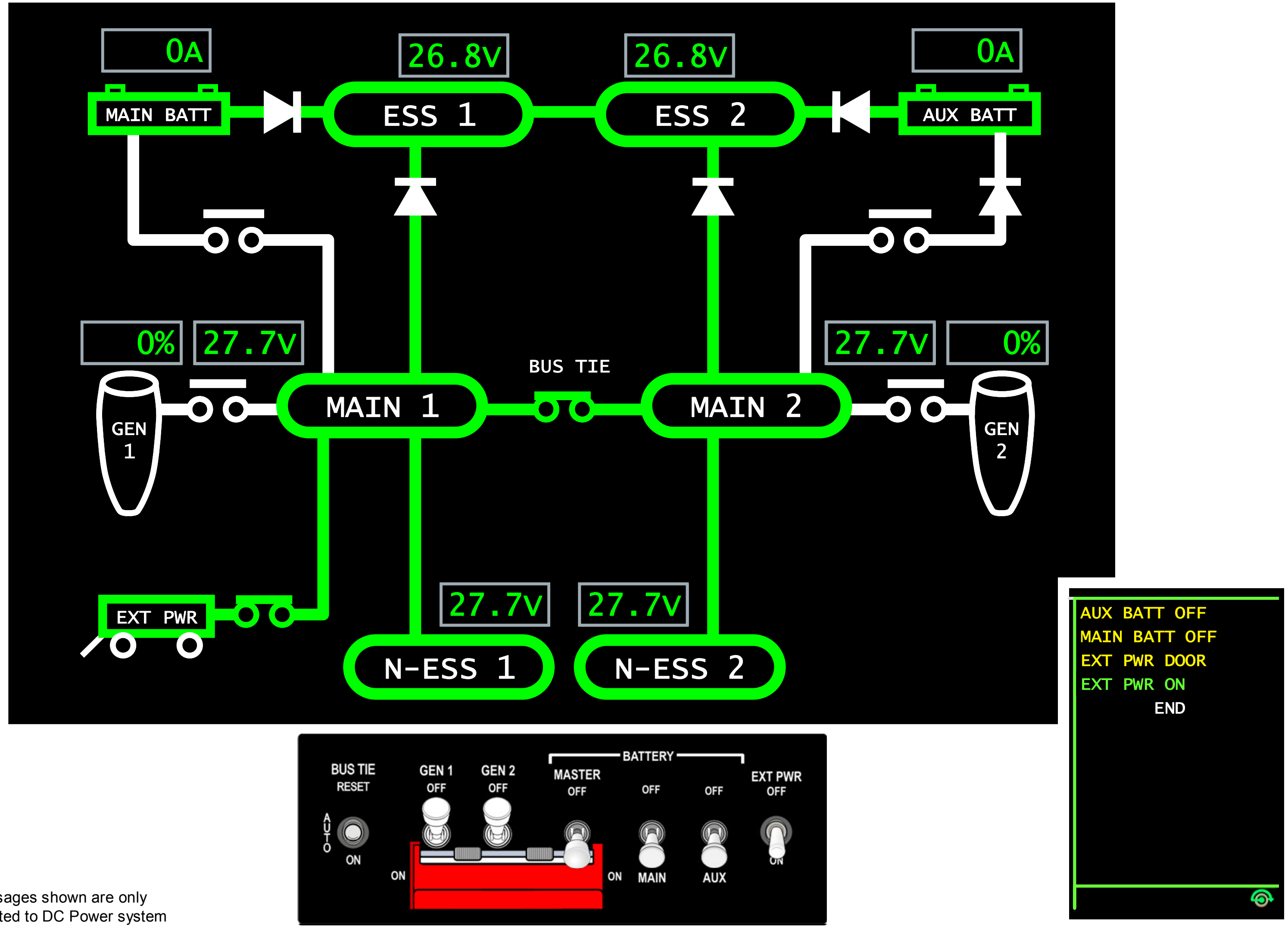
Note: CAS Messages shown are only those related to DC Power system

DC POWER SYNOPTIC – BATT MASTER + MAIN = ON, EXT PWR = READY



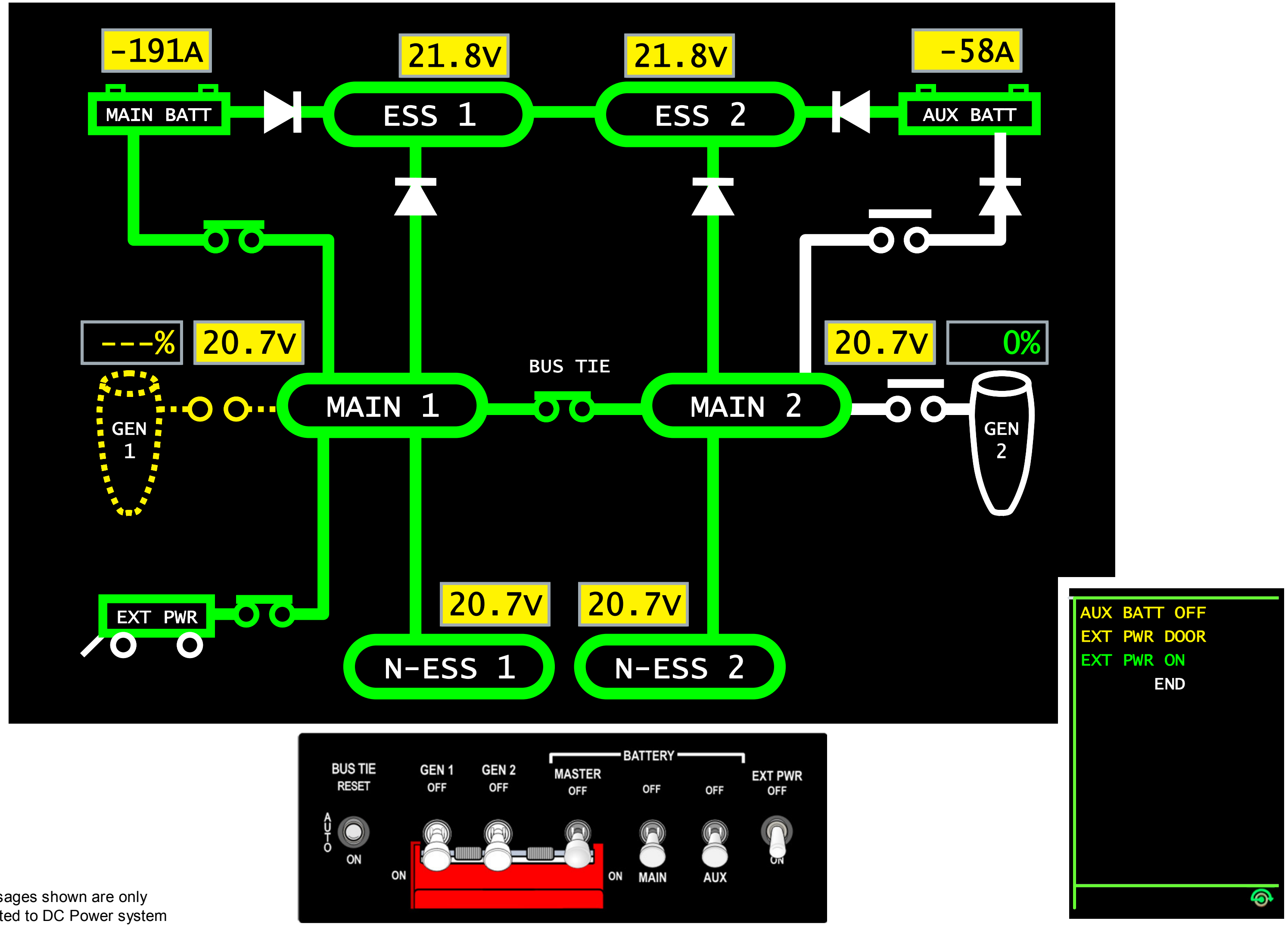
Note: CAS Messages shown are only those related to DC Power system

DC POWER SYNOPTIC – BATT MASTER + MAIN + AUX = ON, EXT PWR = READY



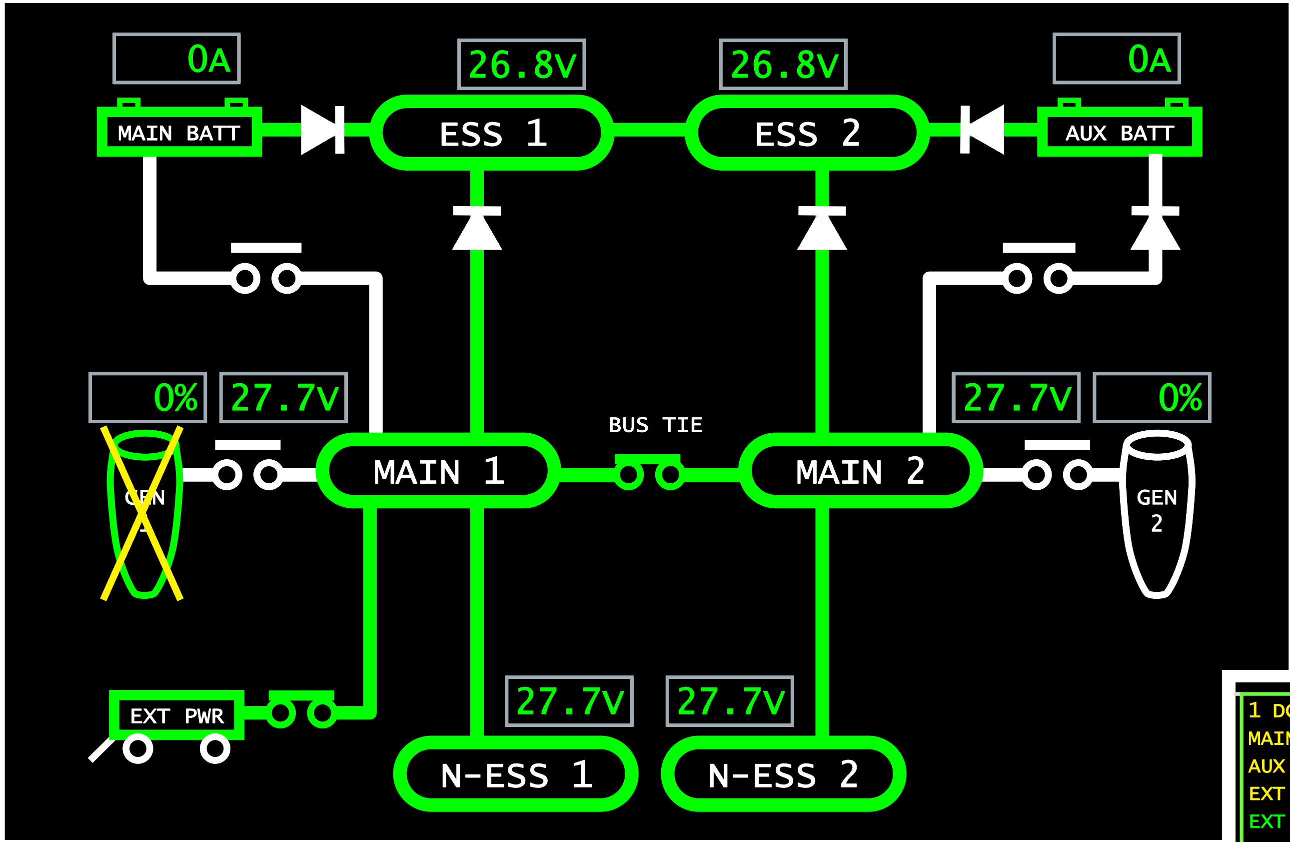
Note: CAS Messages shown are only those related to DC Power system

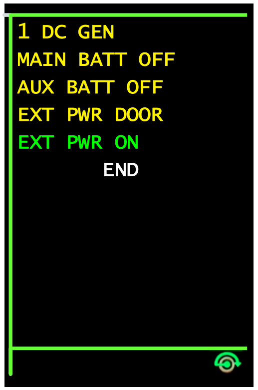
DC POWER SYNOPTIC – BATT + EXT PWR = ON

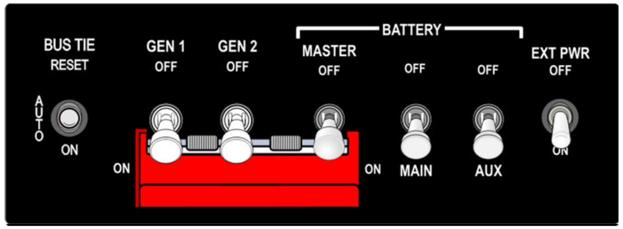


Note: CAS Messages shown are only
those related to DC Power system

DC POWER SYNOPTIC – EXT PWR STARTING (ENGINE NO.1)

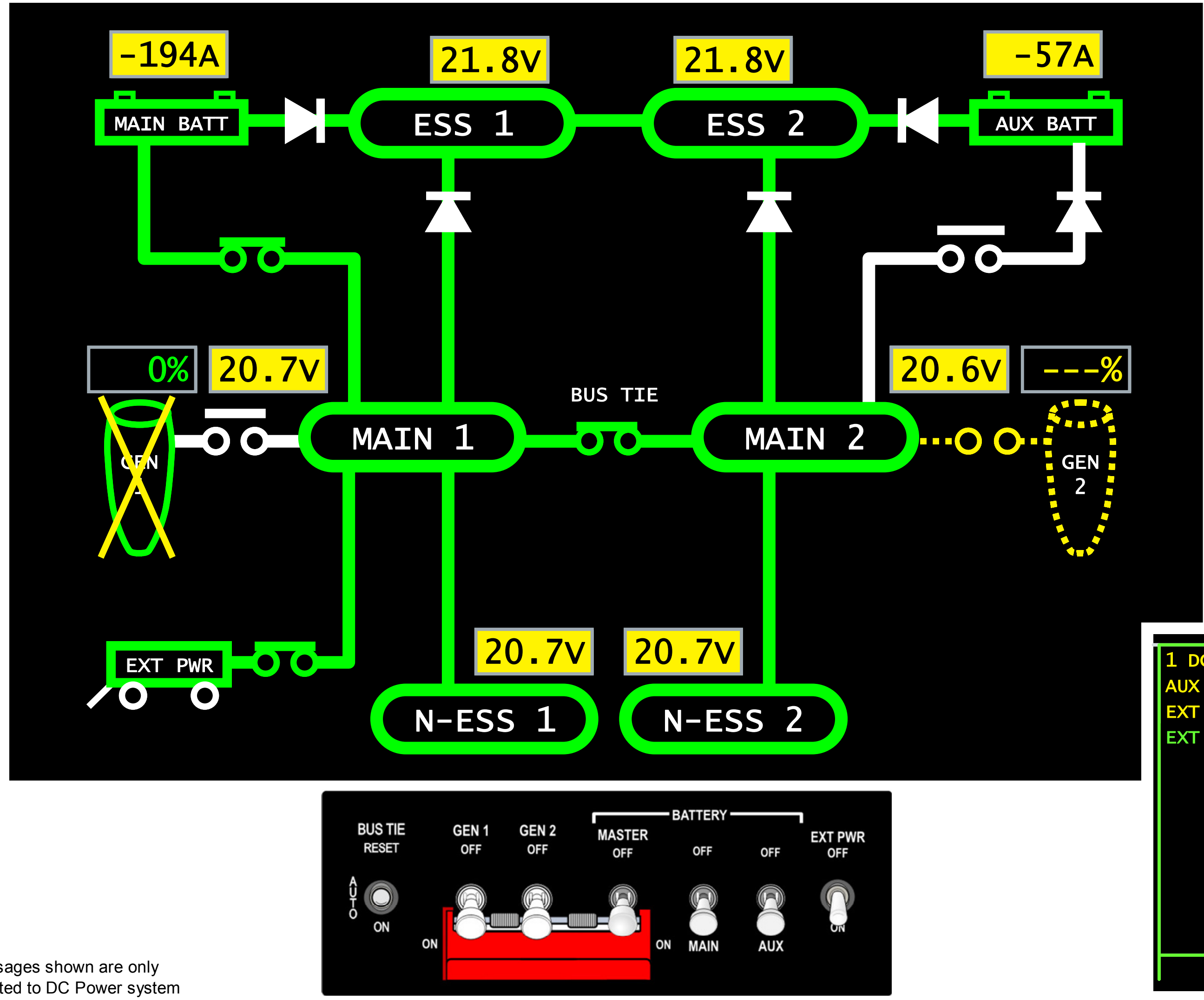


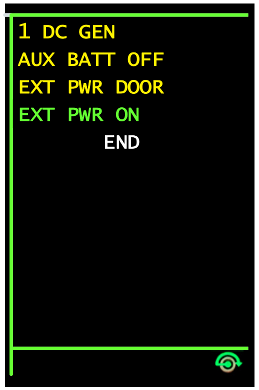




Note: CAS Messages shown are only those related to DC Power system

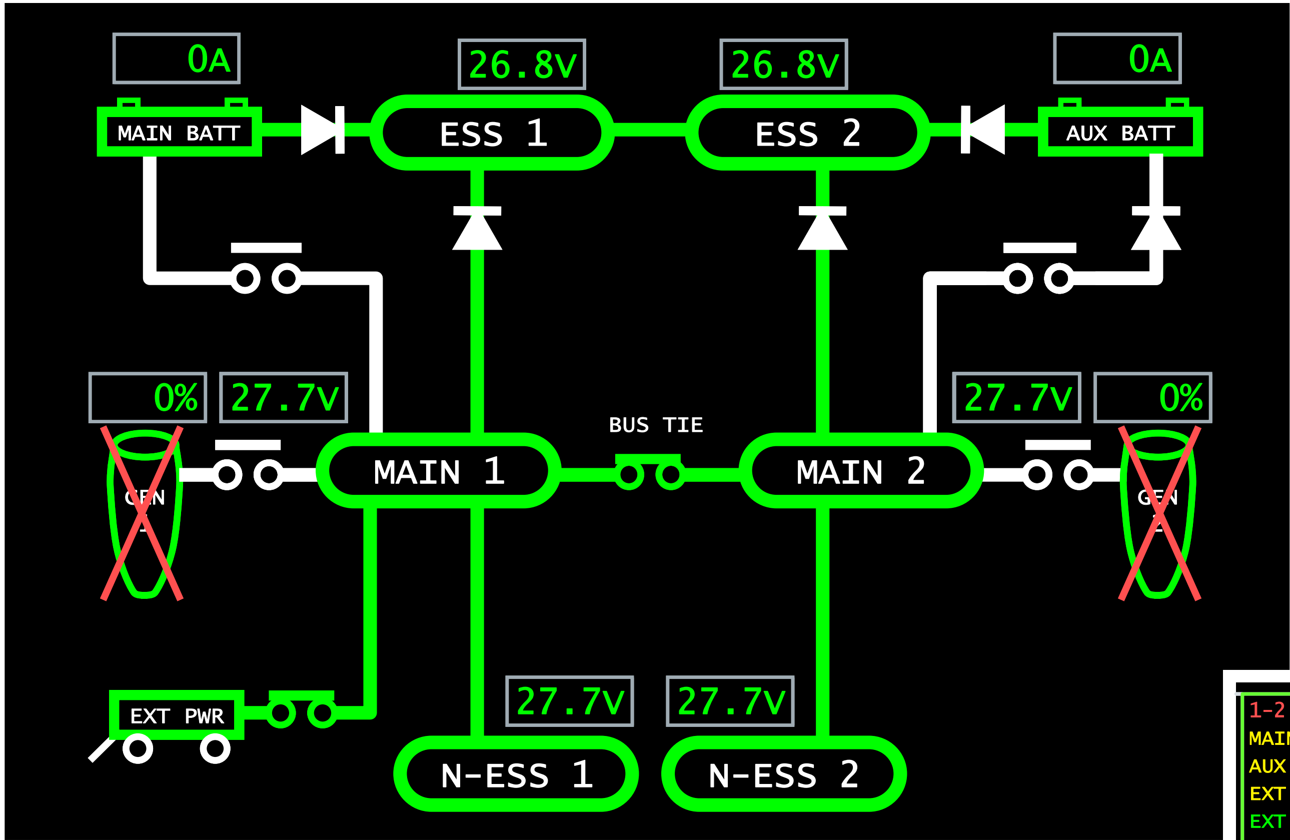
DC POWER SYNOPTIC – EXT PWR = ON, ENGINE NO.1 = RUNNING

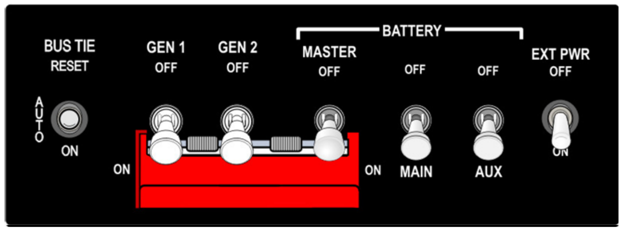


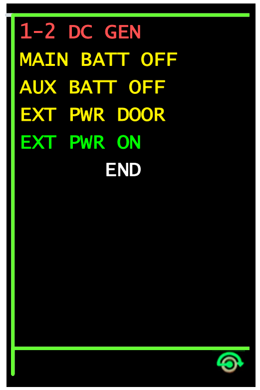


Note: CAS Messages shown are only those related to DC Power system

DC POWER SYNOPTIC – EXT PWR STARTING (ENGINE NO.2)

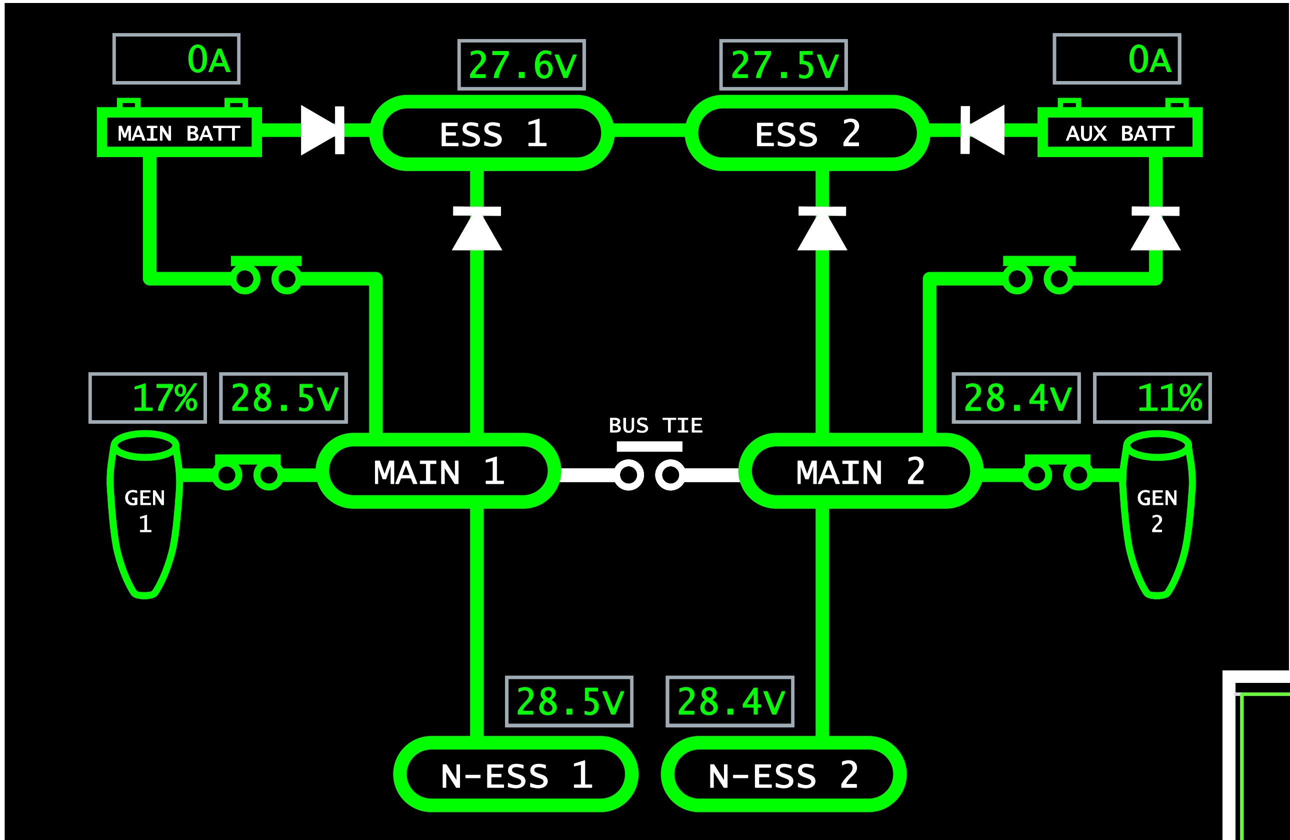


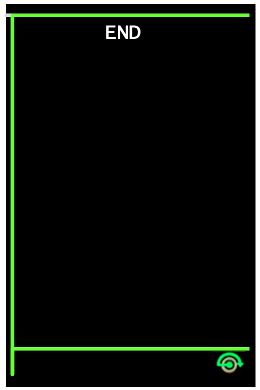


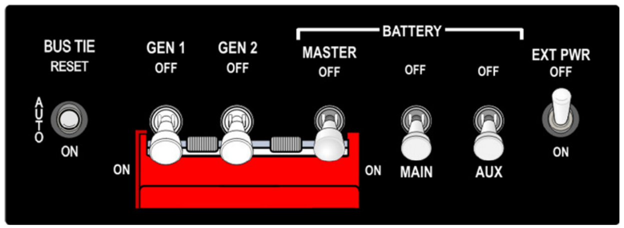


Note: CAS Messages shown are only
those related to DC Power system

DC POWER SYNOPTIC – EXT PWR = ON, BOTH ENGINES RUNNING

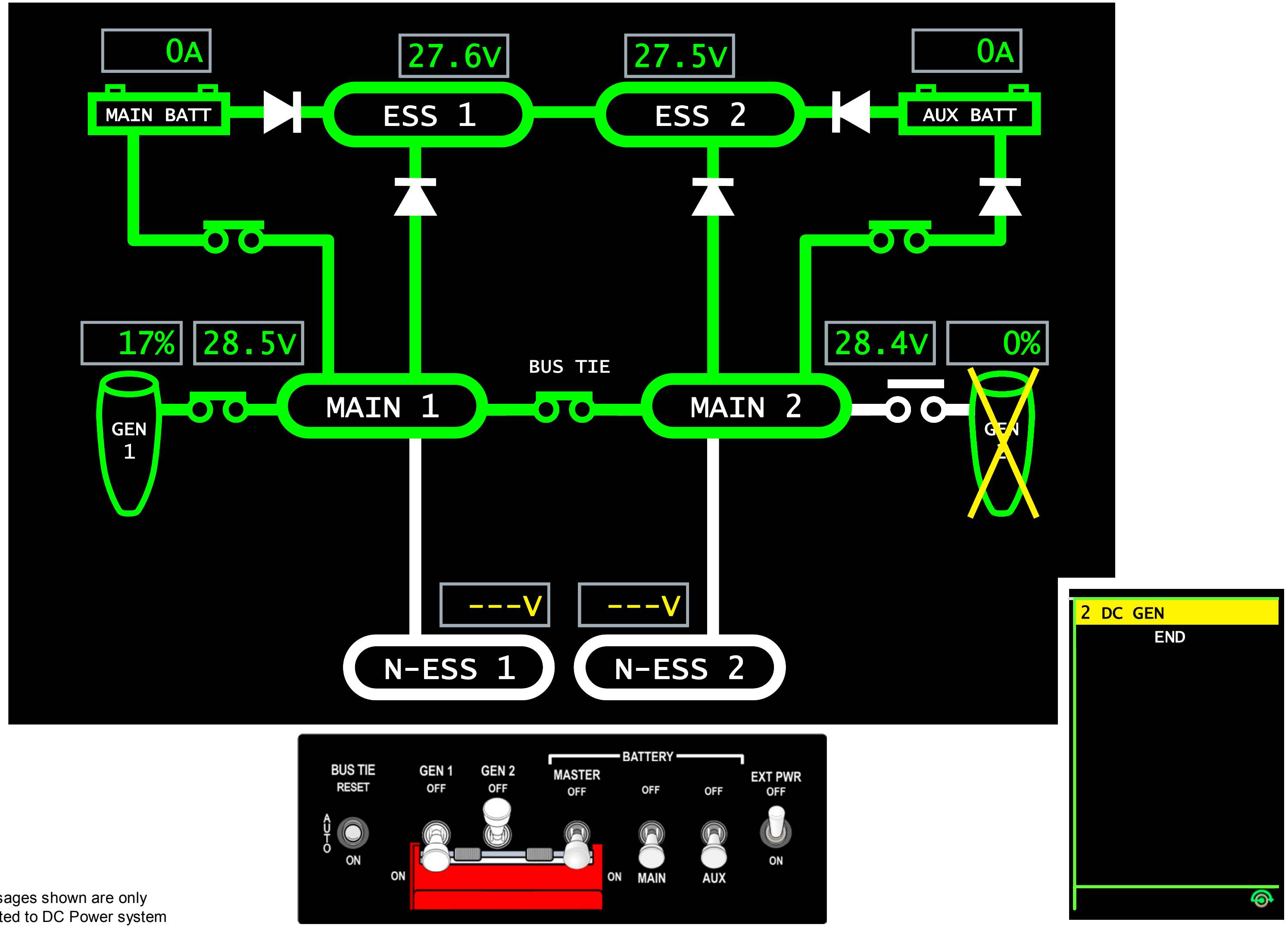






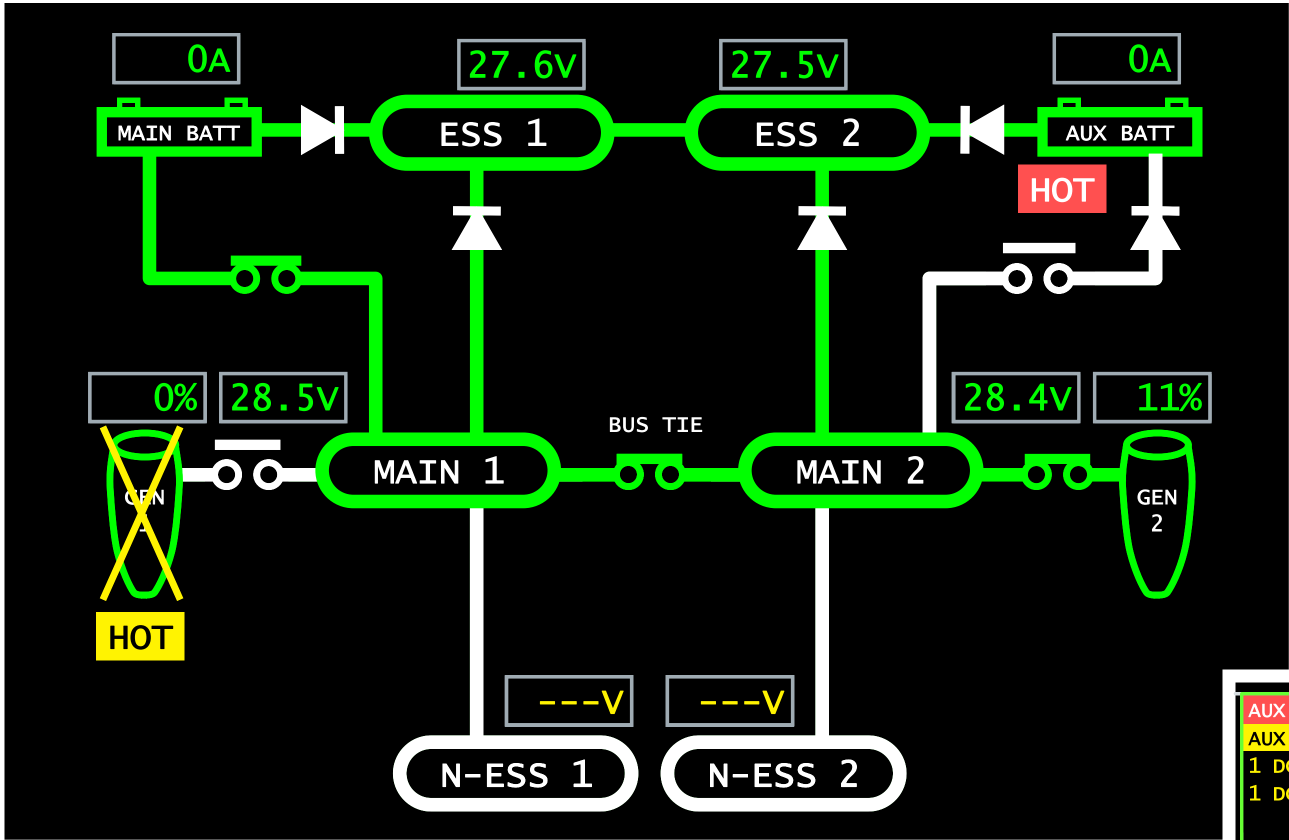
Note: CAS Messages shown are only those related to DC Power system

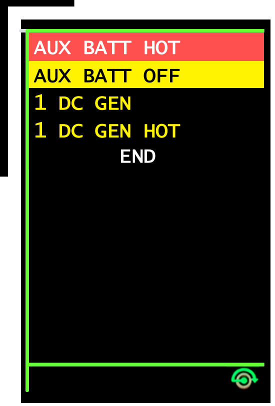
DC POWER SYNOPTIC – NORMAL IN-FLIGHT OPERATION



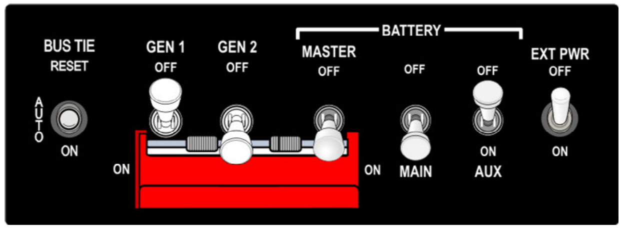
Note: CAS Messages shown are only those related to DC Power system

DC POWER SYNOPTIC – GEN 2 = FAIL

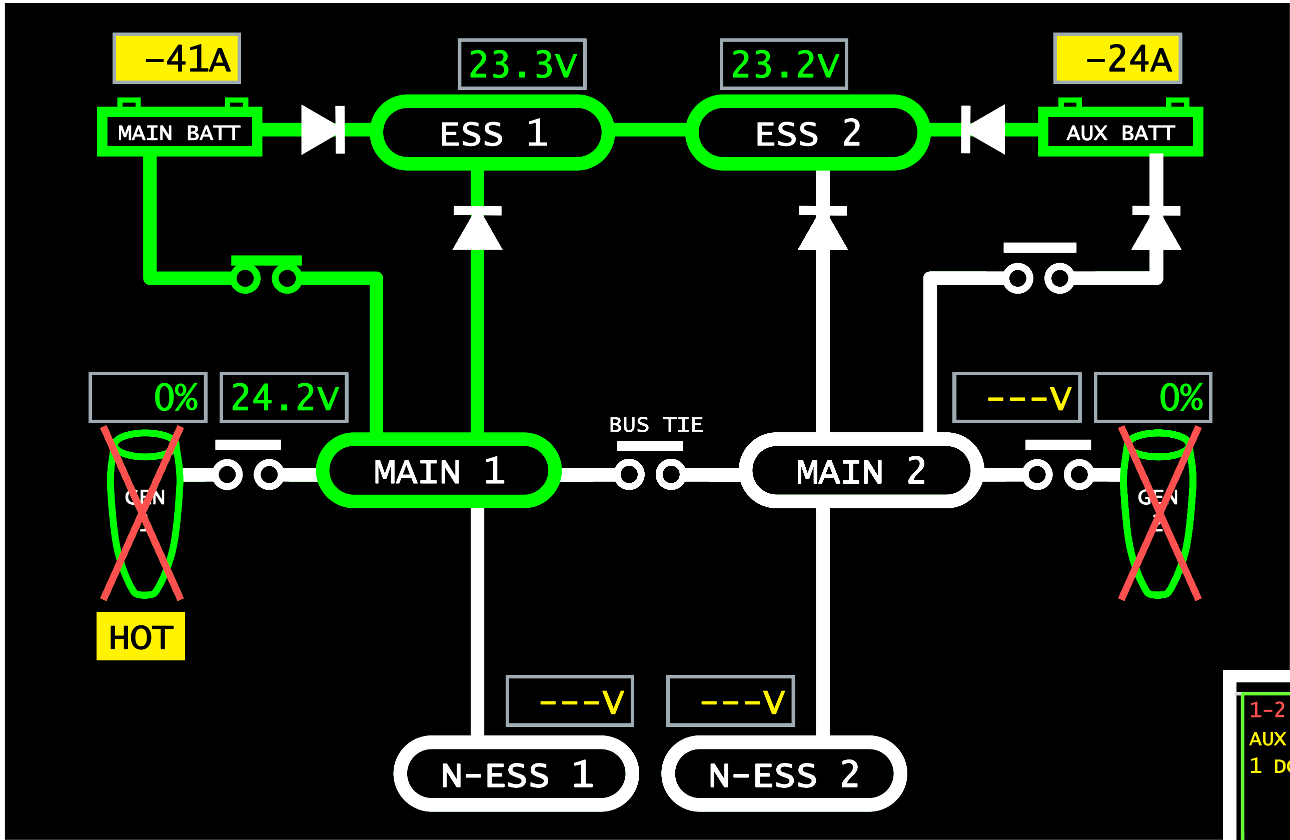


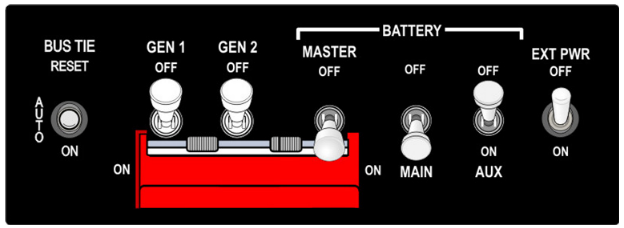


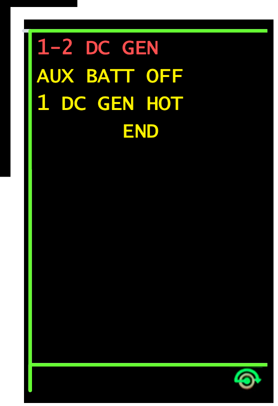
Note: CAS Messages shown are only
those related to DC Power system



DC POWER SYNOPTIC – GEN 1 = HOT & OFF; AUX BATT = HOT & OFF

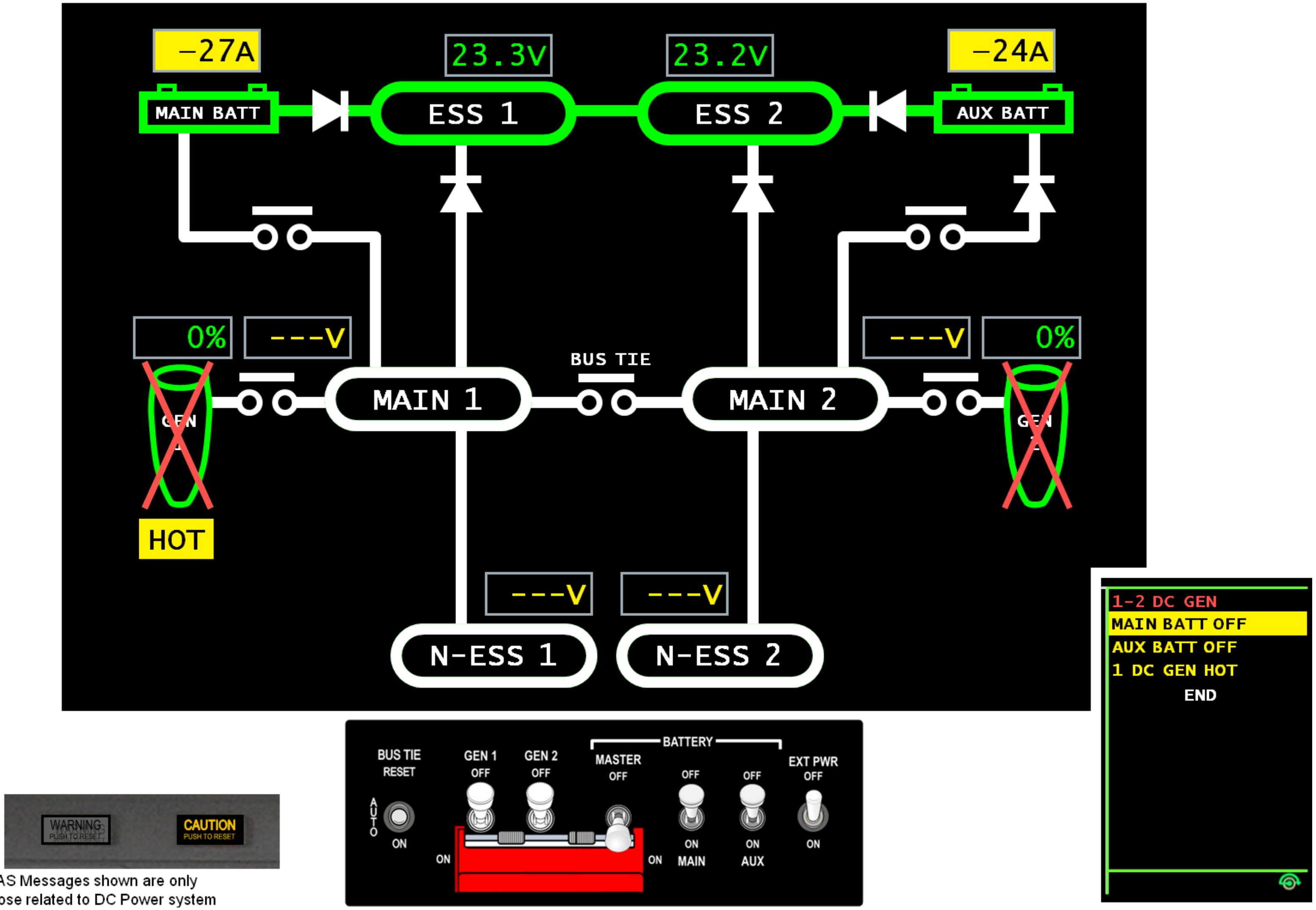






Note: CAS Messages shown are only
those related to DC Power system

DC POWER SYNOPTIC – DUAL GEN FAIL



Note: CAS Messages shown are only those related to DC Power system

DC POWER SYNOPTIC – DUAL GEN FAIL, MAIN BATT = OFF

CAS WARNING MESSAGES

<table>
<tr><th>CAS CAPTION</th><th>FAILURE DESCRIPTION</th><th>PROCEDURE NAME</th><th>AW139-RFM-4D</th></tr>
<tr><td>1-2 DC GEN</td><td>Failure of both generators in flight
– voice warning: WARNING-WARNING repeated once</td><td>DOUBLE GENERATOR FAILURE
EXTENDED FLIGHT ENDURANCE AFTER DOUBLE DC GENERATOR FAILURE</td><td rowspan="4">Section 3
EMERGENCY AND MALFUNCTION PROCEDURES
ELECTRICAL SYSTEM</td></tr>
<tr><td>MAIN BATT HOT</td><td>Main battery overheating
– voice warning: WARNING-WARNING repeated once</td><td rowspan="3">MAIN AND AUXILIARY BATTERY HOT</td></tr>
<tr><td>AUX BATT HOT</td><td>Auxiliary battery overheating
– voice warning: WARNING-WARNING repeated once</td></tr>
<tr><td>AUX-MAIN BATT HOT</td><td>Main and Auxiliary battery overheating
– voice warning: WARNING-WARNING repeated once</td></tr>
</table>

CAS WARNING MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
1 (2) DC GEN	Associated generator failure	SINGLE GENERATOR FAILURE	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES ELECTRICAL SYSTEM
1 (2) DC GEN HOT	Associated generator overheat	DC GENERATOR OVERHEAT	
MAIN BATT OFF	Failure of MAIN battery to MAIN BUS 1	MAIN BATTERY OFF	
AUX BATT OFF	Failure of AUX battery to MAIN BUS 2	AUXILIARY BATTERY OFF	
BUS TIE OPEN	BUS TIE open with one or both generators off-line	BUS TIE OPEN	
BATT OFF LINE	Failure of MAIN and/or AUX battery connection to ESS BUS	LOSS OF MAIN AND/OR AUXILIARY BATTERY SUPPLY	
DC BUS FAIL	DC MAIN BUS 1 and/or 2 fault detected (overcurrent detected by either GCU)	DC MAIN BUS FAILURE	
EXT PWR DOOR	External power socket door not closed	EXTERNAL POWER SOCKET DOOR OPEN	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES MISCELLANEOUS

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
1 (2) MAU OVHT	Associated MAU overheat If MAU1 fails: Electrical parameters not valid (amber dashed): ESS BUS 1 VOLT, MAIN BUS 2 VOLT, NON ESS BUS 1, DC GEN 1 AMP, AUX BATTERY AMP CAS Cautions NOT available: MAIN BATT OFF, EXT PWR DOOR CAS Advisory NOT available: EXT PWR READY If MAU2 fails: Electrical parameters not valid (amber dashed): MAIN BUS 1 VOLT, ESS BUS 2 VOLT, DC GEN 2 AMP, NON ESS BUS 2, MAIN BATTERY AMP CAS Cautions NOT available: AUX BATT OFF CAS Advisory NOT available: EXT PWR ON In case of MAU 1(2) failure, do not use electrical synoptic page information.	MODULAR AVIONICS UNIT OVERHEAT / FAIL	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES AVIONICS

CAS ADVISORY MESSAGES

CAS CAPTION	MESSAGE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
EXT PWR READY	External power voltage available at receptacle but external power not connected to MAIN BUS		Section 2 NORMAL PROCEDURES ADVISORY CAPTION DEFINITIONS
EXT PWR ON	External power connected to MAIN BUS		

ELECTRICAL SYSTEM LIMITATIONS

Refer to AW139-RFM-4D Section 1 – Limitations

In case of 1 (2) MAU failure, do not use electrical synoptic page information

PLACARDS

Since generators are air cooled, the maximum continuous DC GEN load changes with altitude as stated by the placard located on the instrument panel (pilot side).

SERVICES AVAILABLE ON ESSENTIAL BUS 1 AND 2

DC ESS BUS 1

BAGGAGE FIRE DETECT
BUS CONTROL 1
EAPS 1
EEC 1
EEC 1 FAIL INDICATION
EMERG LIGHT
EMERG FLOAT CONTROL (MANUAL)
EMERG FLOAT POWER
ENG 1 FIRE DETECTION
ENG 1 FIRE EXTINGUISHER
ENG 1 IGNITER & START
FCU 1
FDR/CVR
FUEL PUMP 1
FUEL SHUT OFF VALVE 1
FORCE TRIM
GCU 2
ENG GOV CONTROL
HYD SYS 1 & SOV 1&2
ICS PLT (PLT ICS IN BKUP MODE)
LANDING LIGHT CONTROL
LANDING LIGHT POWER
LDG GEAR CONTROL
LDG GEAR INDICATION
MAIN BATT CHARGE
MAU 2 (PRI POWER)
MRC 1 (VHF 1)
MRC 2 (XPDR)
NLG STEERING LOCK
PLT MFD
STBY 1 ATT
TAIL POS LIGHT
TRANS CHIP BURNER

DC ESS BUS 2

ADM 2
AUTO FLOAT CONTROL
AUX BATT CHARGE
BUS CONTROL 2
CARGO HOOK EMERG RELEASE
CLOCK PLT
COCKPIT PLT LIGHT
EAPS 2
EEC2
EEC 2 FAIL INDICATION
ELT
ENG 2 FIRE DETECTION
ENG 2 FIRE EXTINGUISHER
ENG 2 IGNITER & START
EXT SPKR CONTROL
FADEC GSE
FCU 2
FUEL PUMP 2
FUEL SHUT OFF VALVE 2
FUEL X FEED
GCU 1
HOIST CUTTER 2
HOIST ICS OPERATOR
HYD SYS 2
INSTRUMENT PANEL ILLUM.
LDG GEAR EMERG DOWN
LINEAR ACTUATOR 2
MAU 2 (AUX POWER)
MRC 2 (NIM2, VHF2, NAV2)
PFD PLT CONTROL
PITOT 2 FAIL INDICATION
PITOT 2 HEAT
PLT PFD
RTR BRK

SERVICES LOST DURING BUS FAILURES

DC MAIN BUS 1

ADM 1
ANTI-COLL LIGHT
CARGO HOOK RELEASE
CLOCK CPLT
COCKPIT CPLT LIGHT
CPLT ICS
(CPLT ICS in back up mode)
CPLT PFD
CSL ILLUM
FD1
HOIST POWER
HOIST CUTTER 1
HOIST CONTROL
HUMS
HYD ELEC PUMP
LINEAR ACTUATOR 1
MAU1 (PRI POWER)
MCDU PLT
MRC 1 (NIM 1, NAV 1)
OVHD PANEL ILLUM
PA
PFD CPLT CONTROL
PITOT 1 FAIL INDICATION
PITOT HEAT 1
RAD ALT 1
UTIL POWER
W/RADAR
XMSN OIL LEVEL SENSOR

DC MAIN BUS 2

AUTO TRIM
BAGGAGE COMPT LIGHT
CABIN LIGHT
COCKPIT/CABIN HEATER
COCKPIT VENT (PLT)
CPLT MFD
DOME LIGHT
FD2
HOIST LIGHT
MAU 1 (AUX POWER)
MCDU CPLT
MRC 2 (ADF & DME)
PLT W/WIPER
PSU
RAD ALT 2
SEARCH LIGHT CONTROL
SEARCH LIGHT POWER
STORM LIGHT
SUN LIGHT CONTROL
VENT CONTROL 2
V/UHF

DC NON ESS BUS 1

CABIN VENT
COCKPIT VENT (CPLT)
CPLT W/WIPE
ECS (COCKPIT)
FUEL DRAIN VALVE
STEP LIGHT

DC NON ESS BUS 2

ECS (CABIN)
EXT/SPKR POWER

SERVICES AVAILABLE AND SERVICES LOST

CHAPTER 25

EQUIPMENT/FURNISHINGS

SECTION 60 – ELT SYSTEM

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ELT SYSTEM – GENERAL

The purpose of the Emergency Locator Transmitter (ELT) is to help locating the aircraft after a crash or an emergency landing.

The ELT automatically activates following a crash because of the internal g-switch and transmits the standard tone on VHF and UHF guard frequencies (121.5 MHz and 243.0 MHz) and coded information to the SARSAT system (406.025 MHz).
The 406.025 MHz transmitter sends an encoded digital message containing aircraft identity (ELT serial number) and the last aircraft position as received from the FMS.

ELT SYSTEM – MAIN COMPONENTS

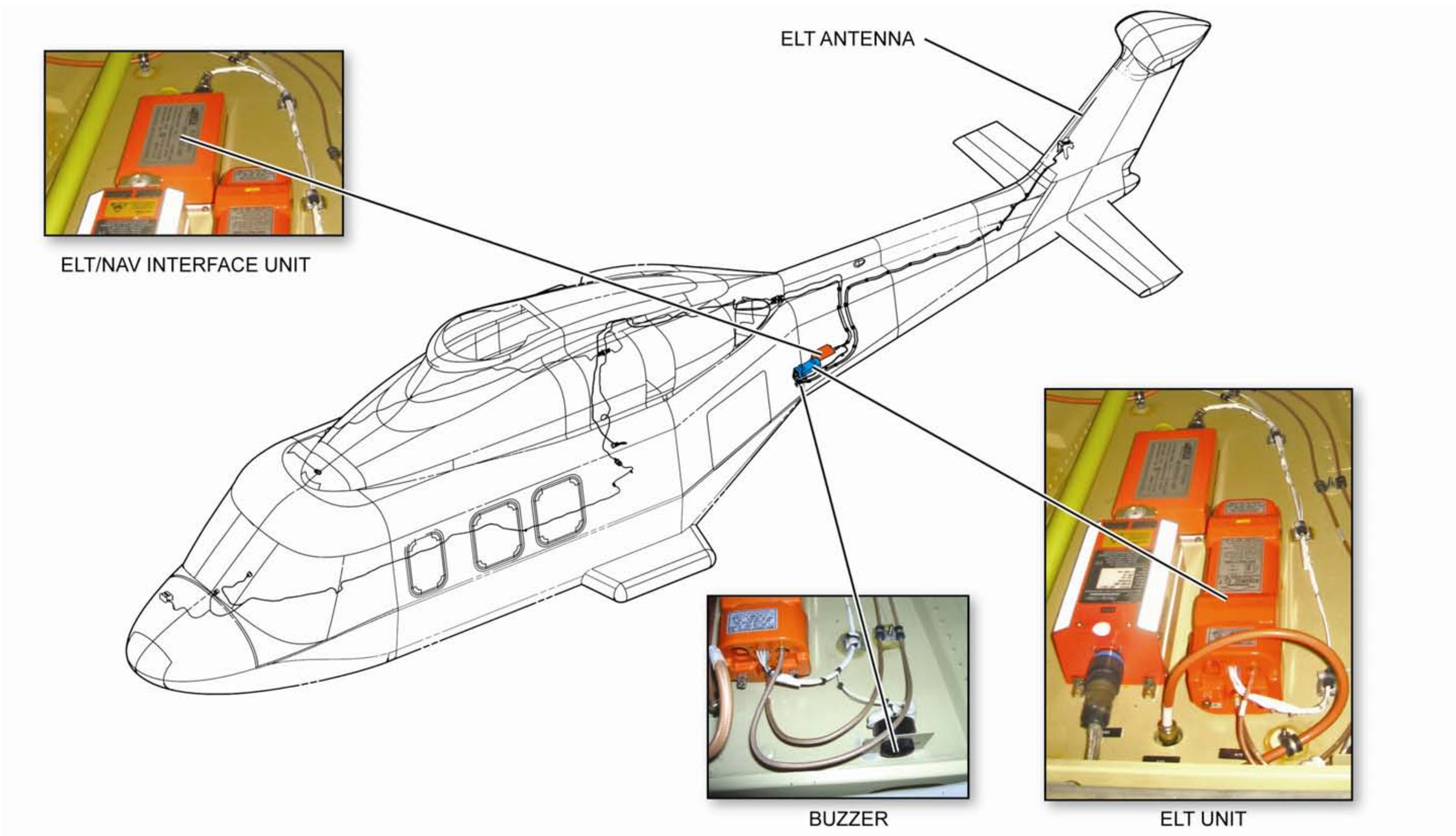
The ELT main components are:

- the ELT unit
- the ELT/NAV Interface Unit which provides the ELT with the FMS position data
- the ELT control panel
- the buzzer which operates whenever the ELT is activated
- the ELT antenna, which has two connectors: one for 121.5/243.0 MHz and one for 406.025 MHz

ELT SYSTEM – PRINCIPLE OF OPERATION

Following an automatic or manual activation, the ELT starts immediately transmitting on both guard frequencies and after 50 seconds it starts transmitting also on 406.025 MHz.
Once activated, the 121.5 MHz and 243.0 MHz transmitter will continue to operate until the battery power is exhausted, which will typically be longer than 48 hours, whilst the 406.025 MHz transmitter will operate for 24 hours before shutting down automatically.

An aural (buzzer) and a visual (red LED) monitor is provided to alert the crew when the ELT is transmitting.
For normal operation the switch on the ELT control panel must be in ARM.



ELT SYSTEM – MAIN COMPONENTS

ELT SYSTEM - CONTROLS AND INDICATORS

1. ELT switch

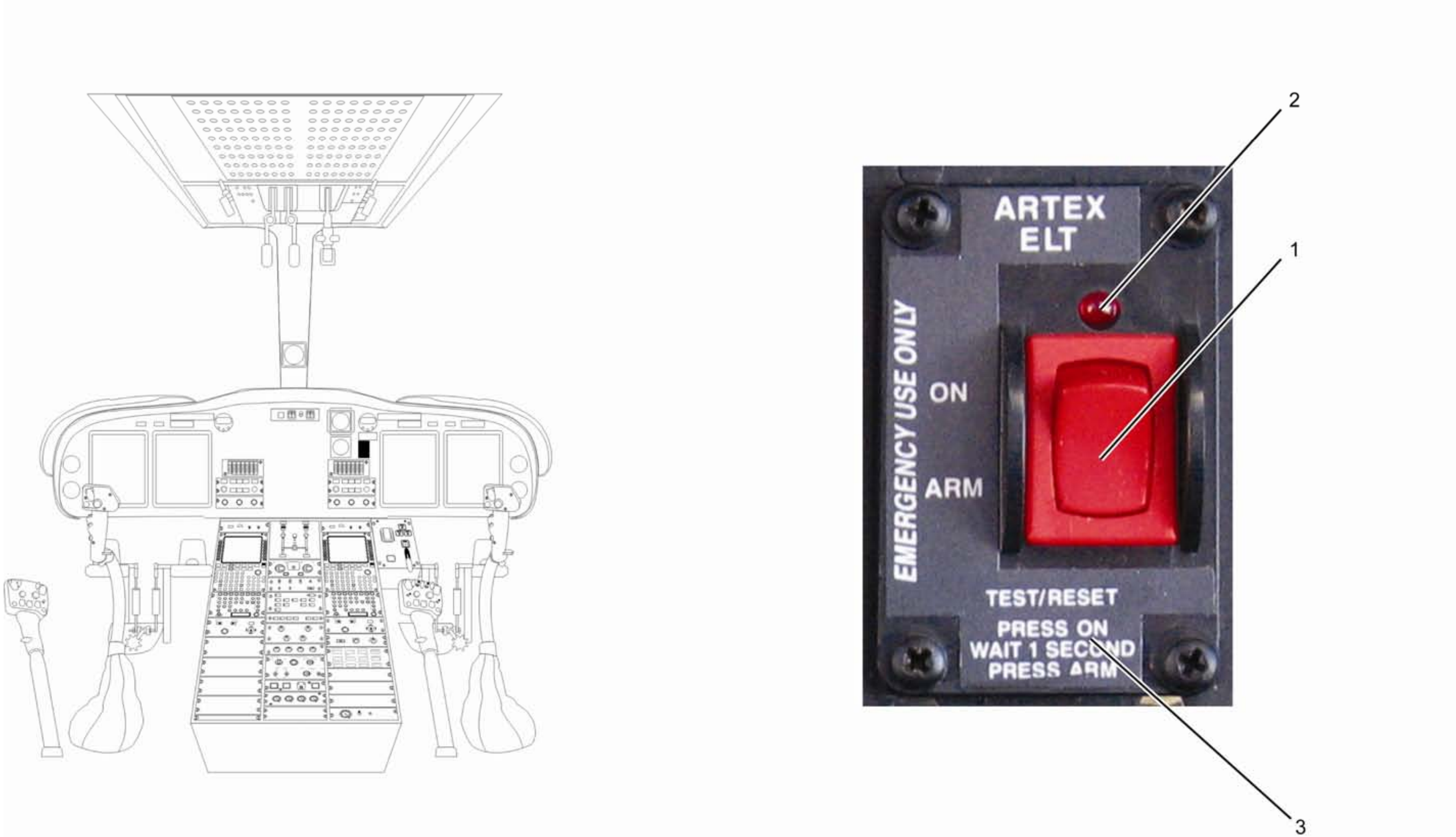
ARM ……...	The ELT automatically activates on impact.
	Moving the switch to ARM turns the transmitter off.
ON ..……….	The ELT is manually activated
	Note: The ON position can be used to test the ELT operation; actual ELT transmission can be monitored using the VHF COM radio

2. ELT light

Red (flashing)	Indicates that the ELT is activated.
	After switching the transmitter off, the light flashes once only. If it flashes more times it indicates internal failure.

3. RESET PROCEDURE

If the ELT is activated accidentally, the switch on the panel must be set to ON and, after 1 second, returned back to ARM.



ELT SYSTEM – CONTROLS AND INDICATORS

CHAPTER 26

FIRE PROTECTION

SECTION 00 – GENERAL

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FIRE PROTECTION – GENERAL

The purpose of the fire protection system is to detect for overheating or fire in the engine compartments and presence of smoke in the baggage compartment.

The fire protection system comprises

- the engine fire detection system
- the engine fire extinguishing system
- the baggage compartment smoke detection

FIRE DETECTION SYSTEM – GENERAL

The purpose of the engine fire detection system is to detect high temperature, fire or hot gas leakage into the engine compartments.
The system comprises a heat fire detector installed in each engine compartment.
When fire or hot gas leak are detected, the indicating system permits to identify the location of the fire.

FIRE DETECTION SYSTEM – MAIN COMPONENTS

HEAT FIRE DETECTOR

The heat detector is made of a continuous firewire element and a responder.

FIREWIRE

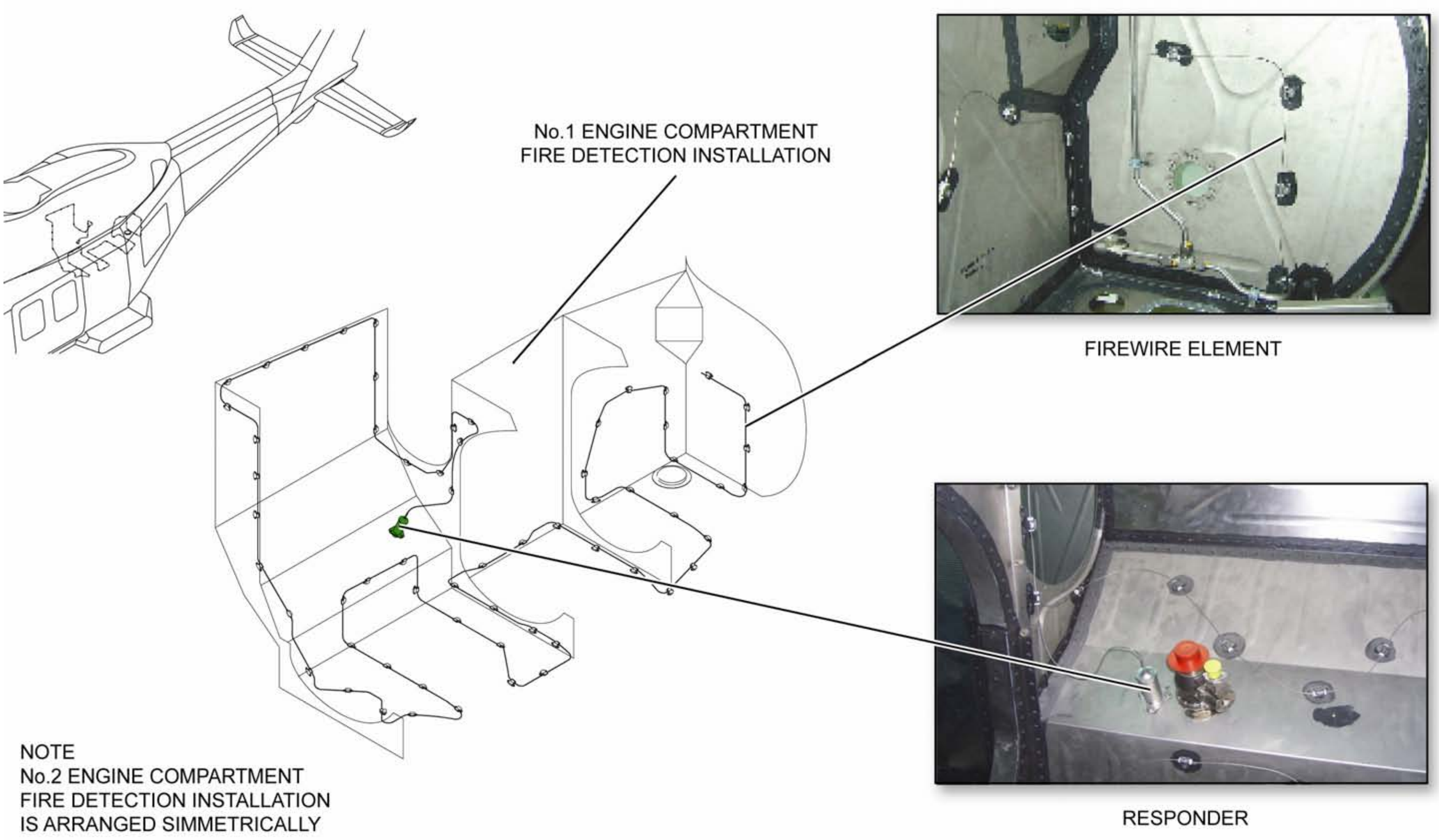
The firewire is the sensing element and consists of:

- a stainless steel tube containing the gas (helium) under pressure;
- a core material (wire) located inside the tube.

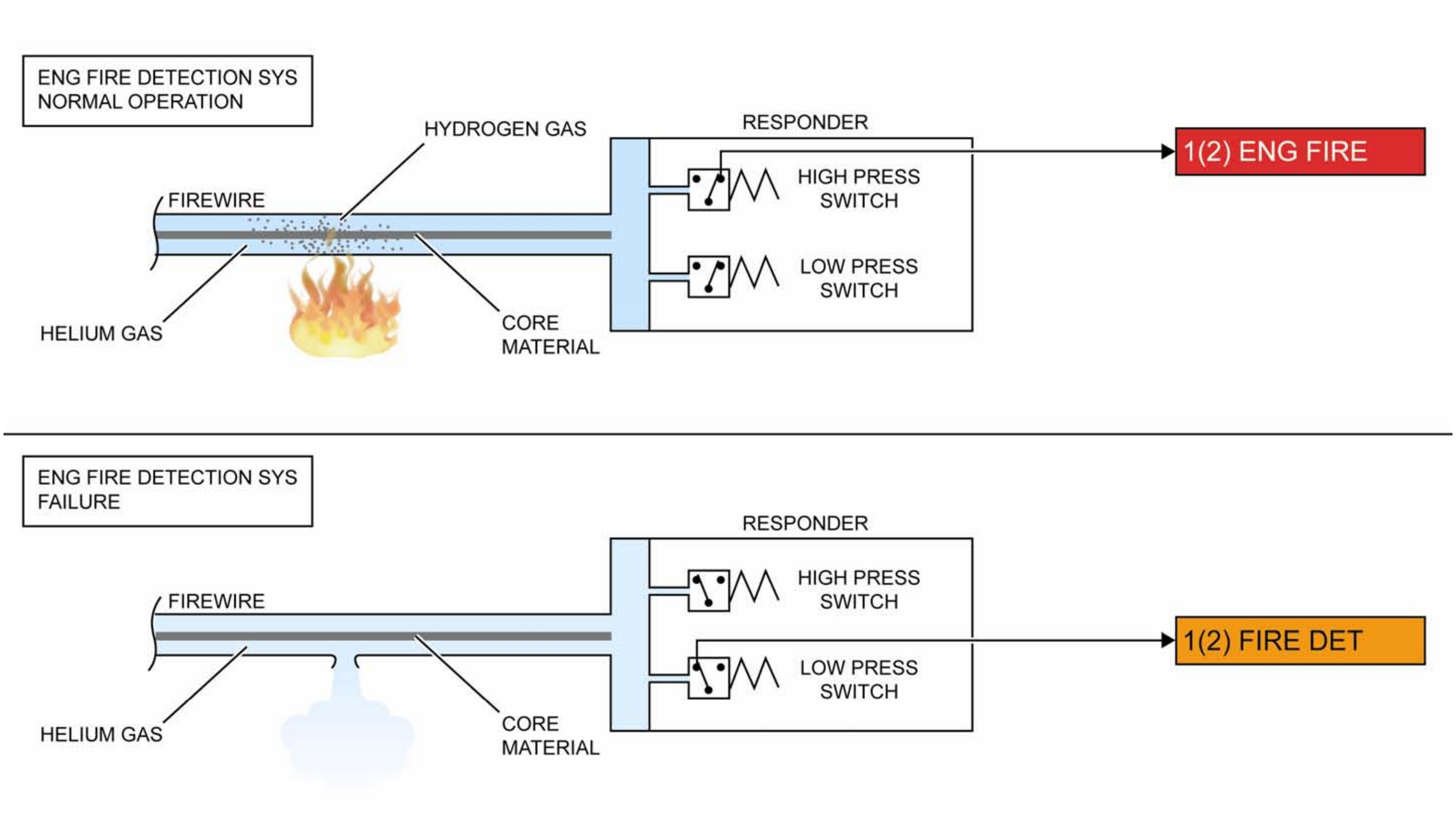
The core material is a gas absorption material impregnated with hydrogen.
One end of the firewire is connected to the responder and the other end is sealed.

RESPONDER

The responder contains two pressure switches: the high pressure switch and the low pressure switch.
The high pressure switch, in case of fire or hot gas leak, generates the ENG FIRE message.
The low pressure switch, in case of failure of the engine fire detection system (like for example, the breaking of a tube) generates the FIRE DET message.



FIRE DETECTION SYSTEM – MAIN COMPONENTS



RESPONDER – SCHEMATIC DIAGRAM AND OPERATIONS

FIRE EXTINGUISHING - GENERAL

The purpose of the fire extinguishing system is to protect the engine compartment from fire. The fire extinguisher system is composed by two identical and interconnected sub-systems one for each engine compartment.

Each system comprises a fire extinguisher bottle and double-check T-valves. The extinguisher bottles are operated manually by the pilot on the FIRE EXTING control panel. When the pilot operates the system, the double-check T-valves make sure that only one of the two extinguisher bottles is operated. If the pilot operates the system a second time, the second bottle will be operated.

In addition two hand-held fire extinguishers are located in the cockpit and in the passenger cabin.

FIRE EXTINGUISHING SYSTEM – MAIN COMPONENTS

EXTINGUISHING BOTTLE

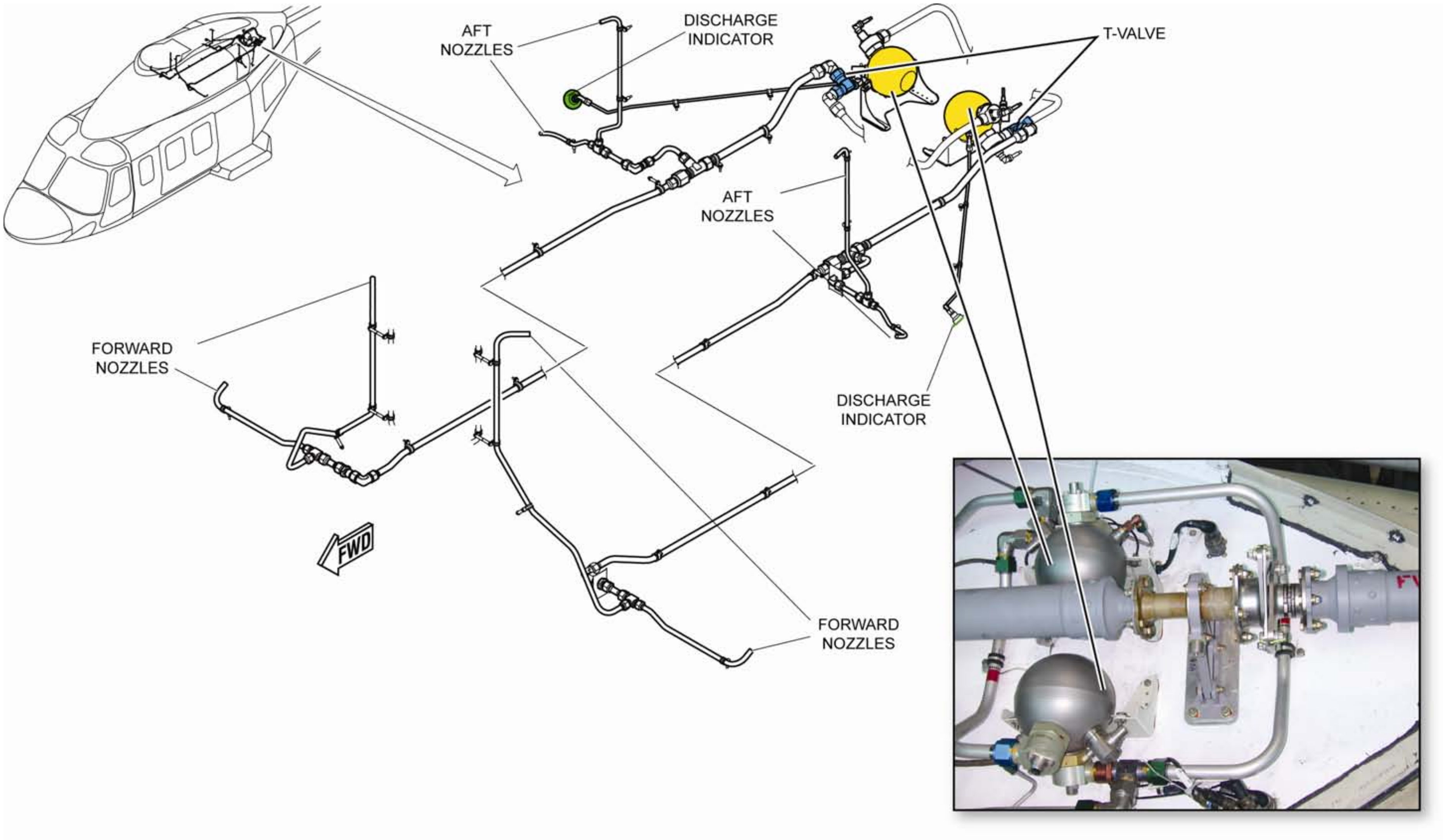
There are two 72 cubic inches extinguishing bottles filled with halon and pressurized with nitrogen gas. The bottles are installed on either side of the engine compartment and are cross-connected so that the content of any one of the two bottle can be discharged into any one engine bay and/or both bottles can be discharged into any one engine bay.

Each bottle is provided with a device which acts as a primary safety relief device. In case of overpressure, the halon agent is fully discharged outside the helicopter through the discharge indicator.

Upon discharge, the outer green disc is fired and a red circular band is displayed providing a visual indication during on ground inspection.

DOUBLE CHECK T-VALVE

Double-check T-valves interconnect the two sub-systems allowing to discharge halon into anyone of the two engine compartments.



FIRE EXTINGUISHING – MAIN COMPONENTS

FIRE PROTECTION - CONTROLS AND INDICATORS

1. BAG indicator

   FIRE (red) illuminated ………. smoke is detected into the baggage compartment

2. ENG 1 guarded indicator / push-button switch

   FIRE (red) illuminated ………. fire is detected in the engine no.1 compartment

   push-button pressed ……..…. arms (ARM light illuminated) or de-arms the fire extinguisher system outlets towards the no.1 engine bay

   ARM (amber) illuminated ……
   - the engine no.1 outlet cartridges on bottles no.1 and no.2 are armed
   - the no.1 fuel SOV (Shut-Off-Valve) is closed
   - the no.1 engine bleed air SOV is closed
   - the no.1 engine particle separator SOV is closed (if installed/optional)

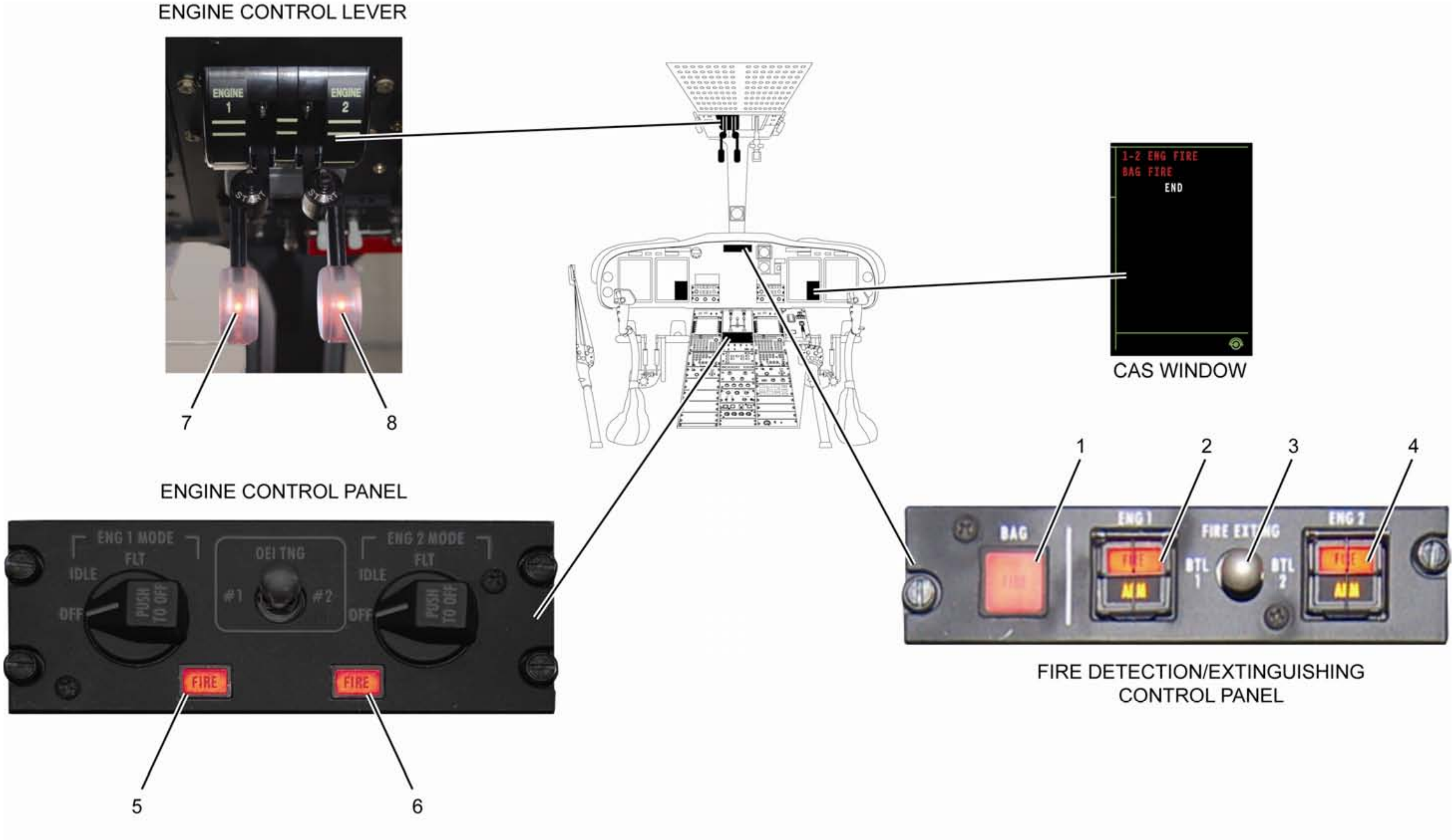
3. FIRE EXTING bottle 3 position selection switch

   central ……………………..…. None extinguisher bottle is selected

   BTL 1 …………………………. the no.1 extinguisher bottle is selected and the shot to extinguish fire take place

   BTL 2 ……………………..….. the no.2 extinguisher bottle is selected and the shot to extinguish fire take place

   NOTE. If fire persists after the first shot, a second shot is available moving the switch to the other BTL position.



FIRE PROTECTION – CONTROLS AND INDICATORS

4. ENG 2 indicator guarded indicator / push-button switch

FIRE (red) illuminated ……….	fire is detected in the engine no.2 compartment
push-button pressed ……..….	arms (ARM light illuminated) or de-arms the fire extinguisher system outlets towards the no.2 engine bay
ARM (amber) illuminated ..….	- the engine no.2 outlet cartridges on bottles no.1 and no.2 are armed
	- the no.2 fuel SOV is closed
	- the no.2 engine bleed air SOV is closed
	- the no.2 engine particle separator SOV is closed (if installed/optional)

5. FIRE indicator

FIRE (red) illuminated………….	fire is detected in the engine no.1 compartment

6. FIRE indicator

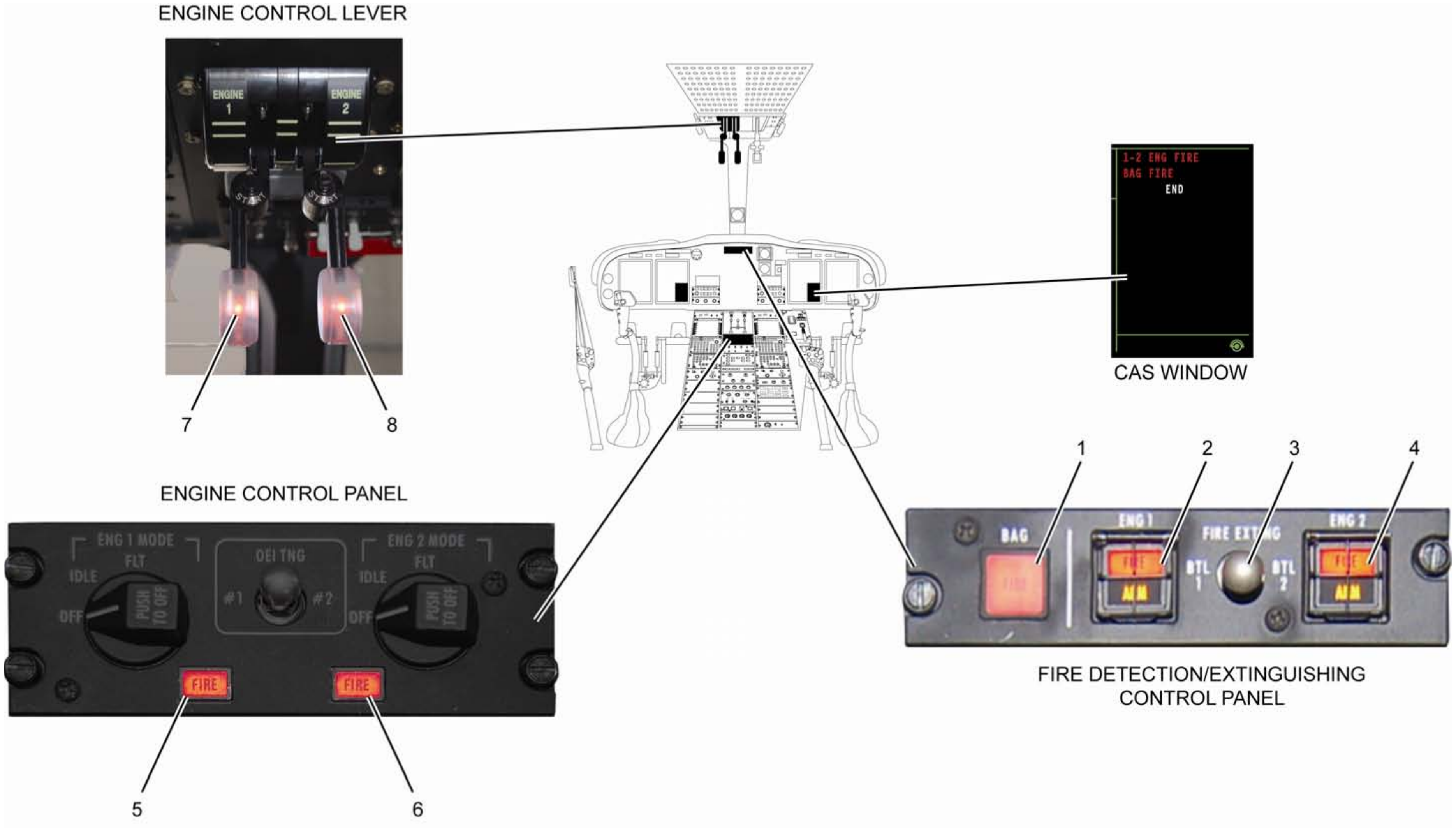
FIRE (red) illuminated …………..	fire is detected in the engine no.2 compartment

7. FIRE indicator

(red) illuminated ……….………..	fire is detected in the engine no.1 compartment

8. FIRE indicator

(red) illuminated ………………….	fire is detected in the engine no.2 compartment



FIRE PROTECTION – CONTROLS AND INDICATORS

FIRE PROTECTION SYSTEM – TEST

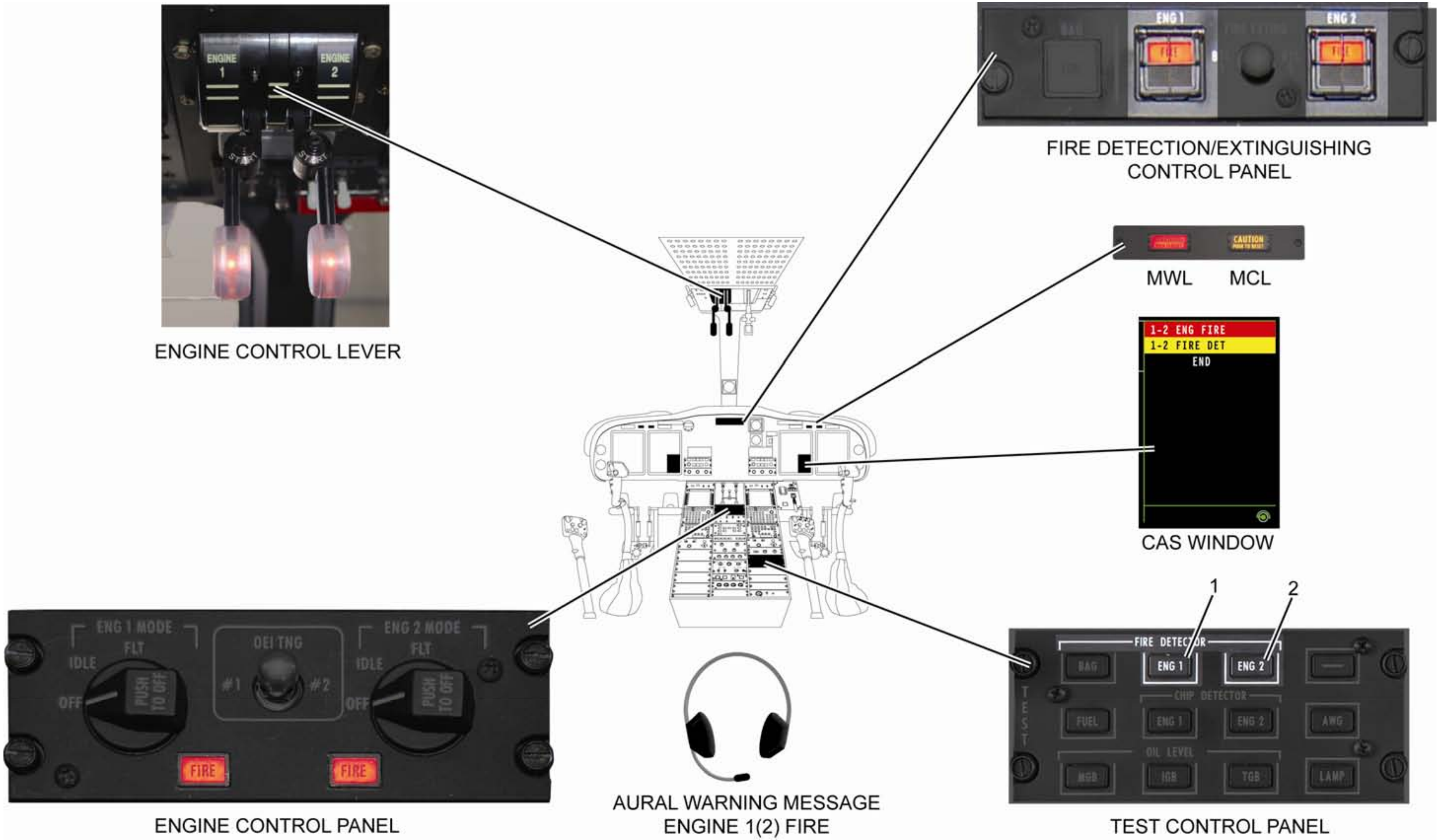
The test of the fire detection system is provided through the TEST panel.

1. FIRE DETECTOR ENG 1 push-button

   pressed ……………………..
   - ENG 1 ECL fire illuminates red
   - ENG 1 FIRE illuminates red on the FIRE EXTING control panel
   - the MWL and MCL illuminate
   - FIRE illuminates red (left indicator) on the ENG control panel
   - 1 ENG FIRE (warning) and 1 FIRE DET (caution) are displayed in the CAS window
   - audio tone and ENGINE 1 FIRE voice warning message

2. FIRE DETECTOR ENG 2 push-button

   pressed ……………………
   - ENG 2 ECL fire illuminates red
   - ENG 2 FIRE illuminates red on the FIRE EXTING control panel
   - the MWL and MCL illuminate
   - FIRE illuminates red (right indicator) on the ENG control panel
   - 2 ENG FIRE (warning) and 2 FIRE DET (caution) are displayed in the CAS window
   - audio tone and ENGINE 2 FIRE voice warning message



ENGINE CONTROL LEVER

FIRE DETECTION/EXTINGUISHING CONTROL PANEL

CAS WINDOW

ENGINE CONTROL PANEL

AURAL WARNING MESSAGE ENGINE 1(2) FIRE

TEST CONTROL PANEL

FIRE PROTECTION SYSTEM – TEST

FIRE DETECTION – PRINCIPLE OF OPERATION

The firewire element takes into account two types of high temperature conditions:

- the first type is a general overheat condition. In this case if the temperature around the firewire element increases over a preset limit. As a consequence the sensor acts in accordance with the law of gases: if the volume of the firewire element is held constant, its pressure will increase as temperature increases. The helium gas in the tube exerts a pressure which closes the high pressure switch that generates the ENG FIRE message.

- the second type is a fire condition at a short section of the firewire sensor element. In this case the core material releases a large volume of hydrogen gas so that the internal pressure of the firewire element increases very quickly and the gas in the tube exerts a pressure which closes the high pressure switch that generates the ENG FIRE message. After the situation is corrected, the material reabsorbs the hydrogen and the system returns to a stand-by mode.

In case of leakage of the helium gas from the firewire element, the pressure will decrease and operates the low pressure switch which generates the FIRE DET message.

FIRE EXTINGUISHING – PRINCIPLE OF OPERATION

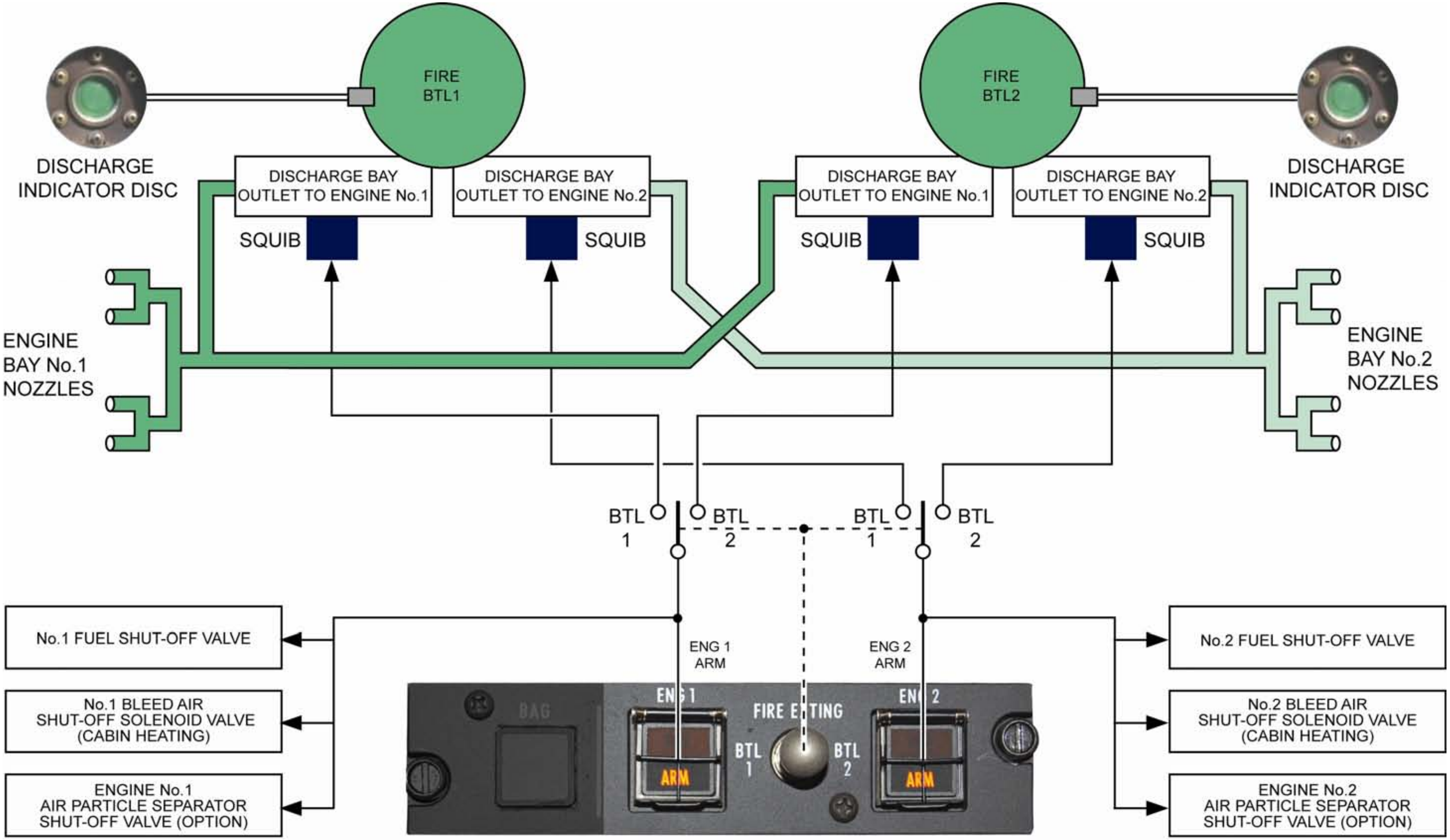
Refer to AW139–Rotor Flight Manual–4D for emergency procedures to be applied in case of fire (on the ground or in flight).

NOTE.

The command to extinguish fire is done via the FIRE switch on the FIRE EXTG panel that, after been pressed, illuminates the amber segment ARM.

The extinction of the fire does not implies the automatic re-opening of the fuel shut-off valve automatically closed when the switch FIRE EXTING has been moved from the central position.

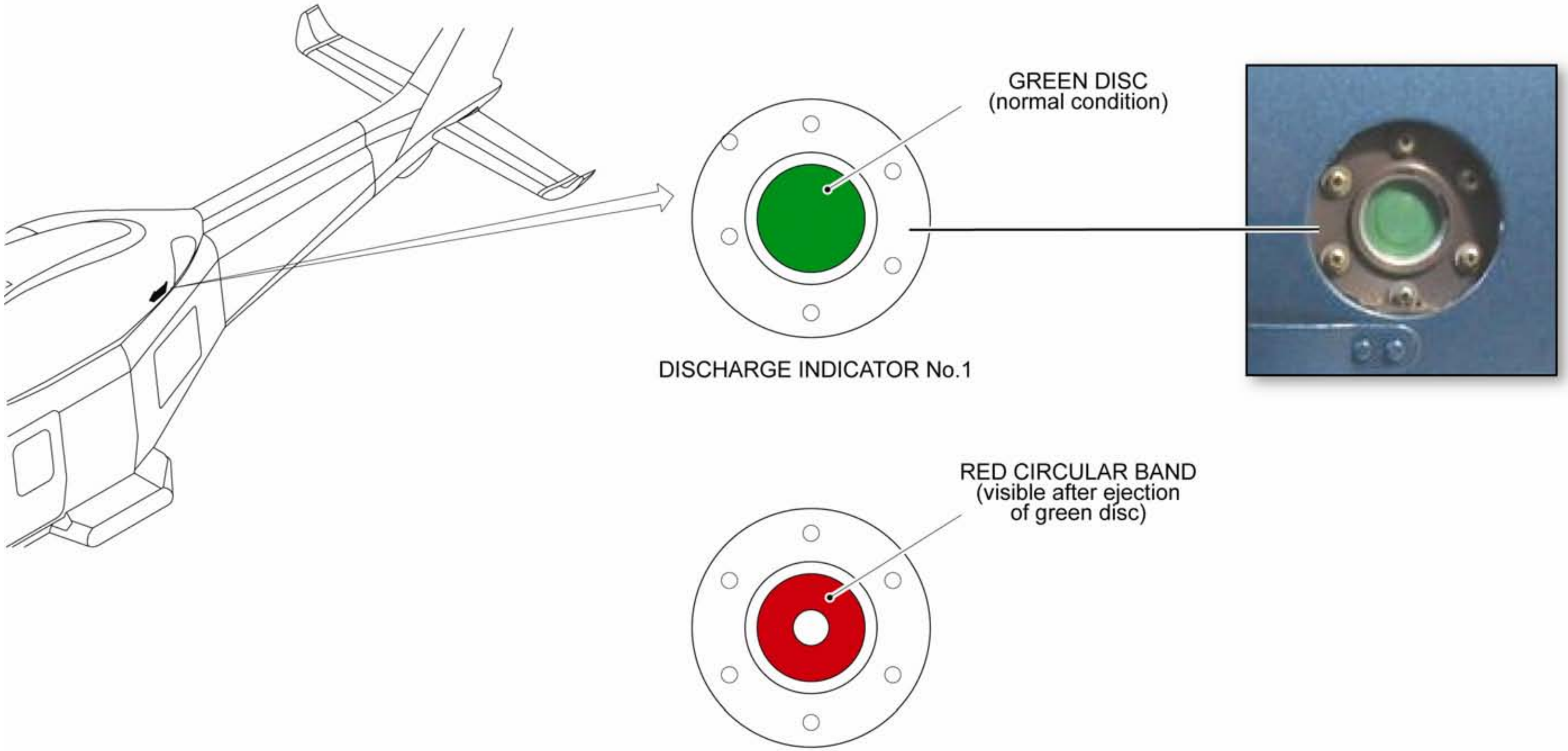
In case the first shot to extinguish fire fails, a second shot is available moving the switch to the second bottle indication.



FIRE EXTINGUISHING – SCHEMATIC

NOTE

THE DISCHARGE INDICATOR No.2 INSTALLATION IS SIMILAR FOR THE ENGINE No.2



DISCHARGE INDICATOR

BAGGAGE FIRE DETECTION SYSTEM – GENERAL

The purpose of the baggage fire detection system is to detect smoke in the baggage compartment. The system comprises a smoke detector installed in the baggage compartment.

BAGGAGE FIRE DETECTION SYSTEM – MAIN COMPONENTS

SMOKE DETECTOR

The smoke sensor is made by a photoelectric device that operates on the light-scattering principle.

The detector alarms when the smoke concentration level exceeds a predetermined level.

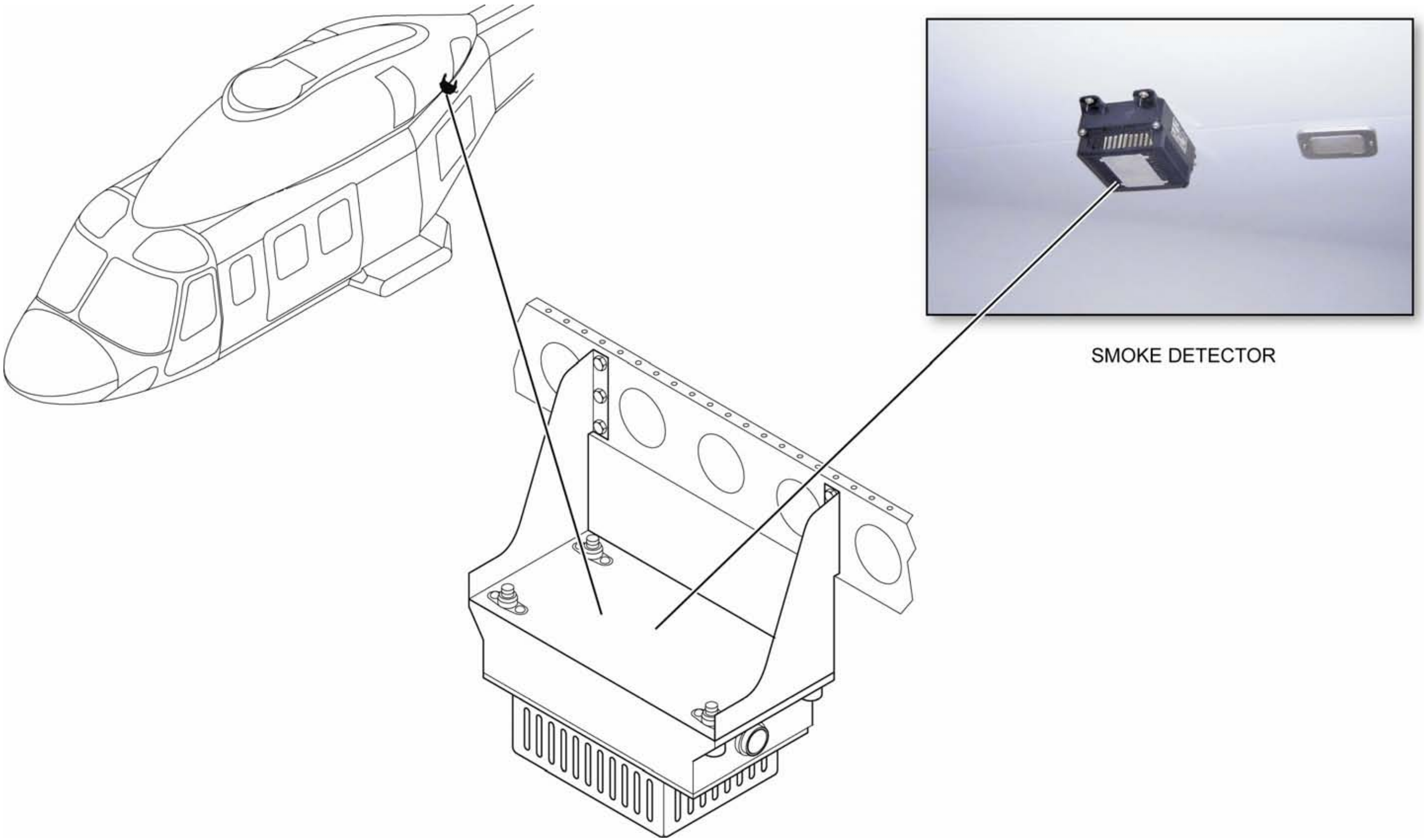
The device employs two light sensing and amplifying channels installed in the baggage bay.

SMOKE DETECTOR – OPERATION

The reference channel senses the amount of light emitted by an internal LED (Light Emitting Diode) source.

The smoke channel senses the amount of emitted light scattered by smoke particles in the baggage.

The smoke detector alarms when the output from the smoke channel exceeds a predetermined smoke concentration level.



FIRE DETECTION SYSTEM – BAGGAGE COMPARTMENT

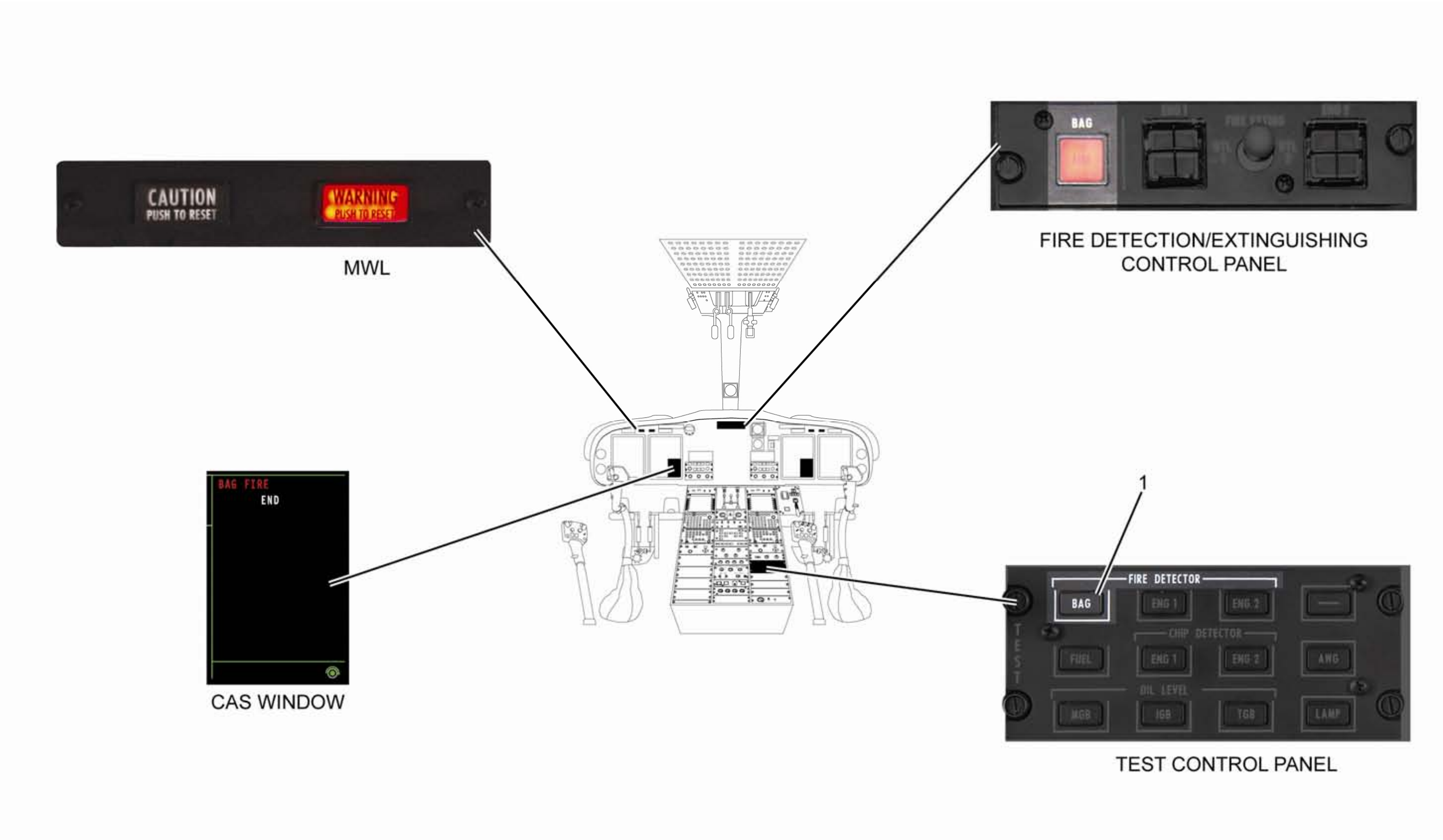
BAGGAGE FIRE DETECTION – TEST

1. FIRE DETECTOR BAG push-button

   pressed …………………….. - MWL illuminate
   - the message BAG FIRE is displayed in the CAS window
   - the BAG indicator illuminates red on the FIRE EXTING control panel

   NOTE.

   When the push-button is released all the above lights and messages go OFF. See AW139-RFM-4D Section 2 Normal Procedures COCKPIT/ENGINE PRE-START CHECKS.



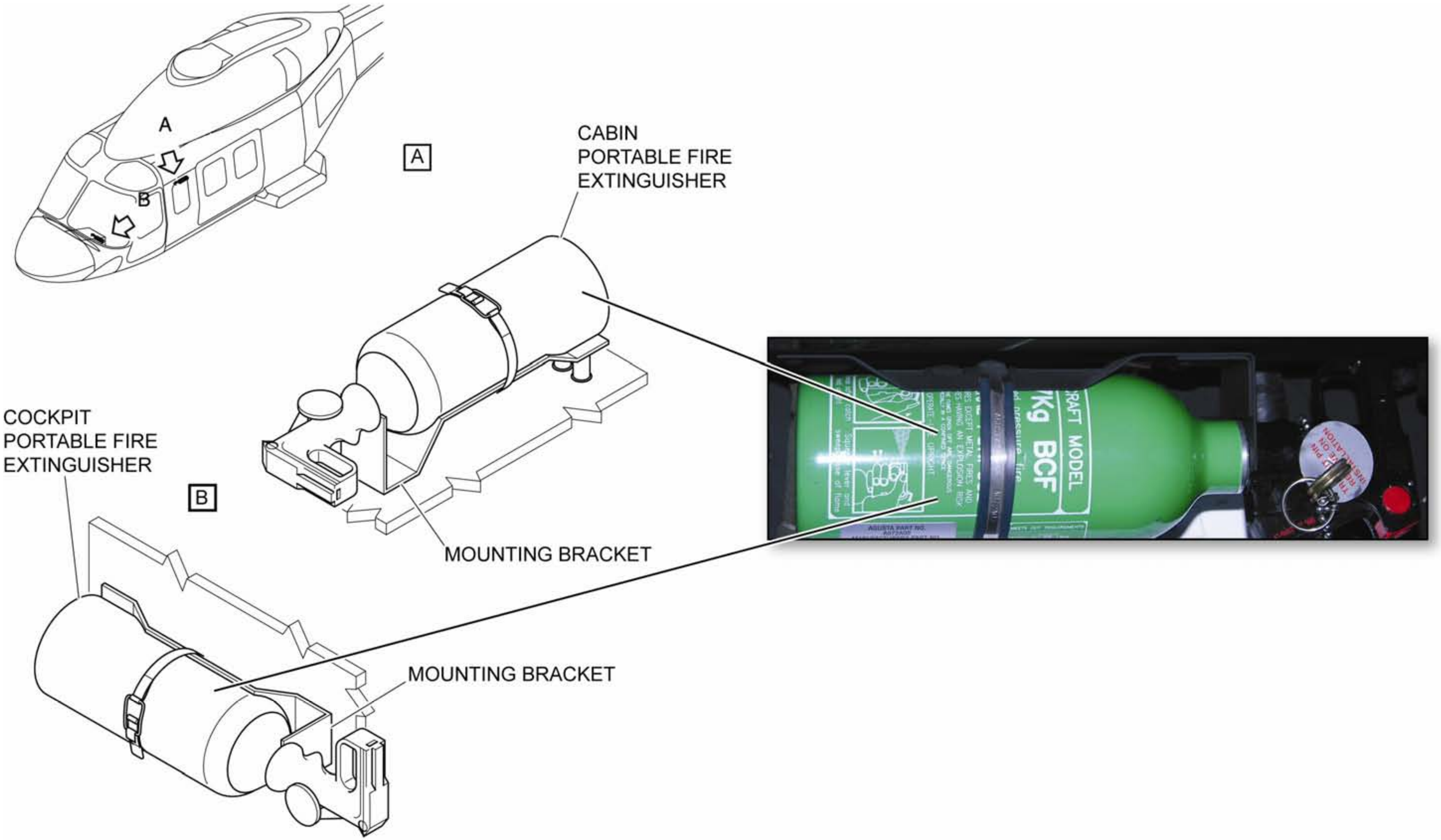
BAGGAGE FIRE DETECTION – TEST

PORTABLE FIRE EXTINGUISHER – GENERAL

One portable fire extinguisher is installed in the cockpit between the seat of the pilot and the co-pilot.

A quick release mounting bracket allows for rapid removal of the fire extinguisher in case of fire. The extinguishing agent is HALON 1211 and the portable fire extinguisher can be used against small carbonaceous fires, flammable liquid fires and electrical fires.

A similar portable fire extinguisher is located into the cabin.



PORTABLE FIRE EXTINGUISHER INSTALLATION

FIRE DETECTION – CAS WARNING MESSAGES

CAS CAPTION	MESSAGE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
1(2) ENG FIRE	Fire is detected in the ENG 1 (2) bay or during the test When this warning is detected the AWG provides two pairs of tones + ENGINE 1 (2) FIRE - ENGINE 1 (2) FIRE aural message This sequence is continuously repeated until the failure is corrected or the reset input activated **CAUTION** In case of a subsequent fire in the other engine bay the initial ARM 1(2) pushbutton must be deselected to allow operation of the ARM 2(1) pushbutton	ENGINE BAY FIRE	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES FIRE
BAG FIRE	Smoke is detected into baggage compartment or during the test When this warning is detected the AWG provides no tone + WARNING - WARNING aural message	BAGGAGE BAY FIRE	

CAS CAPTION	DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
	No single set of detailed procedures can address all the fire scenarios that are possible. The most urgent action is to get the aircraft shut down and evacuated immediately.	COCKPIT / CABIN FIRE (GROUND)	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES FIRE
	An in-flight fire has no single set of detailed procedures that can address all the fire scenarios that are possible in flight. The most urgent action is to get the aircraft on the ground as soon as possible with a reasonable degree of safety.	COCKPIT / CABIN FIRE (FLIGHT)	
	An electrical fire is indicated by a smell of burning insulation and/or acrid smoke. See the RFM for the relevant procedure	ELECTRICAL FIRE/SMOKE (GROUND)	
	Electrical fires are often indicated by a smell of burning insulation and/or acrid smoke. The most important consideration is to maintain safe flight conditions while investigating the cause. Unnecessary electrical equipment must be switched off while detecting the source of an electrical fire. Unless the source of the smoke or fire can be positively identified (CAS display or C/B panel) and the equipment electrically isolated, carry out procedure detailed on RFM.	ELECTRICAL FIRE/SMOKE (FLIGHT)	

FIRE DETECTION – CAS CAUTION MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
1(2) FIRE DET	Associated engine fire detect system not operational	ENGINE FIRE DETECTOR SYSTEM	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES FIRE

CHAPTER 28

FUEL

SECTION 00 – GENERAL

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FUEL – GENERAL

The fuel system consists of the following sub-systems:

- fuel storage
- fuel distribution
- fuel indicating

FUEL STORAGE – GENERAL DESCRIPTION

The storage system is composed by two tanks connected together through a flange located on the lower side of the tanks. Both tanks are gravity refilled through a filler cap located on top of the tank 2(optional left hand gravity filler cap and optional pressure (closed circuit) refuelling are also available).
At the bottom of each tank is located a sump plate where a fuel drain valve and a water drain valve are installed. The tanks are assured to the frame of the helicopter by two different type of restrains: velcro and nylon chord.

To ensure an equal level of pressure between the tanks and the existing ambient pressure, each tank is provided at the top with a venting pipeline routed to the opposite side.

A flame arrestor in each venting pipeline prevents a flashback to the tanks in the event of fire hazards, such as lighting strikes.
The total fuel tank capacity is 1588 litres (1270 kg).

FUEL STORAGE – MAIN COMPONENTS

TANKS

The two tanks (tank 1 and tank 2) are located on the rear side of the cabin between the engine compartment floor and the cabin floor. Each tank retain a quantity of fuel (about 228 kg or 285 litres) that is below the interconnection to avoid that a break of one of the tank drains all fuel.
In case of crash condition, the main landing gear collapses upward, away from the fuel tank: in this case, foams in the lower side of each tank help to maintain the tank in the right position.

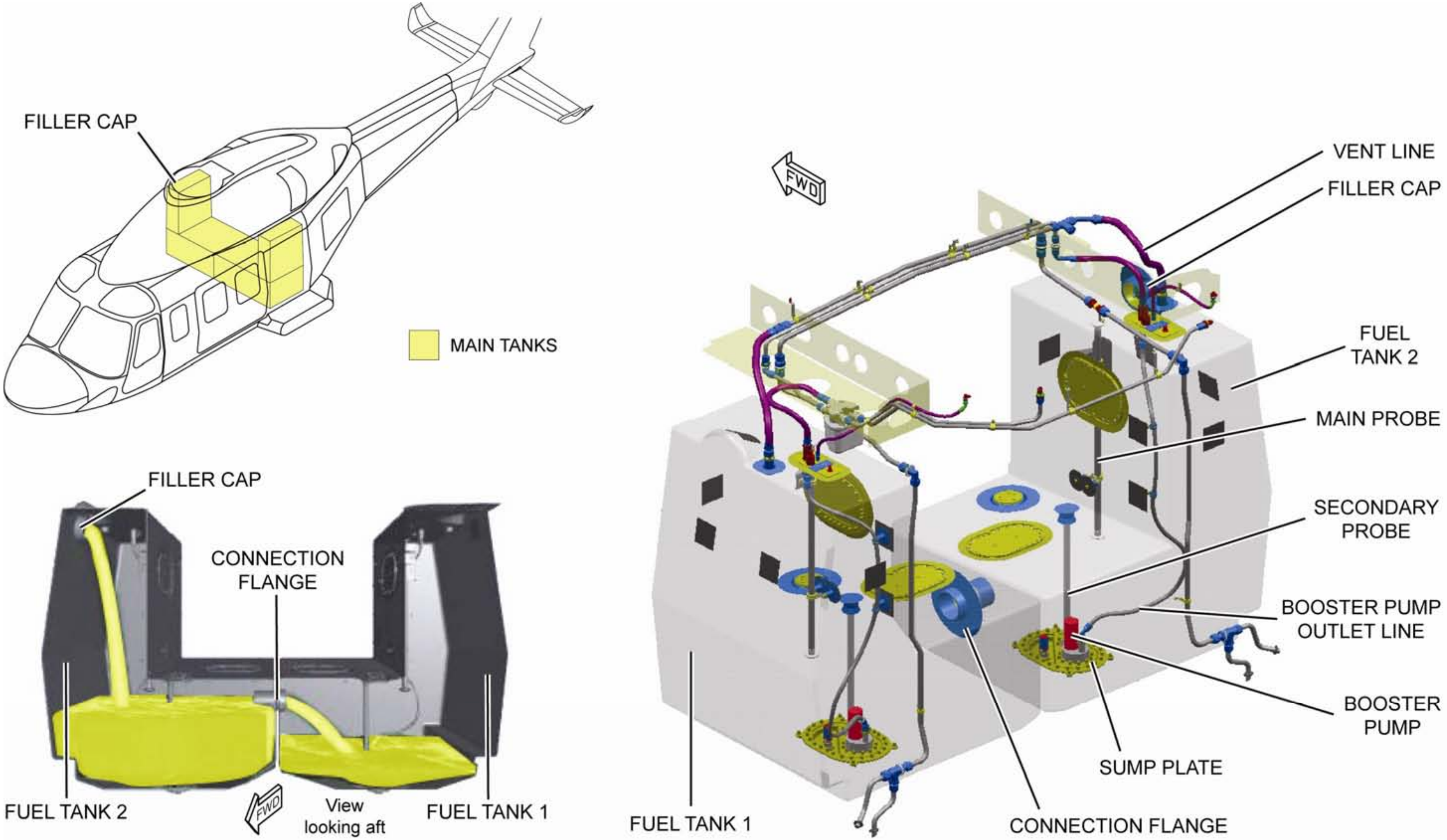
SUMP PLATES

The sump plates are located at the bottom of each tank and are provided with:

- a fuel booster pump
- a secondary fuel quantity probe with a low level sensor
- a fuel drain valve
- a water drain valve

FUEL DRAIN VALVES AND WATER DRAIN VALVES

The water drain valves can be operated electrically with a switch located in the main landing gear sponson; it permits to sump fuel for normal check operations.
The fuel drain valve is used to drain the fuel for maintenance purposes only.



FUEL STORAGE

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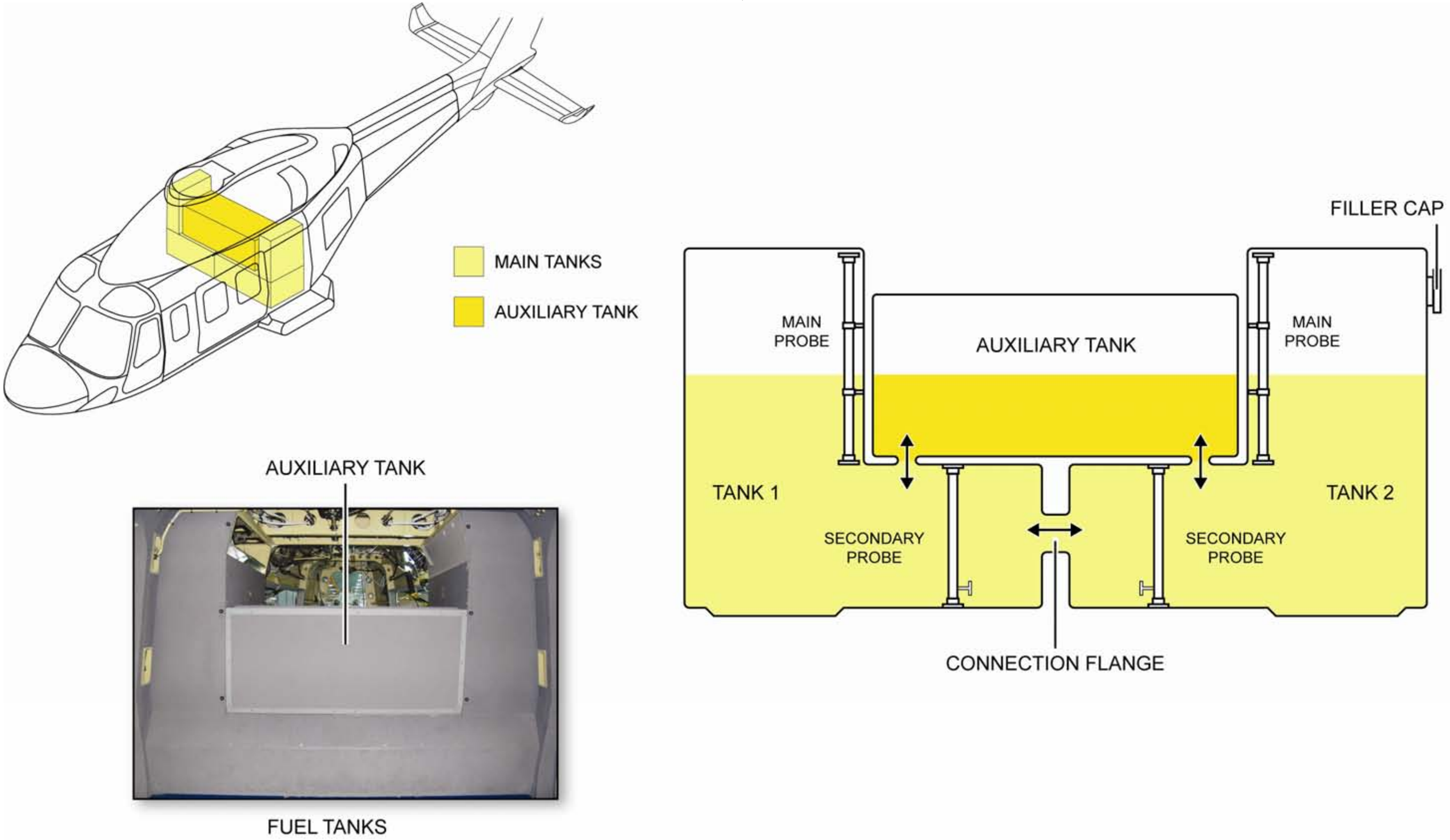
AUXILIARY FUEL TANK

The auxiliary fuel tank is installed between the no.1 and the no.2 main fuel tanks. The auxiliary fuel tank is an anti-crash, bladder-type tank with a capacity of 500 litres (400 kg).

At the bottom of the auxiliary fuel tank are installed two auxiliary ducts that connect the auxiliary fuel tank with the no.1 and no.2 main fuel tanks. All tanks (main and auxiliary) are refuelled through the same filler neck.
The vent line of the auxiliary fuel tank is connected to the vent line of the two main tanks (left and right).

One electrical connector connects the auxiliary tank to the Fuel Computer Unit (FCU) which automatically adjusts for the fuel contained in the auxiliary tank.

No probes are installed inside of the auxiliary tank.



FUEL STORAGE – MAIN AND AUXILIARY TANKS

FUEL DISTRIBUTING – GENERAL

The purpose of the fuel distribution system is to supply fuel to the engines. The fuel distribution system comprises two DC electrically operated booster pumps (one for each fuel tank) and a fuel manifold which allows to supply fuel to the engines under different conditions.

In normal condition each booster pump supplies fuel to the relevant engine through a feeding line. A pressure switch installed on each line detects a low pressure pump outlet. The feeding line goes through a hose to the fuel manifold where a non return valve is installed.

The fuel manifold is composed by a DC electrically operated shut-off valve and a pressure transducer for each feeding line plus one DC electrically operated crossfeed valve.

FUEL DISTRIBUTING – MAIN COMPONENTS

BOOSTER PUMPS

Each booster pump, installed on the sump plate, is provided with an integral overheat protection switch which turns off power whenever an overheat condition exists (power is restored when the overheat is no longer detected).

FUEL MANIFOLD

The fuel manifold, located above the left tank, interconnects the booster pump outlet lines and the fuel feeding lines. It groups in a single LRU the following components:

- two fuel Shut-Off Valves (fuel SOV), one for each fuel feeding line
- two pressure transducers, one for each fuel feeding line (see fuel indicating)
- one crossfeed valve (XFEED)

FUEL SHUT-OFF VALVES (FUEL SOV)

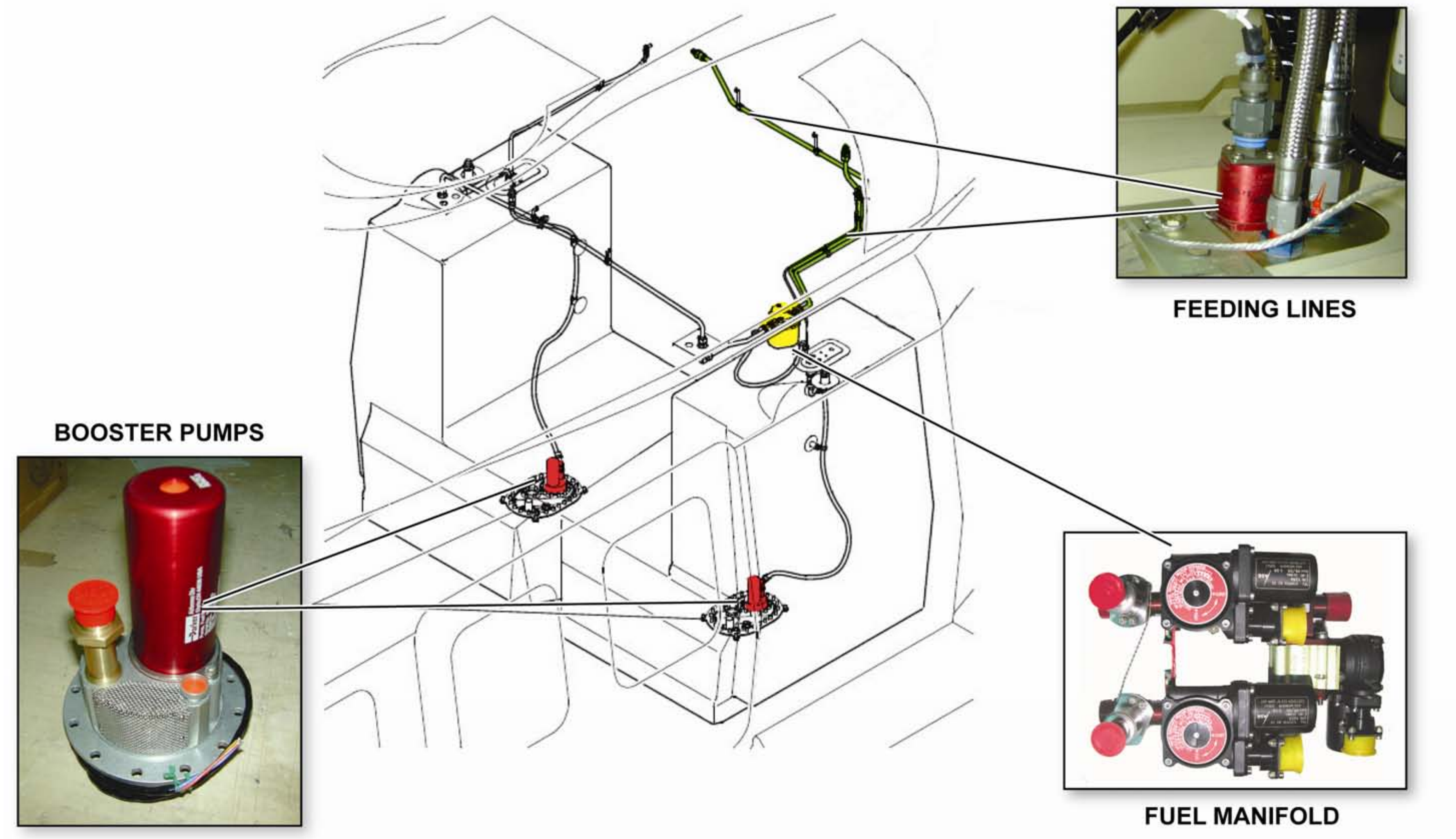
The fuel Shut-Off Valves are two DC electrically-operated motorized valves that permit (SOV opened) or prevent (SOV closed) fuel feeding to the relevant engine.

NON RETURN VALVES (CHECK VALVES)

The non-return valves are installed on each booster pump outlet line where it exits the tank tag side. The non return valves prevent a reverse flow to the relative feeding line in case of failure of the booster pump and the cross-feed valve is open.

CROSS-FEED VALVE (XFEED)

The cross-feed valve is a DC electrically-operated valve. The two fuel feeding lines are connected to each other to permit and allow that in case of any drop of pressure, one of the tank feeds the other engine.



FUEL DISTRIBUTING – MAIN COMPONENTS

FUEL INDICATING – GENERAL

The purpose of the fuel indicating system is

- measure the fuel quantity and the pressure in the fuel supply lines
- provide status information for caution and advisory messages

The fuel indicating system includes the following main components:

- one Fuel Control Unit (FCU)
- two Low Level Detection Module (LLDM)
- two main probes and two secondary probes
- two fuel Low Lever Sensors (LLS)
- two pressure switches
- two pressure transducers
- one Fuel Control Panel (FCP)

FUEL INDICATING – MAIN COMPONENTS

FUEL COMPUTER UNIT (FCU)

The Fuel Computer Unit (FCU) is an electronic device that

- monitors fuel quantity (level) data coming from the probes
- provides the required excitation current to main and secondary probes (LH and RH tanks) and calculates fuel density and converts fuel level into fuel weight
- sends fuel quantity (weight) data to MAU 1 and MAU 2 for fuel quantity indicator on the DUs

The fuel tank geometry is part of the FCU software and the specific fuel density compensation is calculated as a function of the fuel temperature. The FCU has two independent and separated channels. The data link interface permits a comparison between channels and any failure of one channel does not affect the proper operation of the other.
A Built-In-Test (BIT) of the FCU detects internal 1 (2) channel failures providing the caution 1 (2) FCU FAIL.

Two independent Low Level Detection (LLD) modules (one for the FCU channel 1 and one for FCU channel 2) have the function to monitor the fuel low level of each tank. This function is made using a dedicated sensor (Low Level Sensor) and remains active also in case of failure of either or both FCU channels.
The FCU compensates the fuel quantity indication for roll attitude within ± 5°.

FUEL PROBES

One capacitance-type main probe is installed in the upper part of each fuel tank to measure the quantity (level) of fuel. The main probes provide the height of the fuel in the tanks to the FCU which converts data into fuel mass.

One capacitance-type secondary probe is installed in the lower part of each fuel tank to measure the quantity (level) of residual fuel in the tank. The secondary probes provide the height of the fuel in the tanks to the FCU which converts data into fuel mass.

LOW LEVEL SENSORS (LLS)

Two Low Level Sensors (LLS) (thermistor-type) are installed on the lower part of the secondary probe. They are completely independent from the secondary probes and provide the caution 1 (2) FUEL LOW when the fuel level quantity drops below 92 kg (112 litres).

PRESSURE SWITCHES

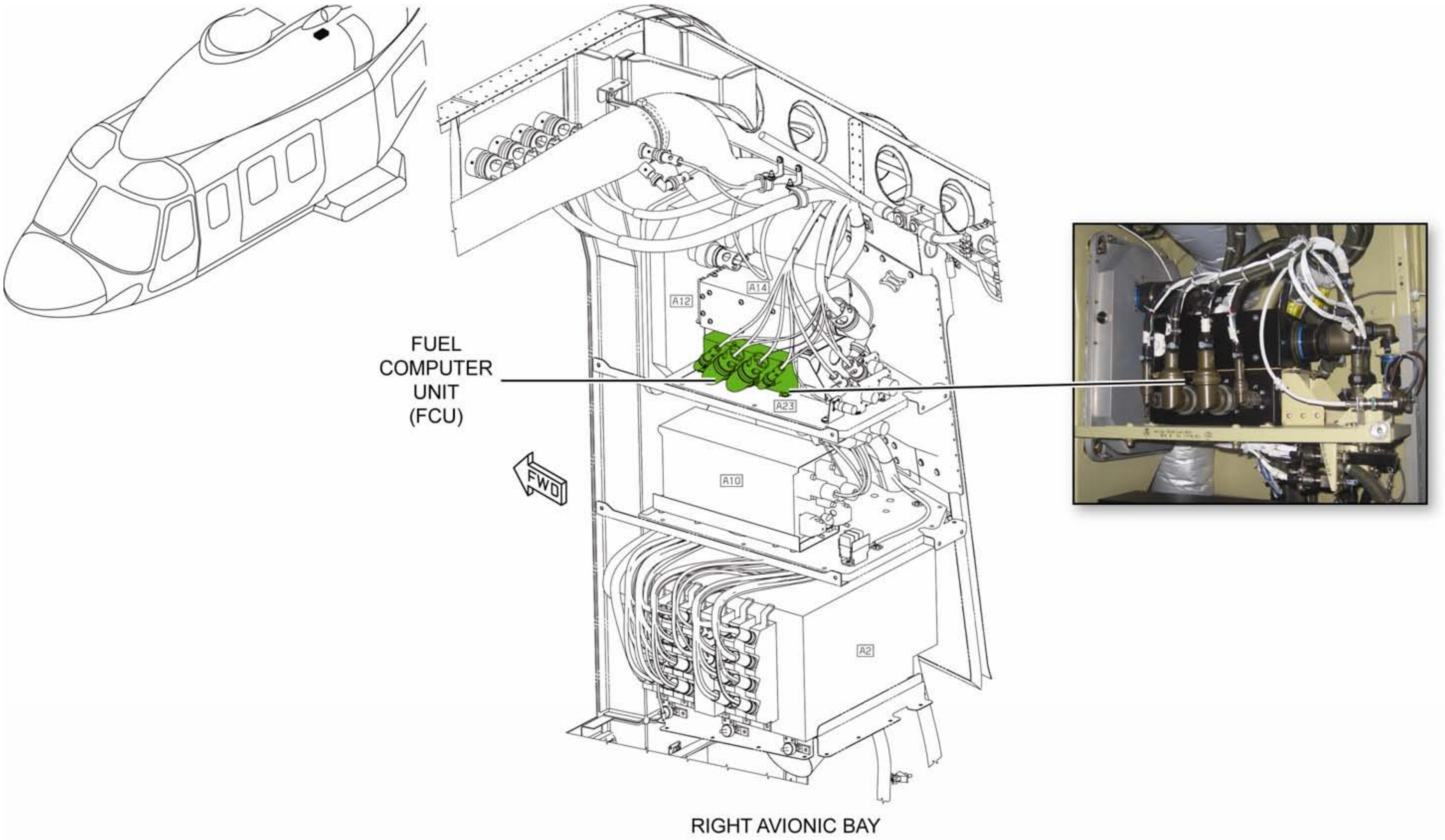
The pressure switch (located at the top of each fuel tank) allows to activate the cross-feed valve when the pressure reaches a preset level providing an automatic cross-feed function when the XFEED switch on the FUEL control panel is set to NORM.

PRESSURE TRANSDUCERS

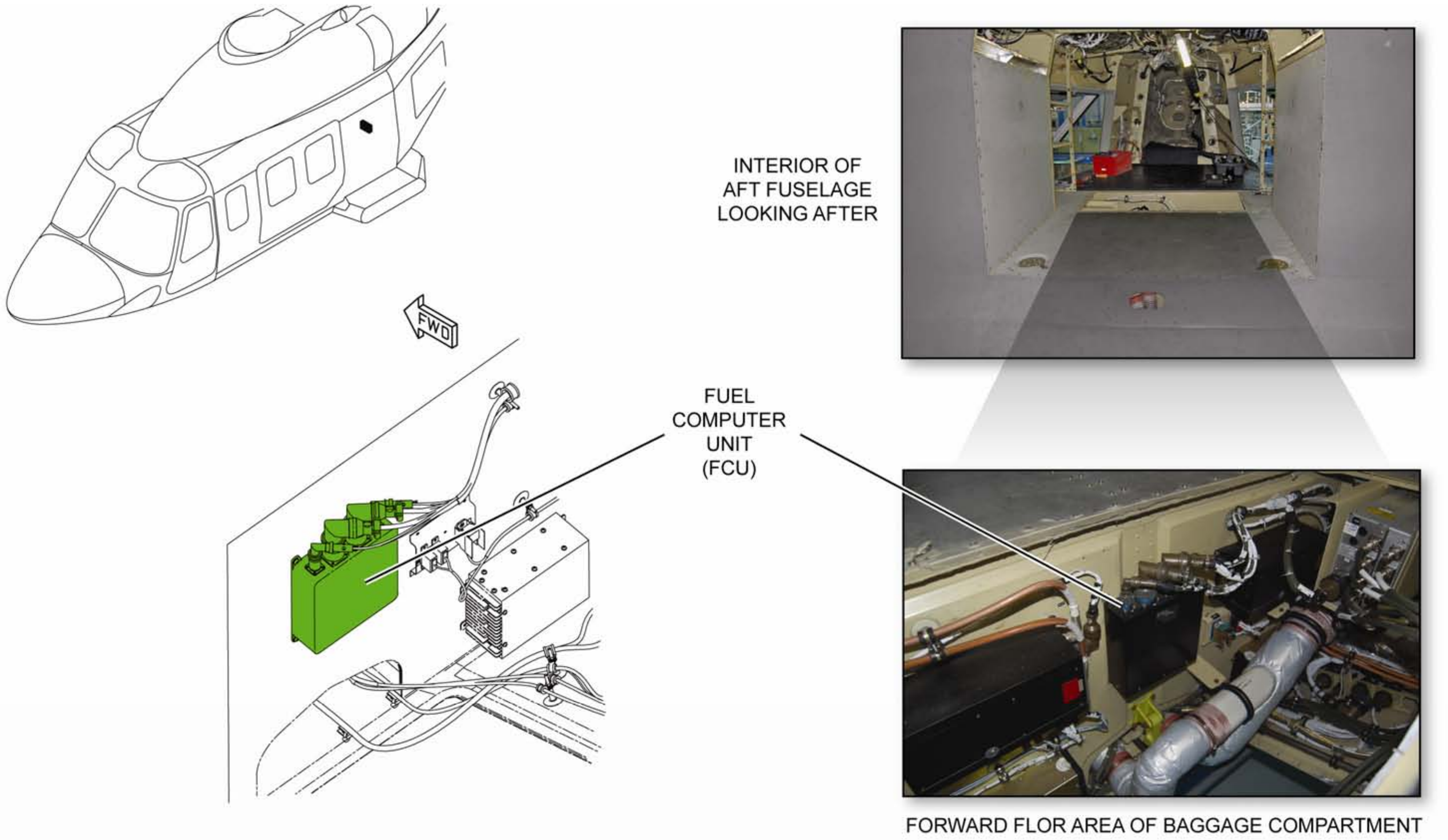
Two pressure transducers, installed on the fuel manifold, provide a continuous pressure monitoring on each fuel feeding line and transmit the information to the MAUs.

FUEL CONTROL PANEL (FCP)

The Fuel Control Panel allows the pilot to control the booster pumps, the fuel Shut-Off Valves (SOV), the cross-feed valve and to monitor the fuel SOVs and XFEED valve conditions (open/closed).



FUEL COMPUTER UNIT (FCU) – SHORT NOSE CONFIGURATION



FUEL COMPUTER UNIT (FCU) - LONG NOSE CONFIGURATION

FUEL – CONTROLS AND INDICATORS

1. FUEL PUMP 1 switch

   OFF ……....……….. the booster pump 1 is OFF

   ON …...………….... the booster pump 1 starts to operate

2. FUEL 1 switch (red cap)

   OFF ...…..........….. the fuel Shut-Off Valve (SOV) of the line 1 is CLOSED

   ON ……......……..... the fuel Shut-Off Valve (SOV) of the line 1 is OPEN

   NOTE. In case of fire, the SOV 1 is automatically closed when the ENG1 guarded push-button on the FIRE EXTING control pane is pressed.

3. SOV 1 indicator

   vertical line ……….. displayed when SOV 1 is open and the fuel flow passes through the valve

   horizontal line ……. displayed when SOV 1 is closed and no fuel flow passes through the valve

4. XFEED switch

   CLOSED ..………... closes manually the crossfeed valve

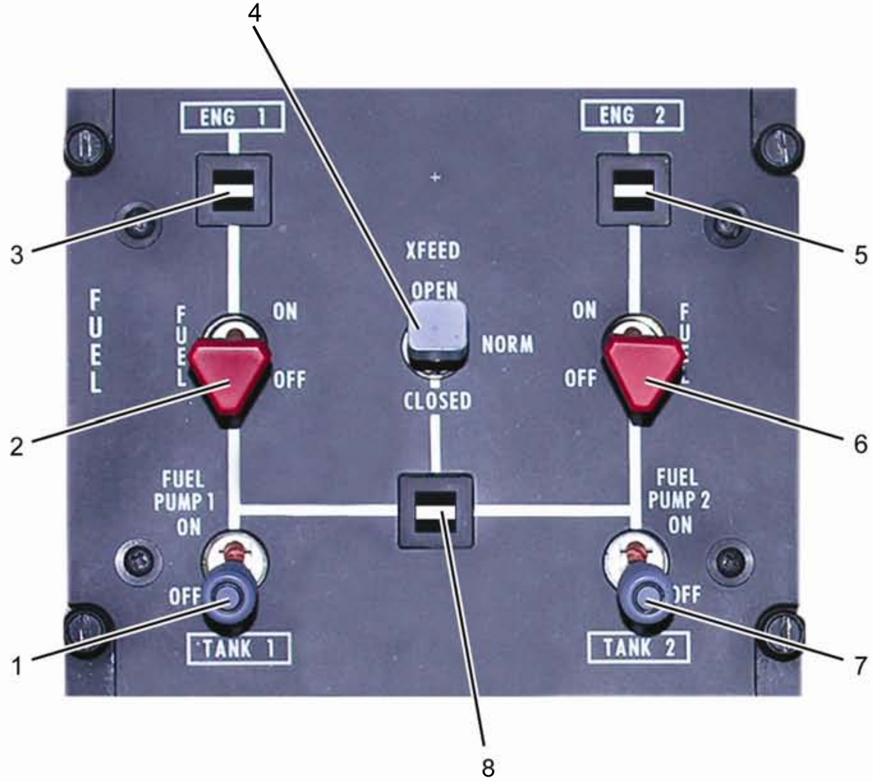
   NORM ..…………... opens automatically the crossfeed valve if the pressure switch in one fuel line detects a low pressure condition

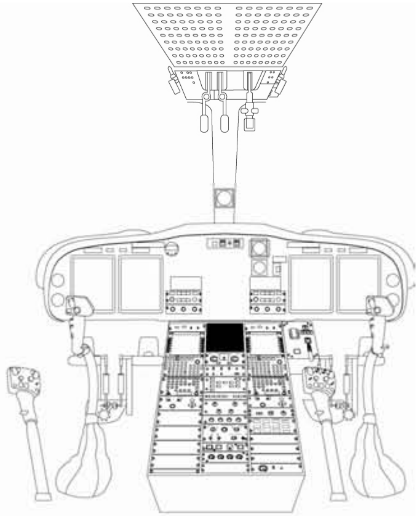
   OPEN ...…………... opens manually the crossfeed valve

5. SOV 2 indicator

   vertical line ………. displayed when SOV 2 is open and the fuel flow passes through the valve

   horizontal line ……. displayed when SOV 2 is closed and no fuel flow passes through the valve





FUEL CONTROL PANEL

FUEL – CONTROLS AND INDICATORS

6. FUEL 2 switch (red cap)

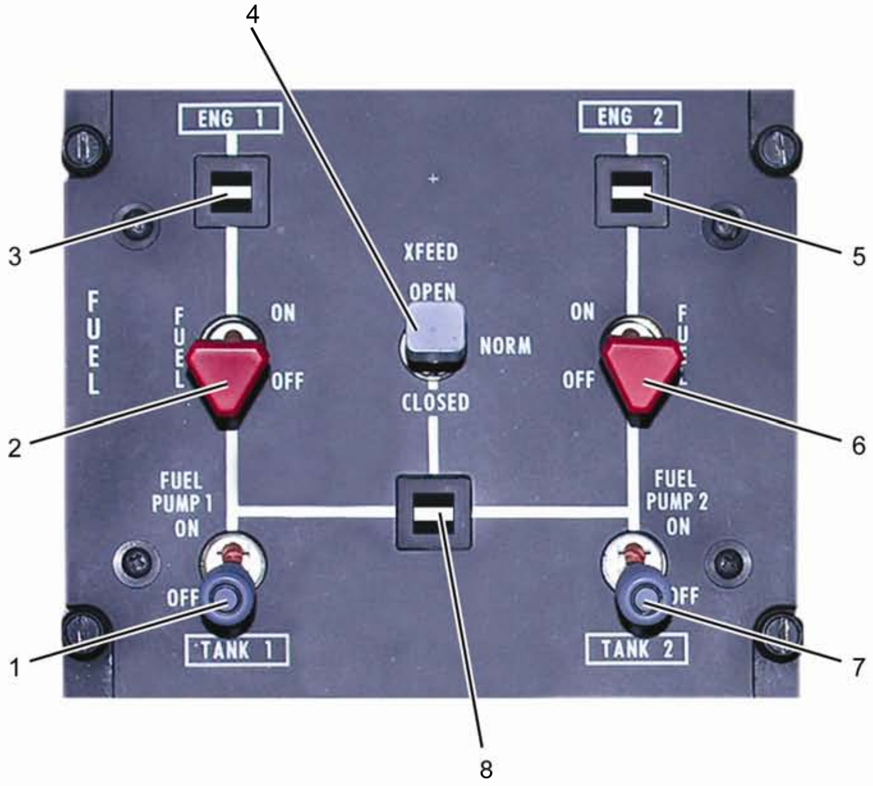
OFF ...…..............…..	the fuel Shut-Off Valve (SOV) of the line 2 is CLOSED
ON ………....................	the fuel Shut-Off Valve (SOV) of the line 2 is OPEN
	NOTE. In case of fire, the SOV 2 is automatically closed when the ENG2 guarded push-button on the FIRE EXTING control pane is pressed.

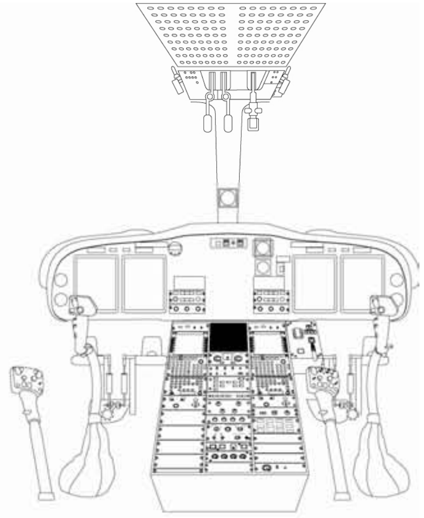
7. FUEL PUMP 2 switch

OFF ..…..............…...	the booster pump 2 is OFF
ON …...……………...	the booster pump 2 starts to operate

8. XFEED indicator

black ………………..	when the helicopter is disconnected from an electrical power supply
vertical line …………	displayed when the cross-feed valve is closed and each engine is fed only by the relevant tank
horizontal line	displayed when the cross-feed valve is open and the fuel is sucked from one tank to feed both engines





FUEL CONTROL PANEL

FUEL – CONTROLS AND INDICATORS

FUEL INDICATING

1. FUEL PRESSURE

The fuel pressure indicator shows LH and RH fuel feeding line pressures measured after the relevant fuel SOV.

2. FUEL QUANTITY

Digital readouts of the fuel quantity in the left tank, the tank 2and the total.

The fuel quantity indicator is calibrated to measure fuel quantity in pounds (LB) or kilograms (KG).

When cross feeding, the tank with pump off, NOT supplying the engines, will have a maximum quantity of unusable fuel of 228 kg (285 litres). This unusable fuel quantity value will change to grey to indicate the tank can no longer supply fuel.

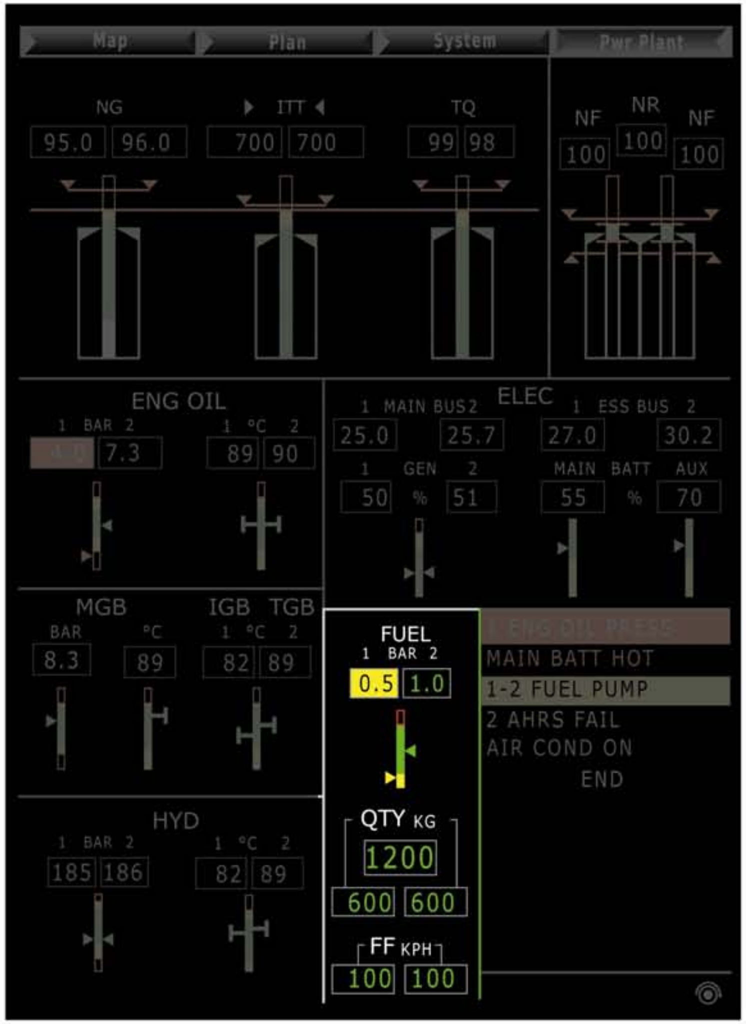
Close X-FEED to restore the availability of up to 228 kg (285 litres) of fuel (fuel level value returns to green).

3. FUEL FLOW

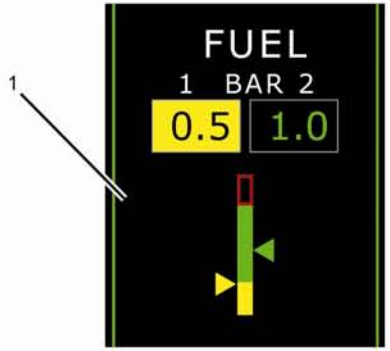
The FF (Fuel Flow) data in KPH (Kilo Per Hour) are displayed on the MFD after the engines have reached the IDLE condition (Ng > 62%).

The FF value is computed using data from

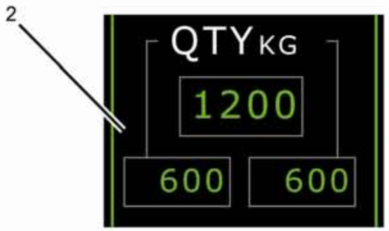
- the Electronic Engine Control (EEC) unit of each engine (because of no fuel flow-meters are installed on the fuel system)
- the Air Data System (ADS) which supplies the value of TAS (True Air Speed).



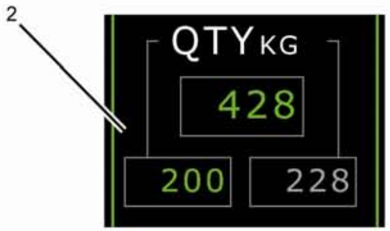
POWER PLANT FORMAT



FUEL PRESS INDICATION



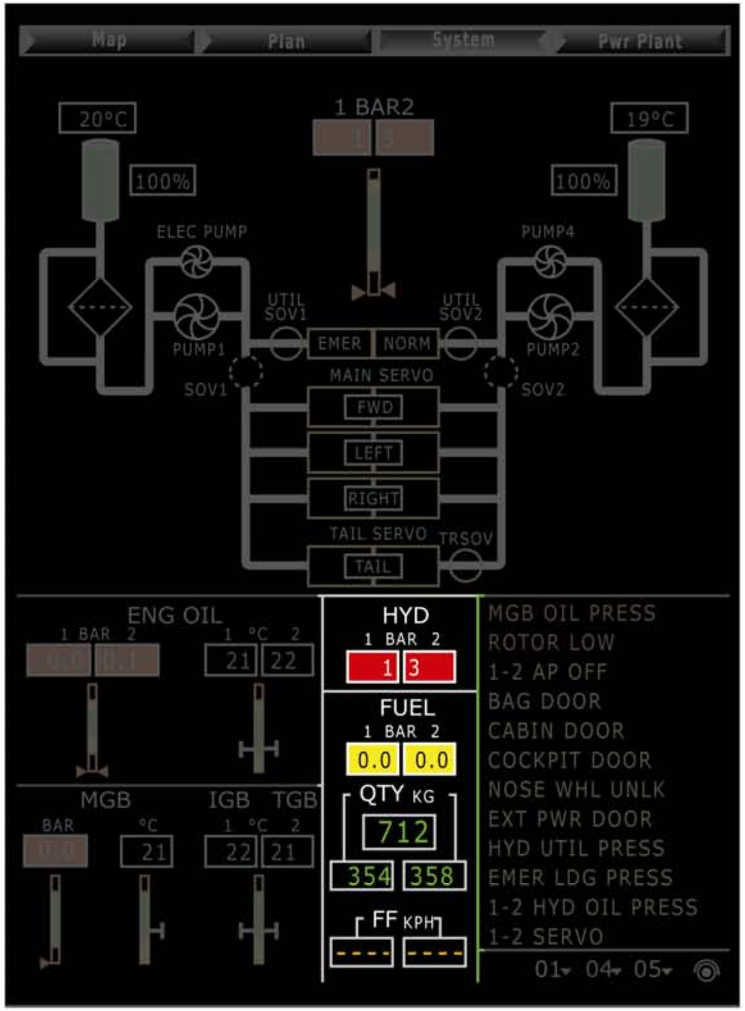
FUEL QUANTITY INDICATION
(NORMAL OPERATIONS)



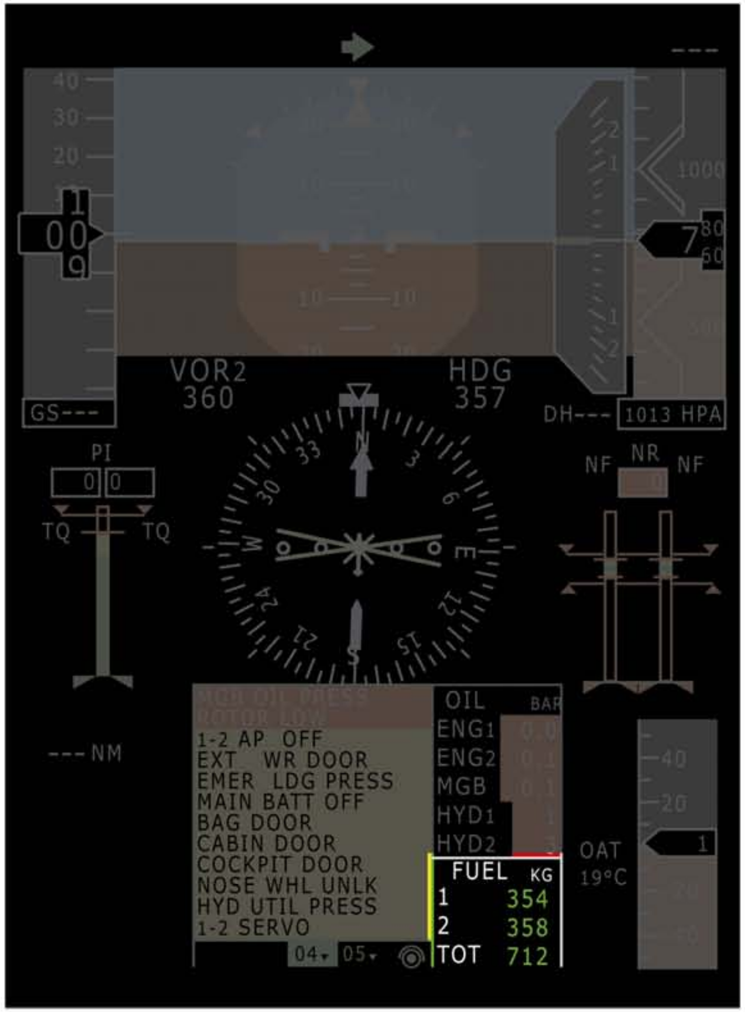
FUEL QUANTITY INDICATION
(RH TANK UNUSABLE FUEL)

FUEL FLOW INDICATION

FUEL INDICATING (1 OF 2)



HYDRAULIC SYNOPTIC FORMAT



COMPOSITE FORMAT

FUEL INDICATING (2 OF 2)

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FCU TEST

A manual test of the FCU can be operated (only on the ground) by selecting the push-button FUEL on the TEST control panel.

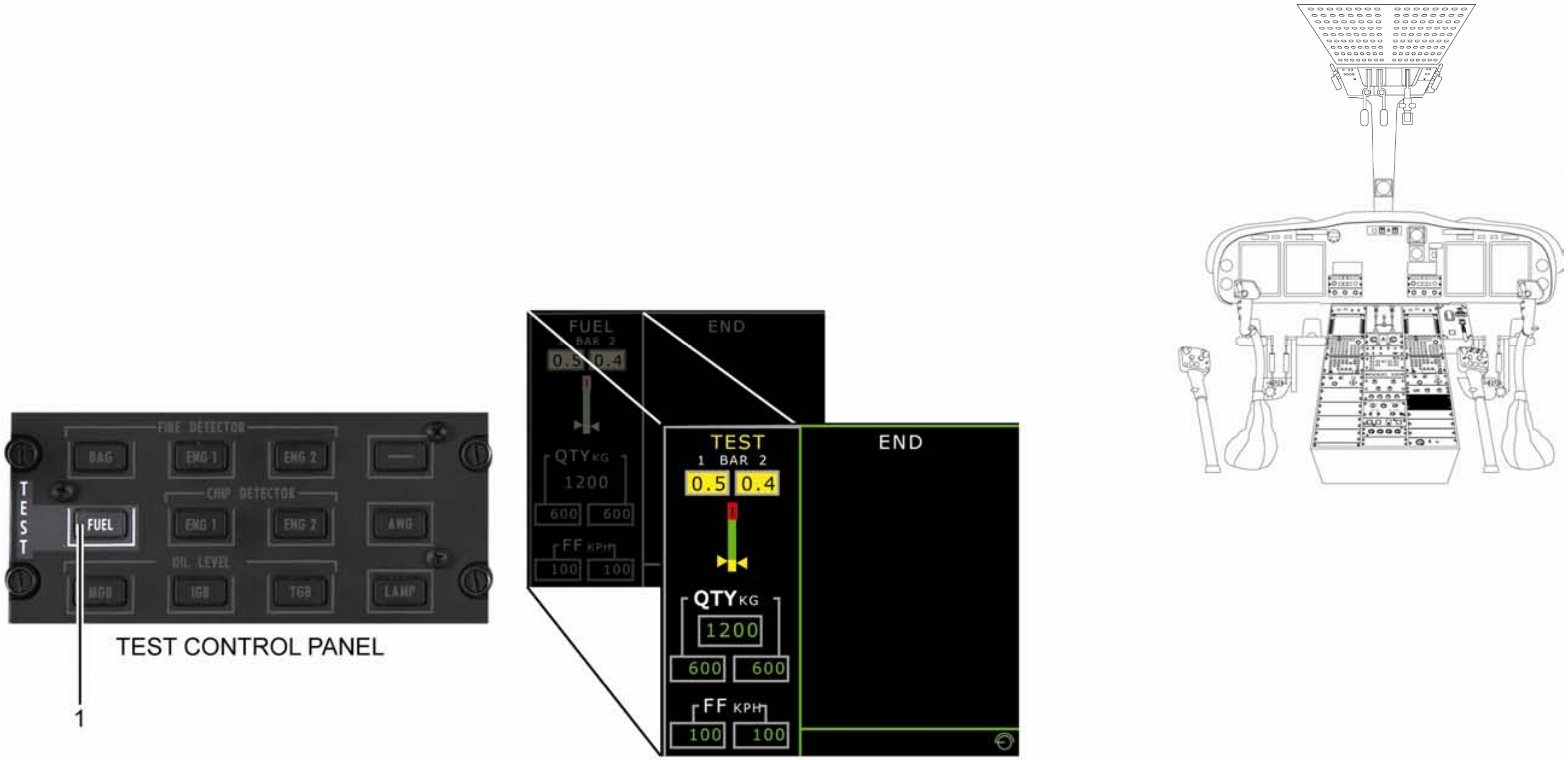
If the test result is unsuccessful, the caution 1 (2) FCU TEST FAIL is provided on the MFD.

1. FUEL push-button (on TEST control panel)

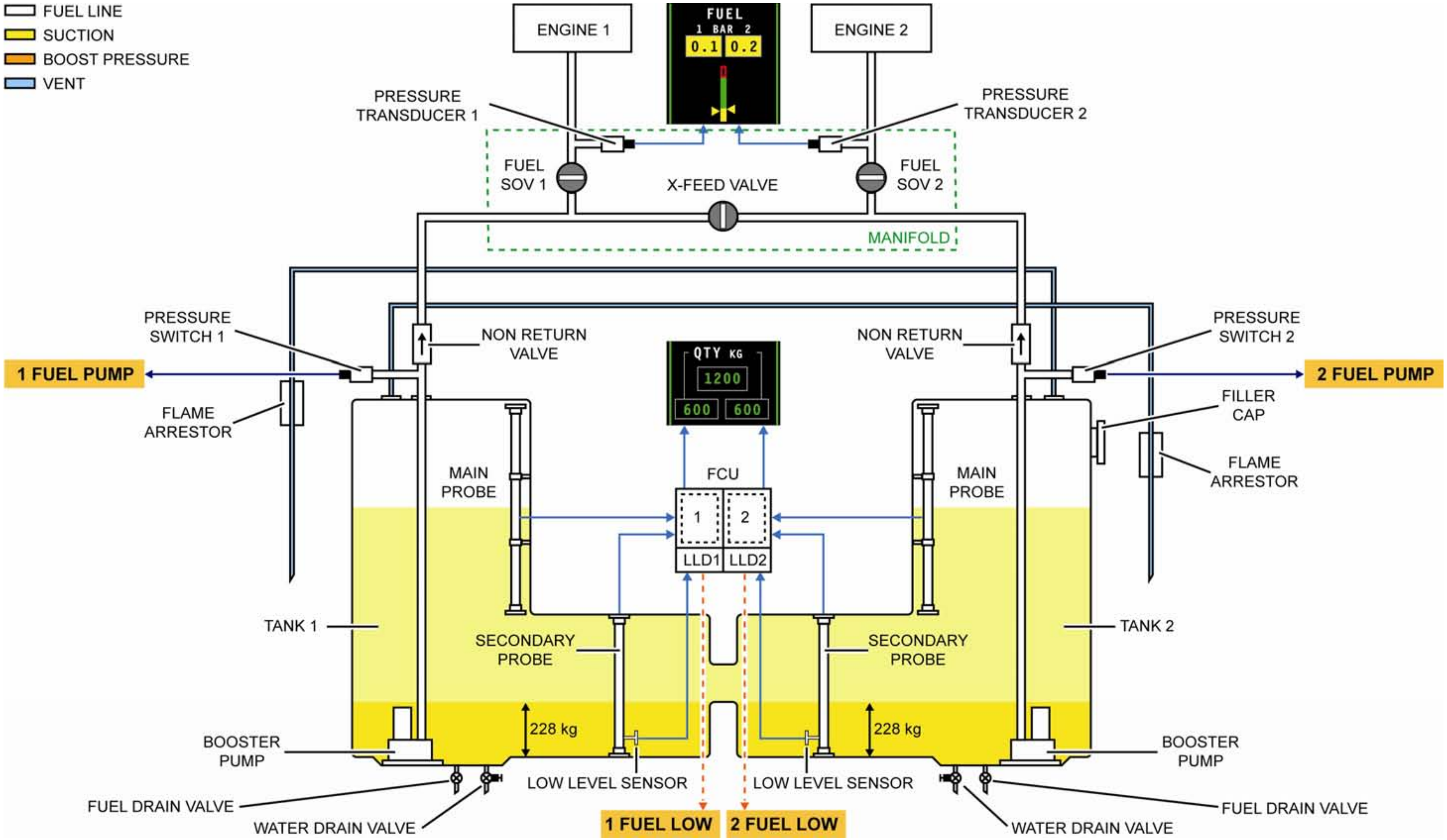
PRESSED …………. the white legend FUEL change to amber TEST and back to FUEL at the end of test

NOTE 1. The FUEL test can be performed ONLY ON THE GROUND.

NOTE 2. At power up the test is automatically performed and the amber legend TEST appears.


TEST
FIRE DETECTOR
BAG
ENG 1
ENG 2
CHIP DETECTOR
FUEL
ENG 1
ENG 2
AWG
OIL LEVEL
MGB
IGB
TGB
LAMP
TEST CONTROL PANEL
1
FUEL
BAR 2
0.5
0.4
QTY KG
1200
600
600
FF KPH
100
100
END
TEST
1 BAR 2
0.5
0.4
QTY KG
1200
600
600
FF KPH
100
100
END

FCU – TEST



FUEL SCHEMATIC

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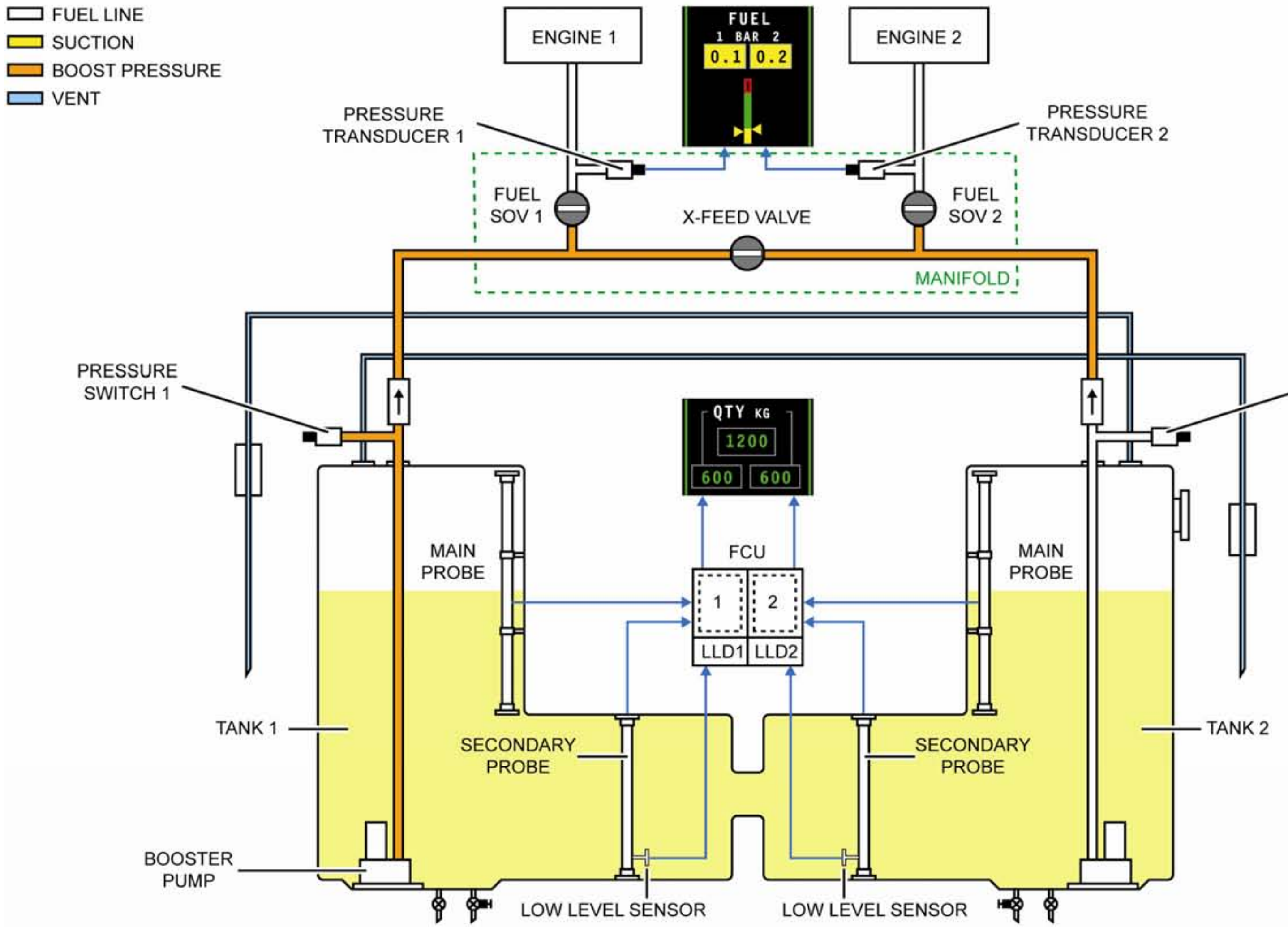
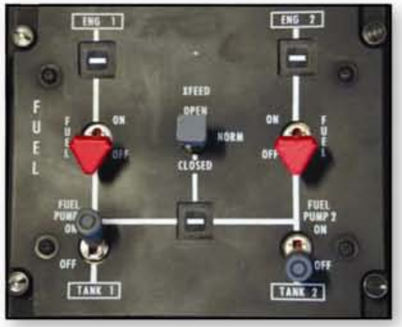
FUEL DISTRIBUTING – NORMAL OPERATION

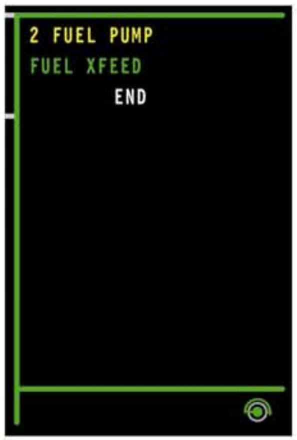
Booster pumps and fuel SOV are operated by a switch located on the FUEL control panel. In normal condition the booster pump in each tank provides a positive pressure fuel supply to the engines and the cross-feed valve is closed.

The fuel SOV controls the delivery of the fuel to the engines. When the fuel SOV is open and a positive pressure is detected, the pressure transducer gives the read-out of the fuel pressure on the MFD.

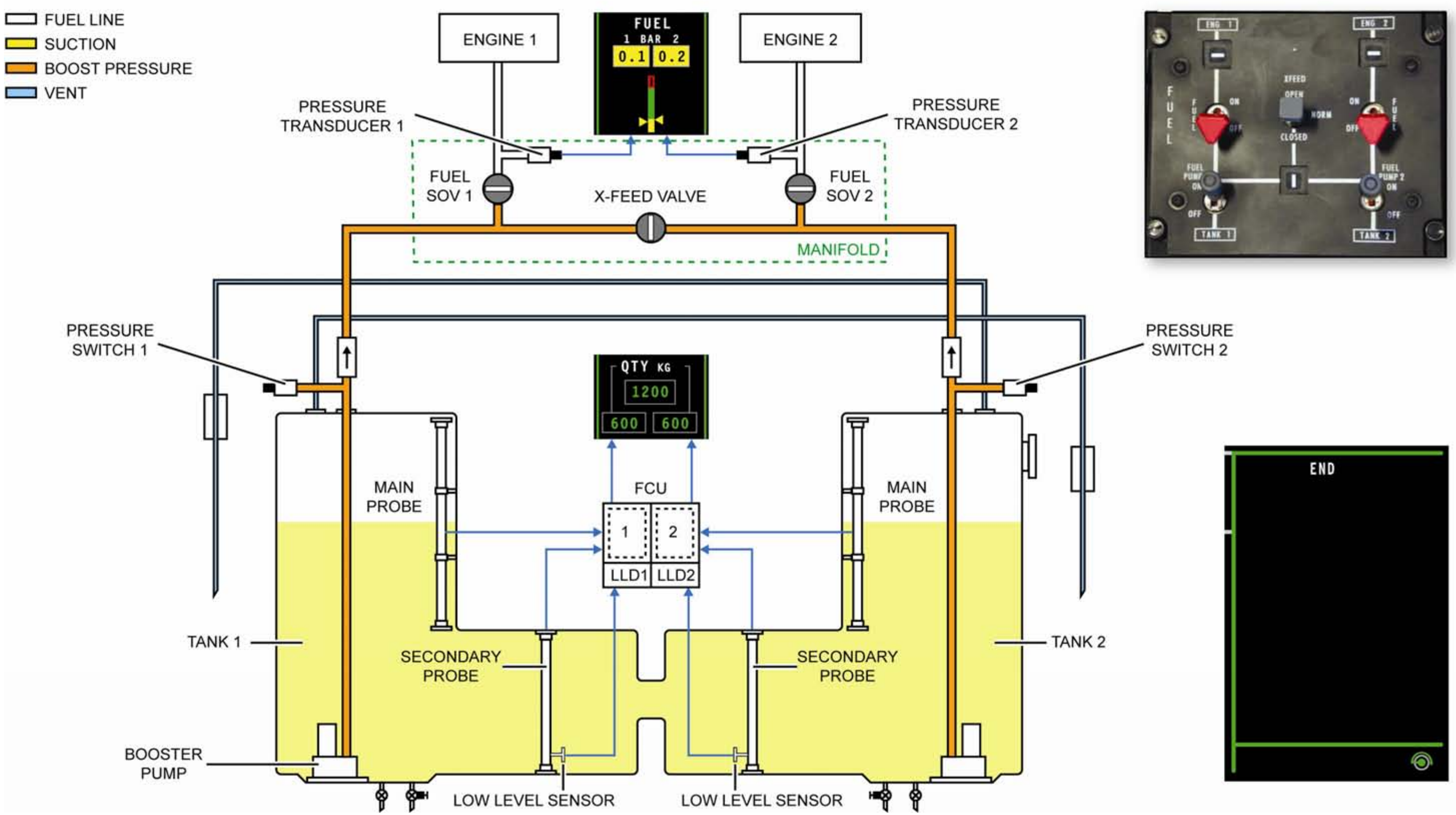
NOTE.
In the event of loss of fuel pressure from one tank due to a booster pump failure, the cross-feed valve automatically opens allowing one booster pump to feed both engines without operating limitations.



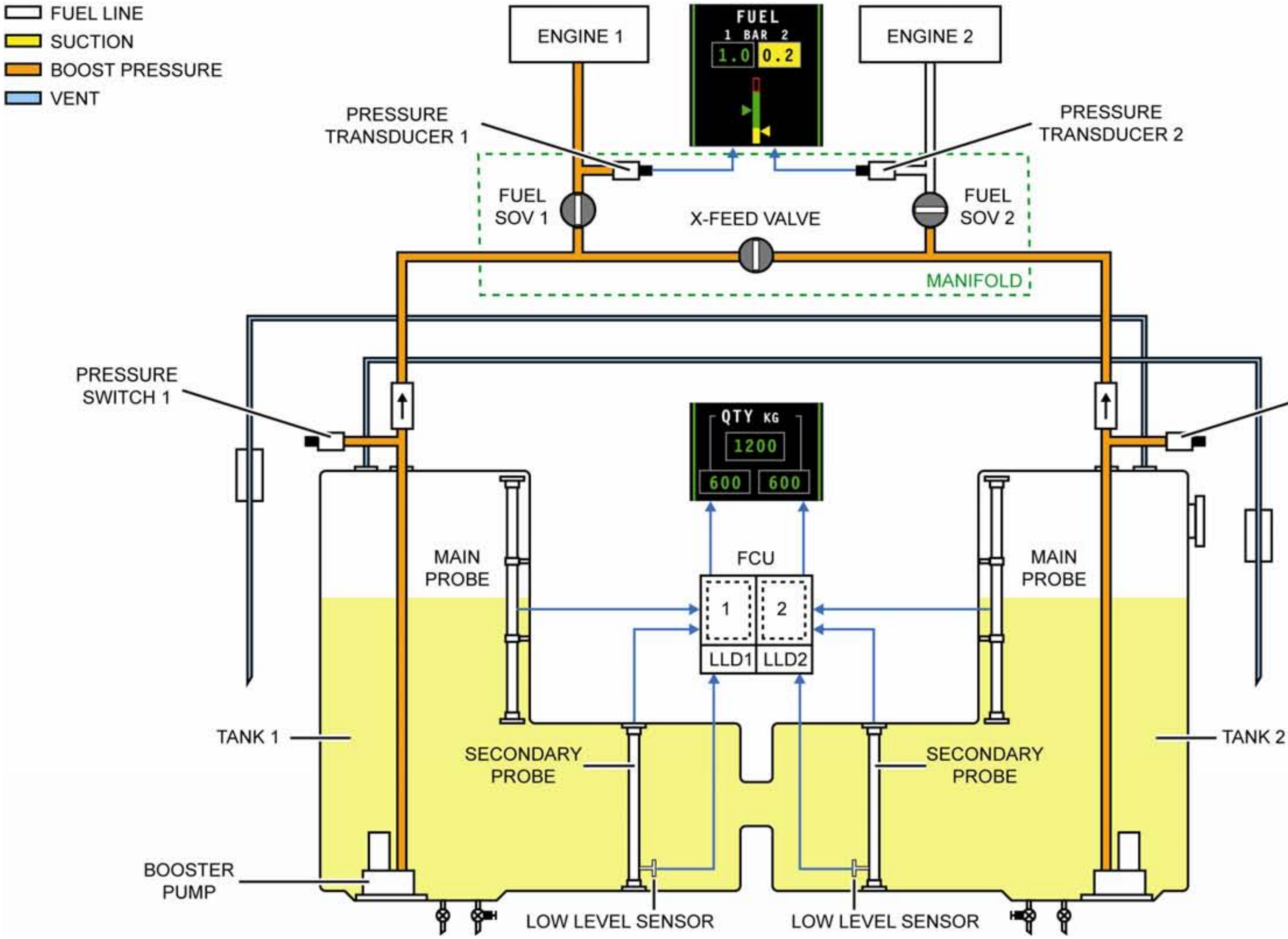


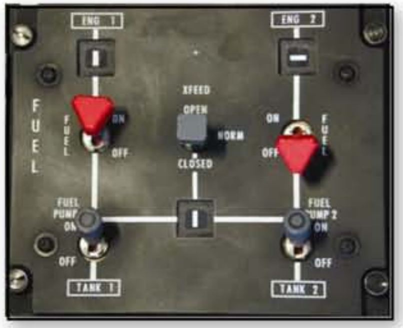


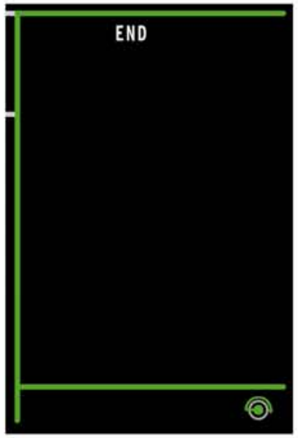
BOOSTER PUMP NO.1 ON



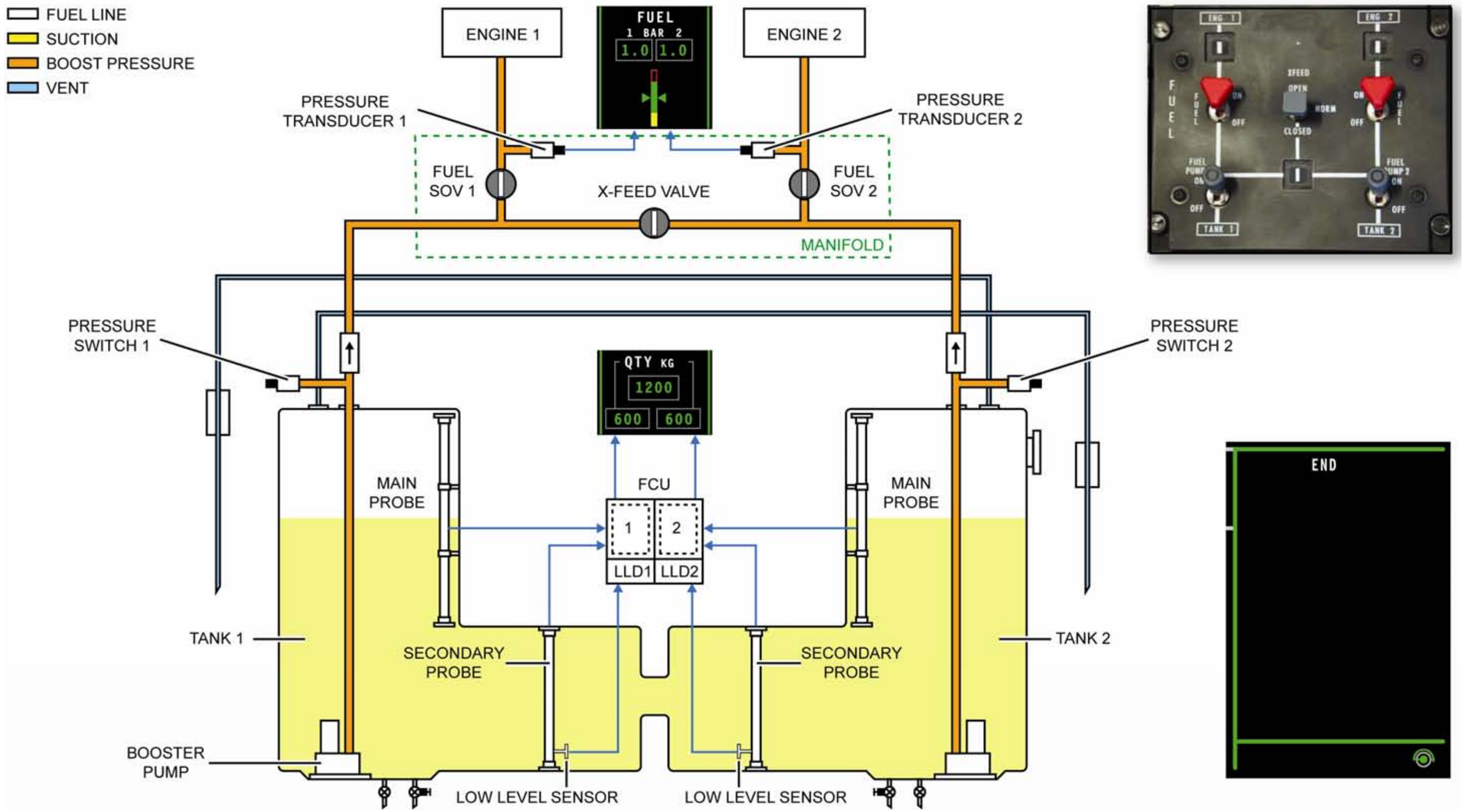
BOOSTER PUMP NO.1 ON AND NO.2 ON







BOOSTER PUMP NO.1 ON AND NO.2 ON AND FUEL SOV NO.1 ON



BOOSTER PUMP NO.1 AND NO.2 ON & FUEL SOV NO.1 AND NO.2 ON

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FUEL DISTRIBUTING – CROSSFEED CONDITION

The pilot can operate the cross-feed valve at any time setting the XFEED switch to OPEN, or CLOSED or NORMAL.

In normal operation the cross-feed valve is closed (switch set to NORM).

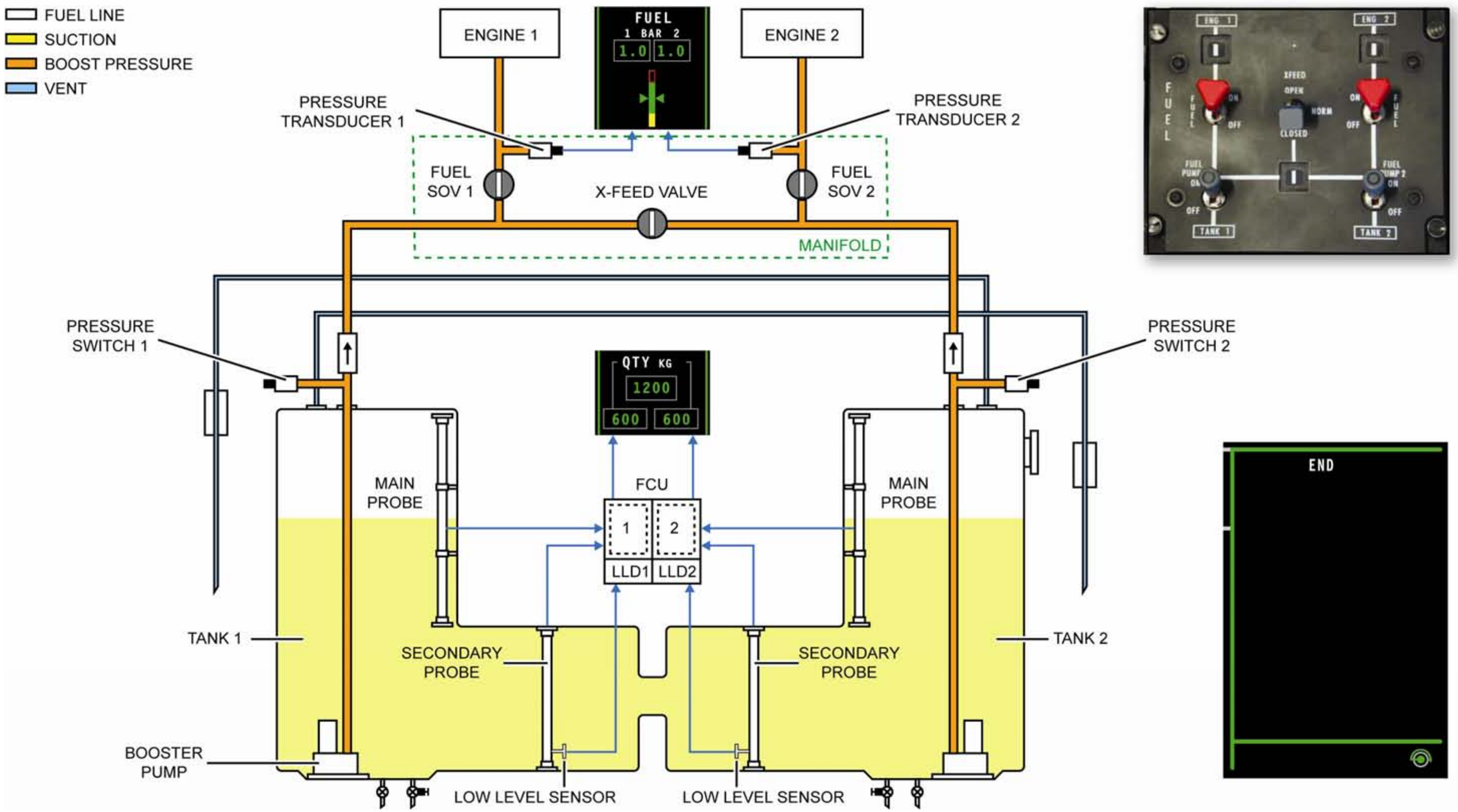
If the XFEED switch is set to NORM the cross-feed valve is automatically opened as a result of single fuel line pressure drop or booster pump failure. In this case the remaining booster pump supply fuel to both engines.

When the XFEED valve is open and pressure is restored in the failed supply line, the transducer installed in the fuel manifold provides indications of the restored pressure.

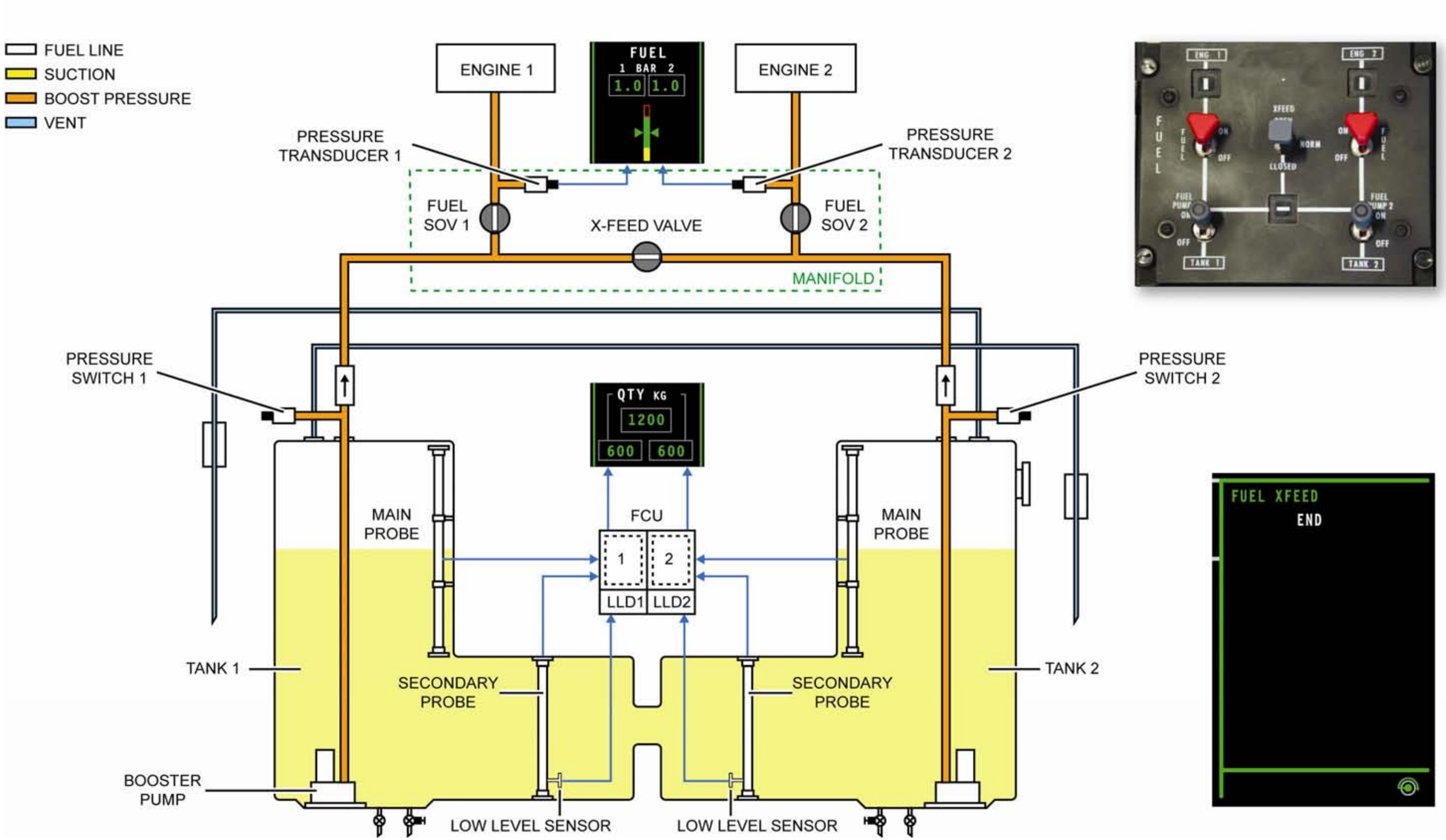
When the XFEED valve is open but the pressure in the failed supply line is not restored, possible fuel leak. Close manually the XFEED valve (SUCTION MODE). The pilot has to be attentive for sign of fuel leak or engine loss of power.

When cross feeding, the tank with pump off, NOT supplying the engines, will have a maximum quantity of unusable fuel of 228 kg (285 litres). This unusable fuel quantity value will change to grey to indicate the tank can no longer supply fuel.

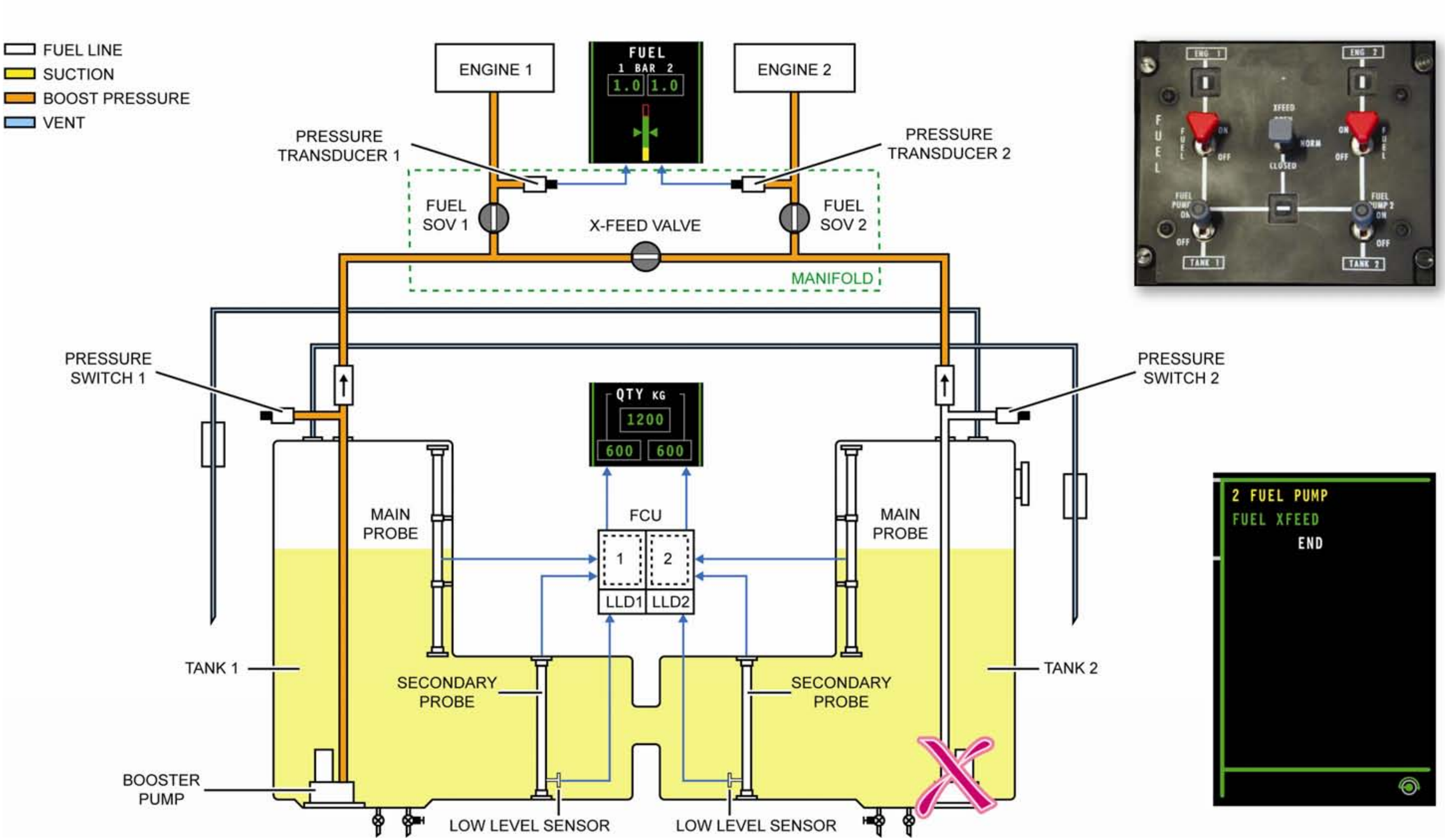
Close X-FEED to restore the availability of up to 228 kg of fuel (fuel level value returns to green). Engine operation, in suction mode, is assured and FUEL pressure, on MFD, is invalid displaying amber dashed. Avoid abrupt aircraft manoeuvres.



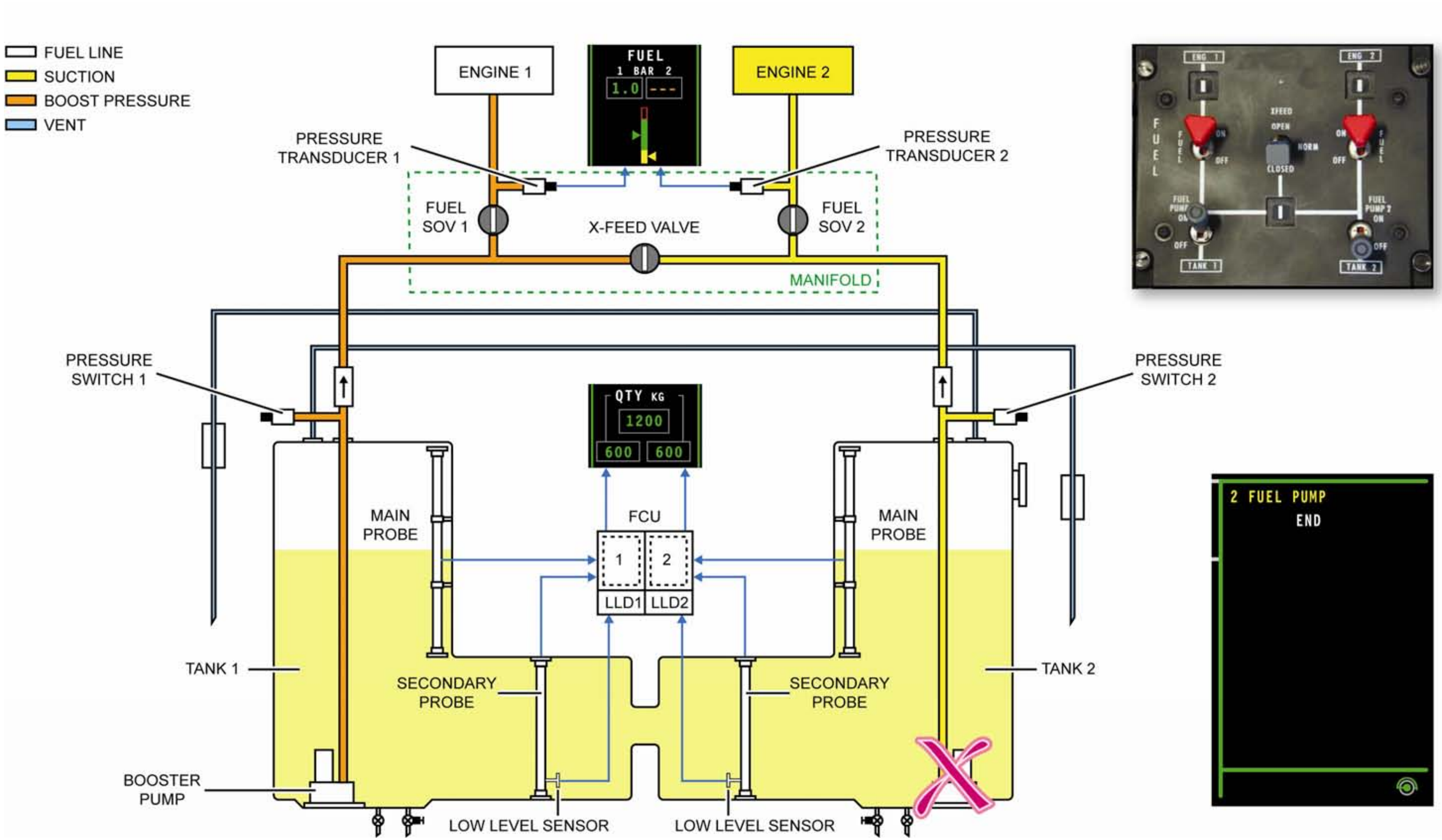
XFEED MANUALLY CLOSED



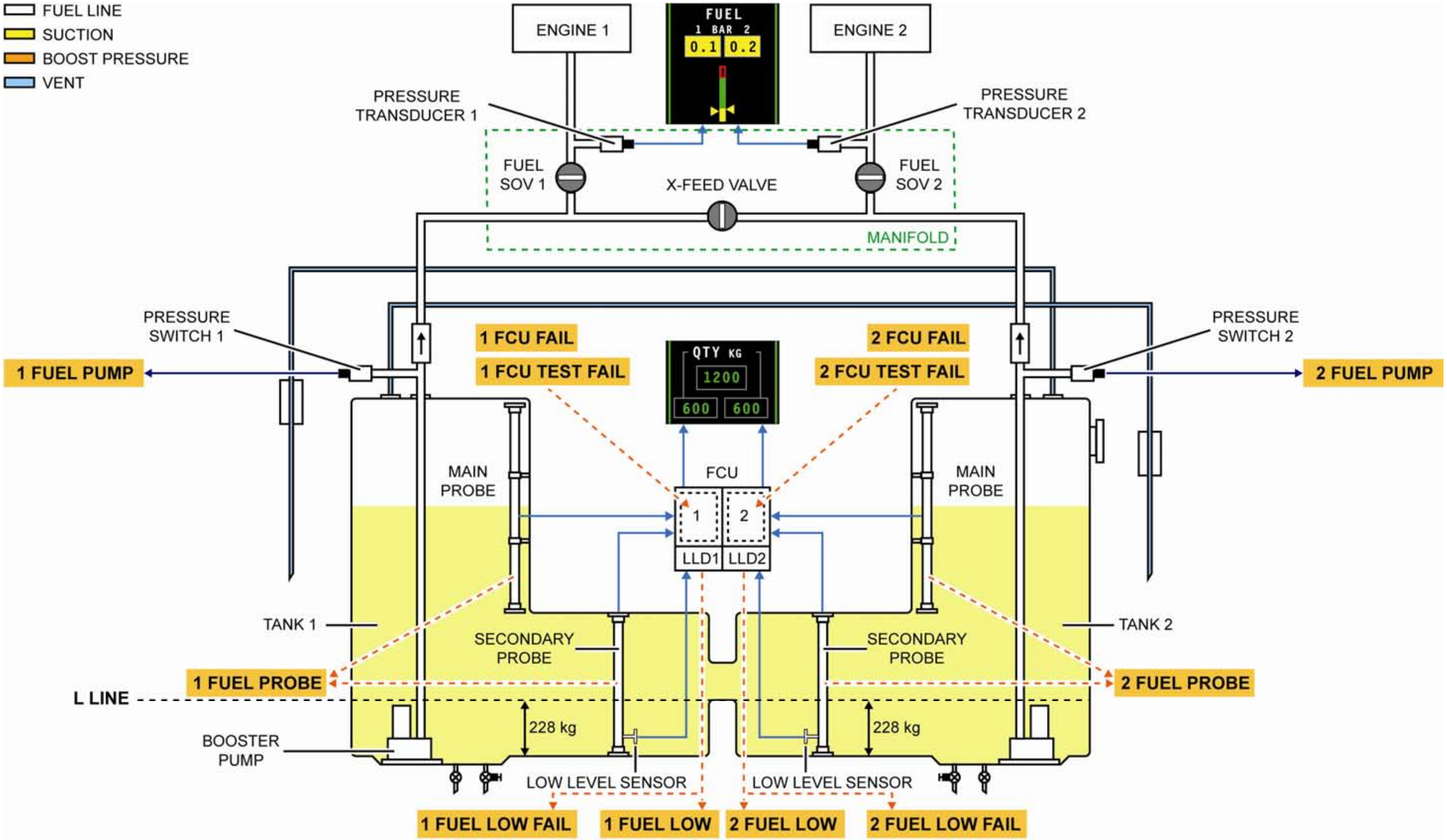
XFEED MANUALLY OPEN



XFEED AUTOMATICALLY OPEN (PRESSURE RESTORED)



XFEED MANUALLY CLOSED (SUCTION MODE FOR ENGINE NO.2)



CAUTION MESSAGES

CAS CAUTION MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
1(2) FUEL LOW	On affected tank fuel contents below 92 kg (112 litres) or below	FUEL LOW	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES FUEL SYSTEM
1(2) FUEL PUMP	Associated fuel pressure low (less than 0.6 bar) (goes out when above 0.7 bar)	FUEL PRESSURE LOW	
1-2 FUEL PUMP	Fuel pressure low (less than 0.6 bar) in both fuel systems	DOUBLE FUEL PUMP FAILURE	
	Monitor fuel quantity frequently. If an abnormal fuel consumption is confirmed, a fuel leakage may be present	ABNORMAL FUEL CONSUMPTION	
1(2) FCU FAIL	Associated FCU failure and possible loss or degradation of fuel contents indication	FUEL CONTENTS GAUGING UNIT FAILURE	
1(2) FUEL LOW FAIL	Associated fuel low sensor failure	FUEL LOW SENSOR FAILURE	
1(2) FUEL PROBE	Associated fuel probe failure and degradation of fuel contents indication	FUEL PROBE FAILURE	
1(2) FCU TEST FAIL	Associated fuel contents unit test system failed (only active on the ground)	FUEL CONTENTS GAUGING UNIT TEST SYSTEM FAILURE	

CAS ADVISORY MESSAGES

CAS CAPTION	MESSAGE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
FUEL XFEED	Fuel crossfeed valve open both manually or automatically		Section 2 NORMAL PROCEDURES

FUEL - LIMITATIONS

Refer to AW139-RFM-4D for limitations of fuel system. Refer to AW139-RFM-4D for authorized fuel compliant to Pratt and Whitney PT6C-67C engines.

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CHAPTER 29

HYDRAULIC POWER

SECTION 00 – GENERAL

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HYDRAULIC POWER - GENERAL

The purpose of the hydraulic power system is to supply the hydraulic power necessary to operate:

- the flight control circuit (main rotor and tail rotor servo-actuators)
- the utility circuit (landing gear extension and retraction system)

The hydraulic power system is constituted by two independent circuits, the system no.1 and the system no.2, that supply hydraulic fluid at a nominal pressure of 207 bar (3000 psi).

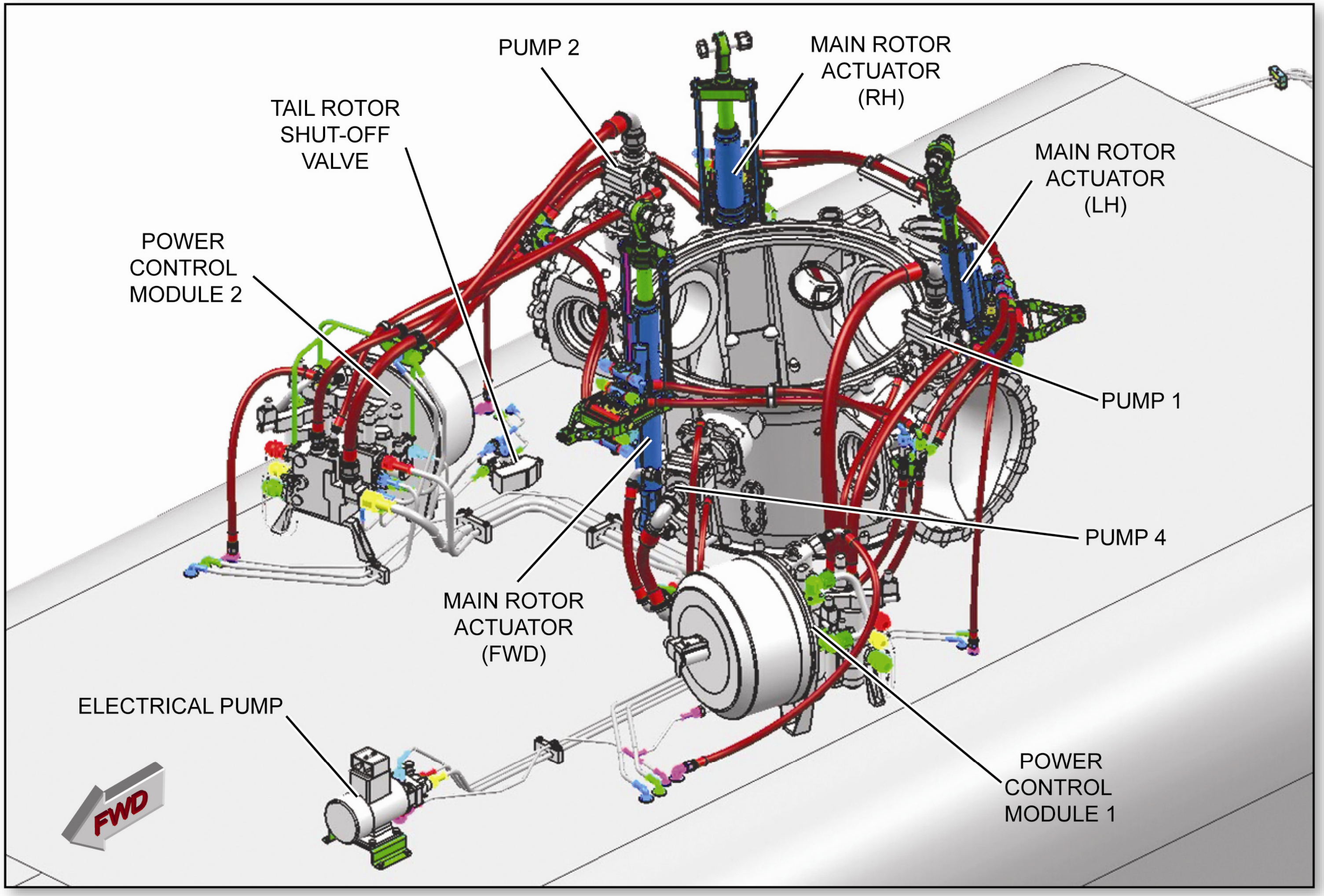
Both systems supply hydraulic power to the flight controls.

The system no.1 is used to operate the landing gear only in emergency situations.

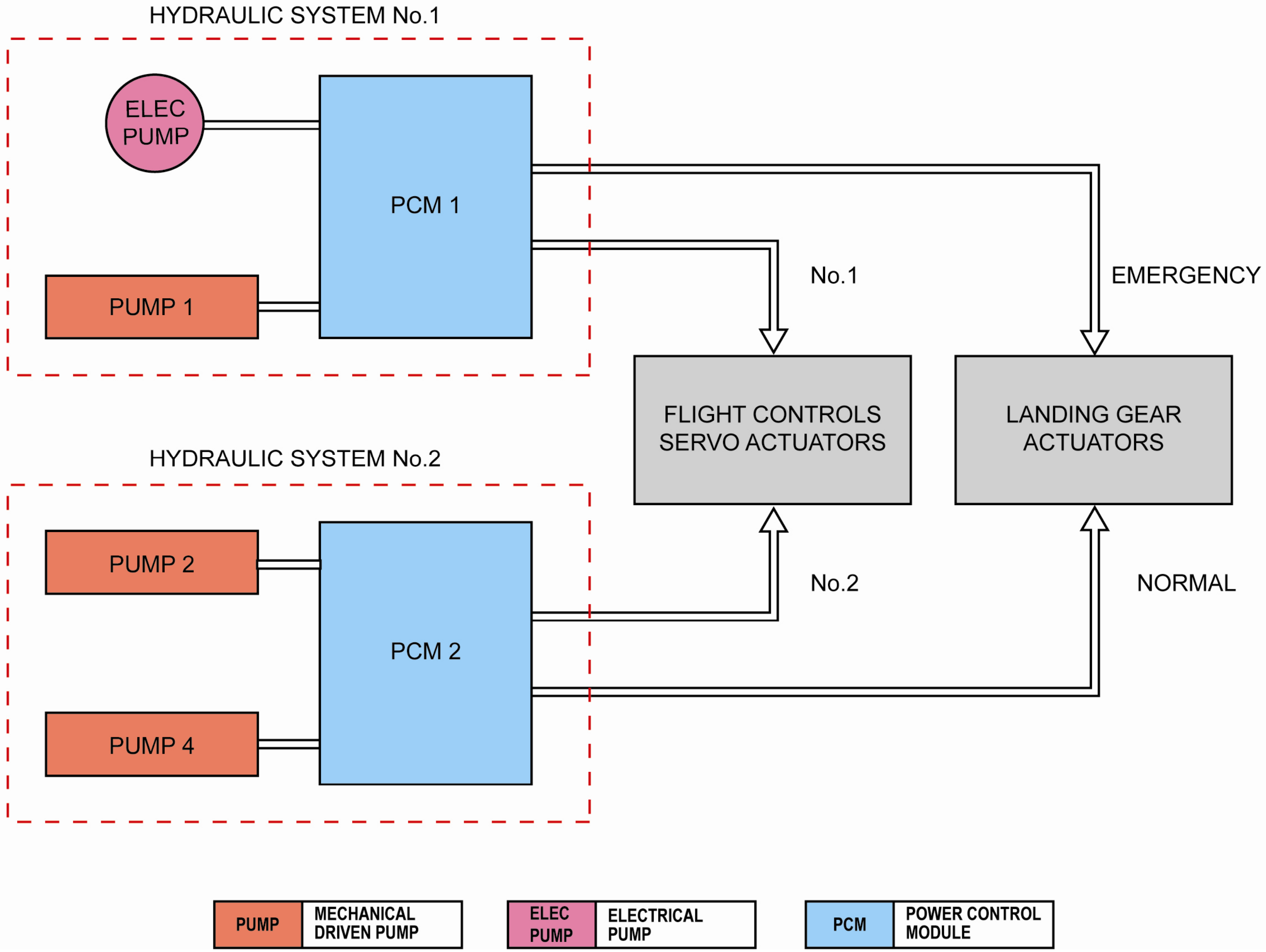
The system no.2 is used to operate the landing gear in normal conditions and is provided with a Tail Rotor Shut-Off Valve (TRSOV) to prevent a possible fluid leakage.

In the system no.1 an electrical pump can supply hydraulic fluid at reduced pressure for pre-flight check on ground only.

The HYD control panel shows over temperature and over pressure of the hydraulic system and allows operating on the system itself.



HYDRAULIC POWER – GENERAL LAYOUT



HYDRAULIC POWER – BLOCK DIAGRAM

HYDRAULIC SYSTEM NO.1 - MAIN COMPONENTS

POWER CONTROL MODULE NO.1 (PCM 1)

The Power Control Module 1 (PCM 1) is an integrated assembly which permits to store and distribute the hydraulic fluid in the hydraulic system no.1. It is equipped with different sensors that provide control and monitoring of the hydraulic fluid. The PCM 1 is connected to the main rotor and tail rotor servo actuators and to the Landing Gear Control Valve (LGCV).

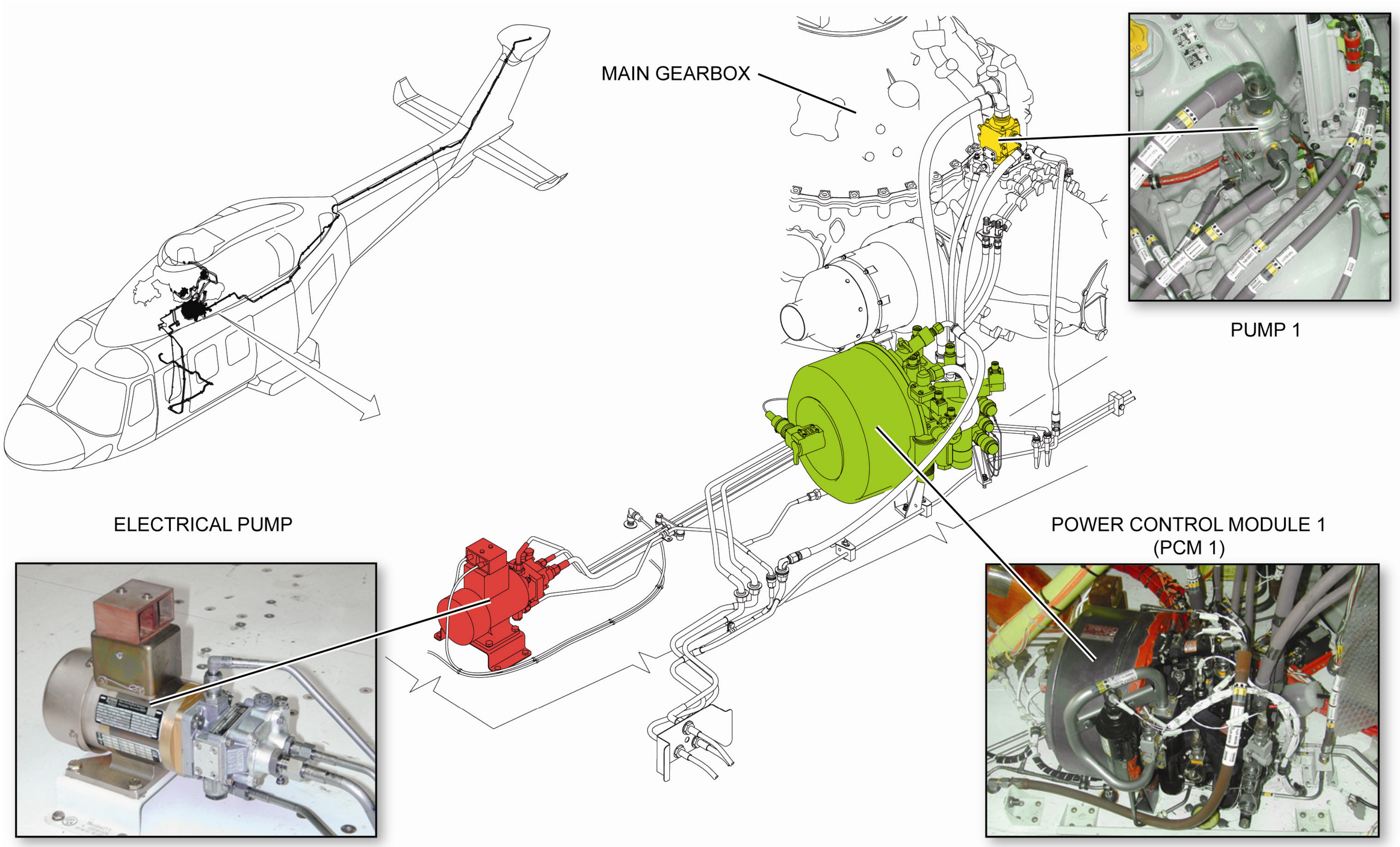
PUMP1

The pump 1 is a mechanical pump driven by the Main Gear Box (MGB) that keeps a constant pressure output of 207 bar (3000 psi) with a variable flow rate in the hydraulic system no.1.

ELECTRICAL PUMP (EP)

The electrical pump is powered by the aircraft battery. The pump is equipped with a 2 minutes timer relay used to save battery power.
The electrical pump supplies hydraulic fluid at reduced pressure (1523 psi or 105 bars) and allows to operate the main rotor and tail rotor actuators during the pre-flight check on ground only.

NOTE. In this conditions, the reaction of all the flight controls are slower than in normal conditions.



HYDRAULIC SYSTEM NO.1 – MAIN COMPONENTS

HYDRAULIC SYSTEM NO.2 – MAIN COMPONENTS

POWER CONTROL MODULE NO.2 (PCM 2)

The Power Control Module 2 (PCM 2) is an integrated assembly which permits to store and distribute the hydraulic fluid in the hydraulic system no.2. It is equipped with different sensors that provide control and monitoring of the hydraulic fluid. The PCM 2 is connected to the main rotor and tail rotor servo actuators and to the Landing Gear Control Valve (LGCV).

PUMP 2 AND PUMP 4

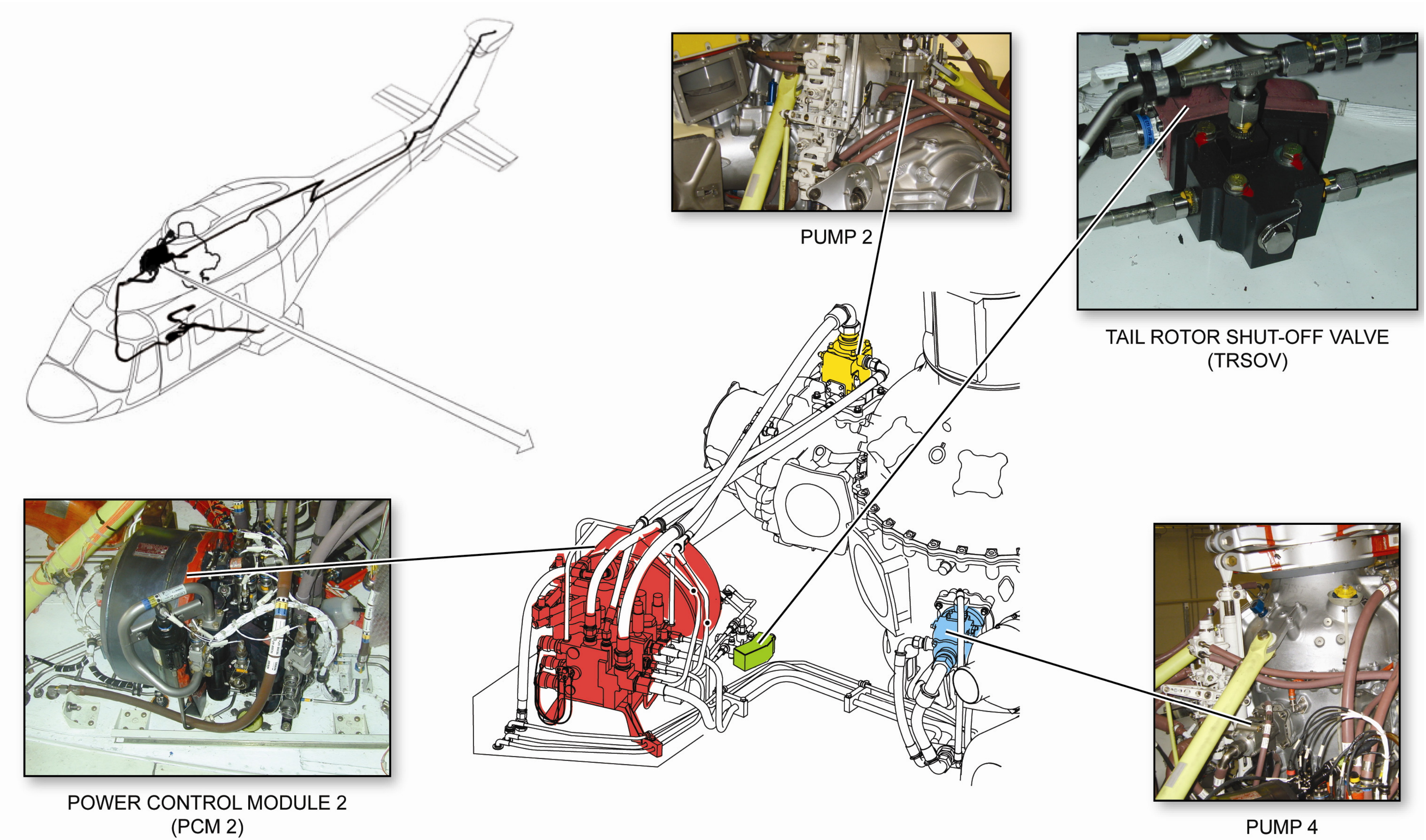
The pump 2 and the pump 4 are two identical mechanical pumps driven by the MGB that keep a constant pressure output of 207 bar (3000 psi) with a variable flow rate in the hydraulic system no.2. As a safety measure, the two pumps are driven separately by the MGB.

TAIL ROTOR SHUT-OFF VALVE (TRSOV)

The Tail Rotor Shut-Off Valve is installed on the supply line of the TR actuator of the system no.2. It automatically cut off the supply of hydraulic fluid to the TR actuator in case the fluid falls below the minimum level.

SHUT-OFF VALVE NO.2 (SOV2)

The hydraulic system no.2 includes a Shut-Off Valves (SOV) that can be closed to isolate the flight control circuit in case of system overheating.



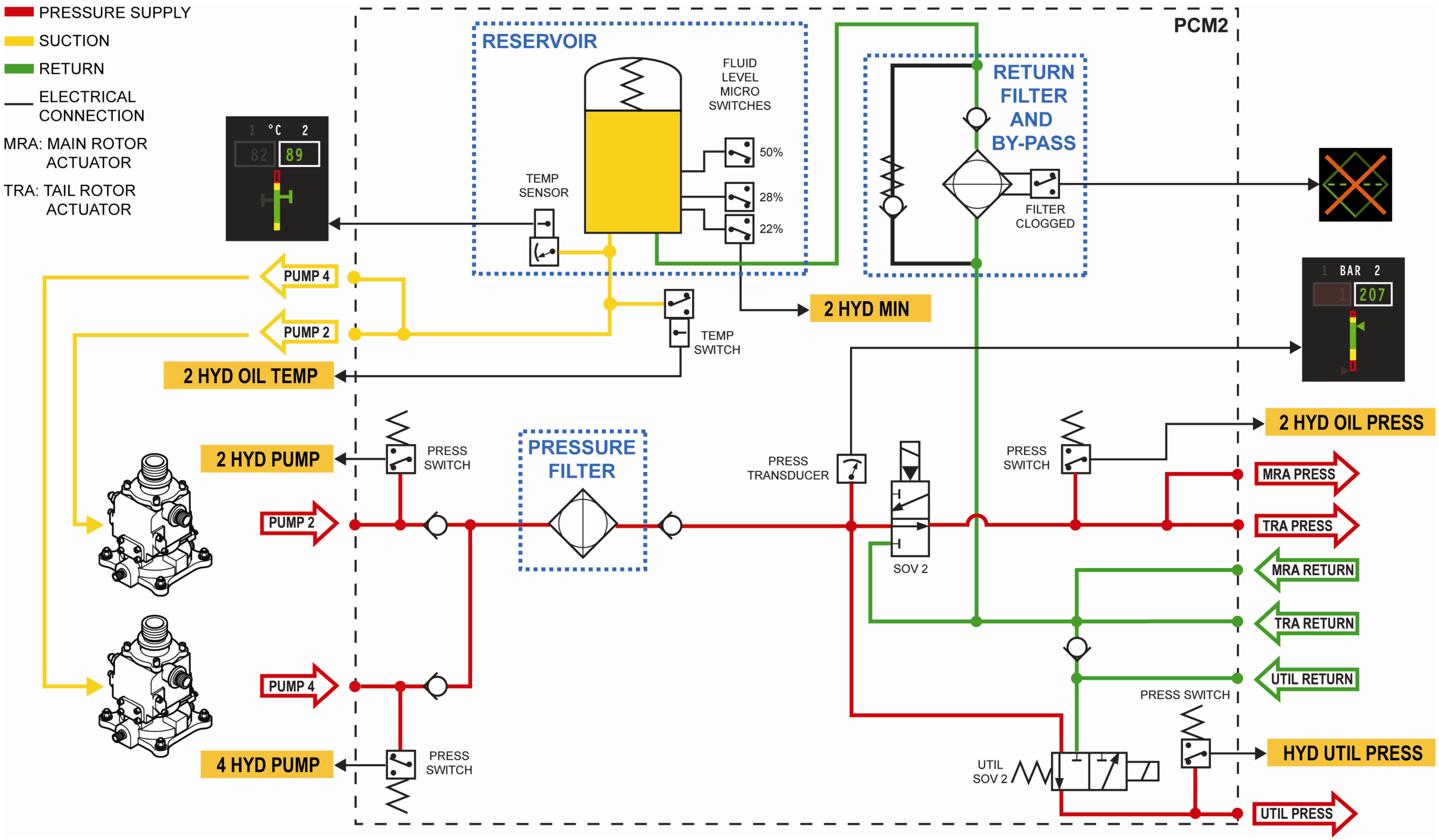
HYDRAULIC SYSTEM NO.2 – MAIN COMPONENTS

POWER CONTROL MODULE – GENERAL

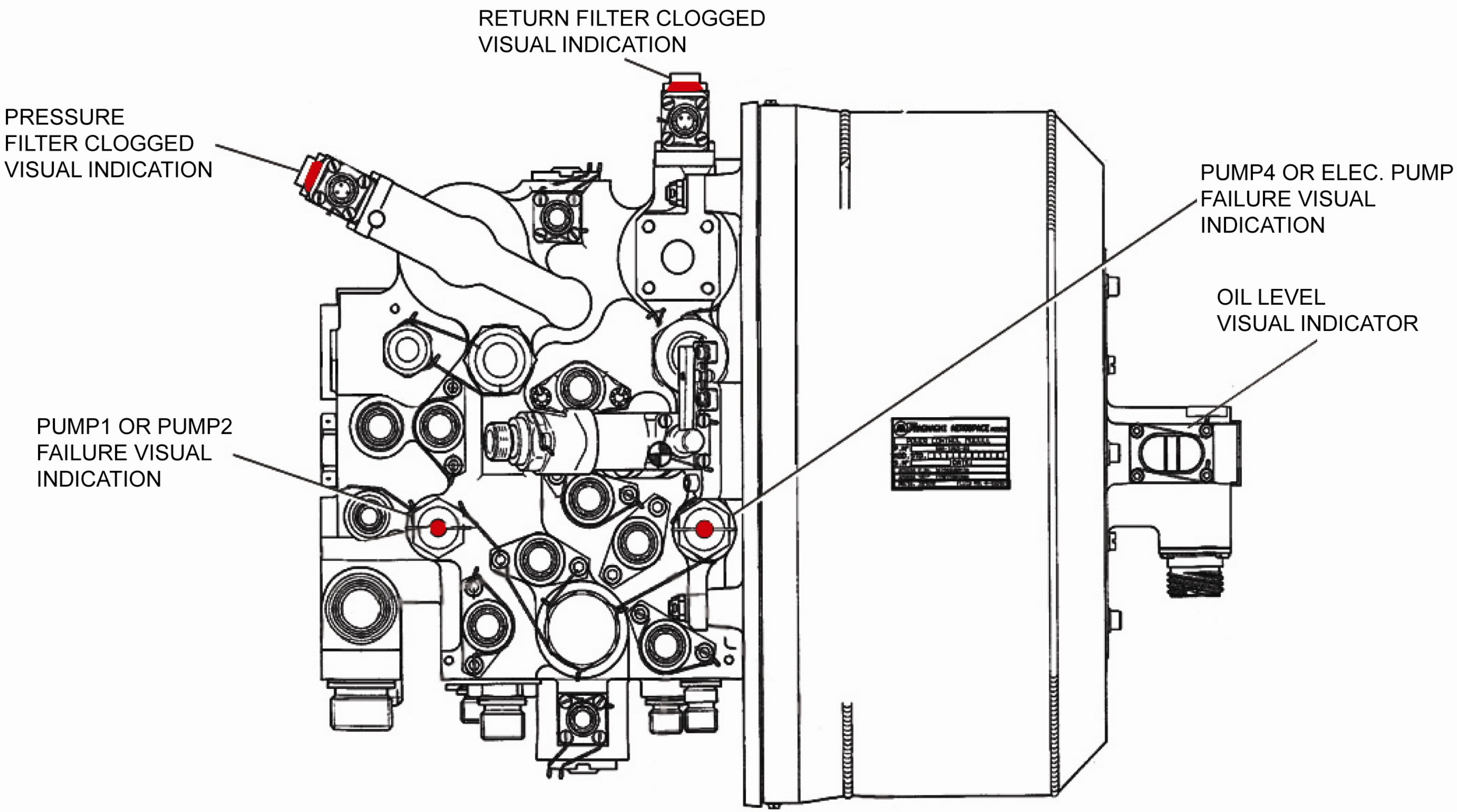
Each one of the two Power Control Module (PCM) consists of the following main components:

- a reservoir inside which the hydraulic fluid is separated from the air
- three fluid level microswitches
- a visual hydraulic fluid level indicator located on the back side of reservoir
- one pressure and one return filter assembly which provides filtration of the hydraulic fluid. Only the return filter is provided with a by-pass valve to allow hydraulic fluid recirculation in case of filter clogging. Return filter clogged condition is shown on the hydraulic synoptic
- one filter clogged visual indicator for each filter. The mechanical indicator (RED POP-OUT) once actuated remains extended until reset manually (position hidden from view)
- the Flight Controls (FC) circuit shut-off valves (identify as SOV1 and SOV2 on the hydraulic synoptic) permit to isolate the flight controls from the hydraulic circuit in case of system overheating
- the utility circuit shut-off valves (identify as UTIL SOV1 and UTIL SOV2 on the hydraulic synoptic) permit to isolate the utilities from the hydraulic circuit in case of leakage. These valves are automatically controlled by the fluid level microswitches inside the reservoir
- pressure sensors and pressure switches
- pump suction, pressure and drain lines ports
- flight controls pressure and return circuit lines connection ports
- pressure and return ground equipment connection ports

PCM1 supplies hydraulic pressure to the system 1 while the PCM2 supplies pressure to the system 2. The next figure schematizes the PCM 2.



POWER CONTROL MODULE – SCHEMATIC



POWER CONTROL MODULE – VISUAL INDICATIONS

HYDRAULIC POWER – INDICATIONS

The PWR PLANT PAGE displays the values of pressure and temperature in the HYD area.

1. PRESSURE

   The values of pressure in the systems no.1 and no.2 are represented by digital readouts under the label 1 BAR 2 respectively. Graphically these values are represented on a vertical scale by means of two pointers (triangles) that match the color of the area on the scale.

   The digital readouts and the pointers are displayed red when the pointer is in the red zone (warning), amber when the pointer is in the amber zone (caution) and green in normal conditions.

2. TEMPERATURE

   The fluid temperature values in the systems no.1 and no.2 are represented by digital readouts under the label

   1 °C 2. Graphically these values are represented on a vertical scale by means of two pointers (T symbols) that match the color of the area on the scale.

   Green band of the analogue vertical scale represents a normal condition for the hydraulic fluid temperature and so the associated digital readout values in Celsius degrees.

   The amber band represents a caution condition while the red band is associated to a warning condition.

The COMPOSITE FORMAT displays the values of pressure.

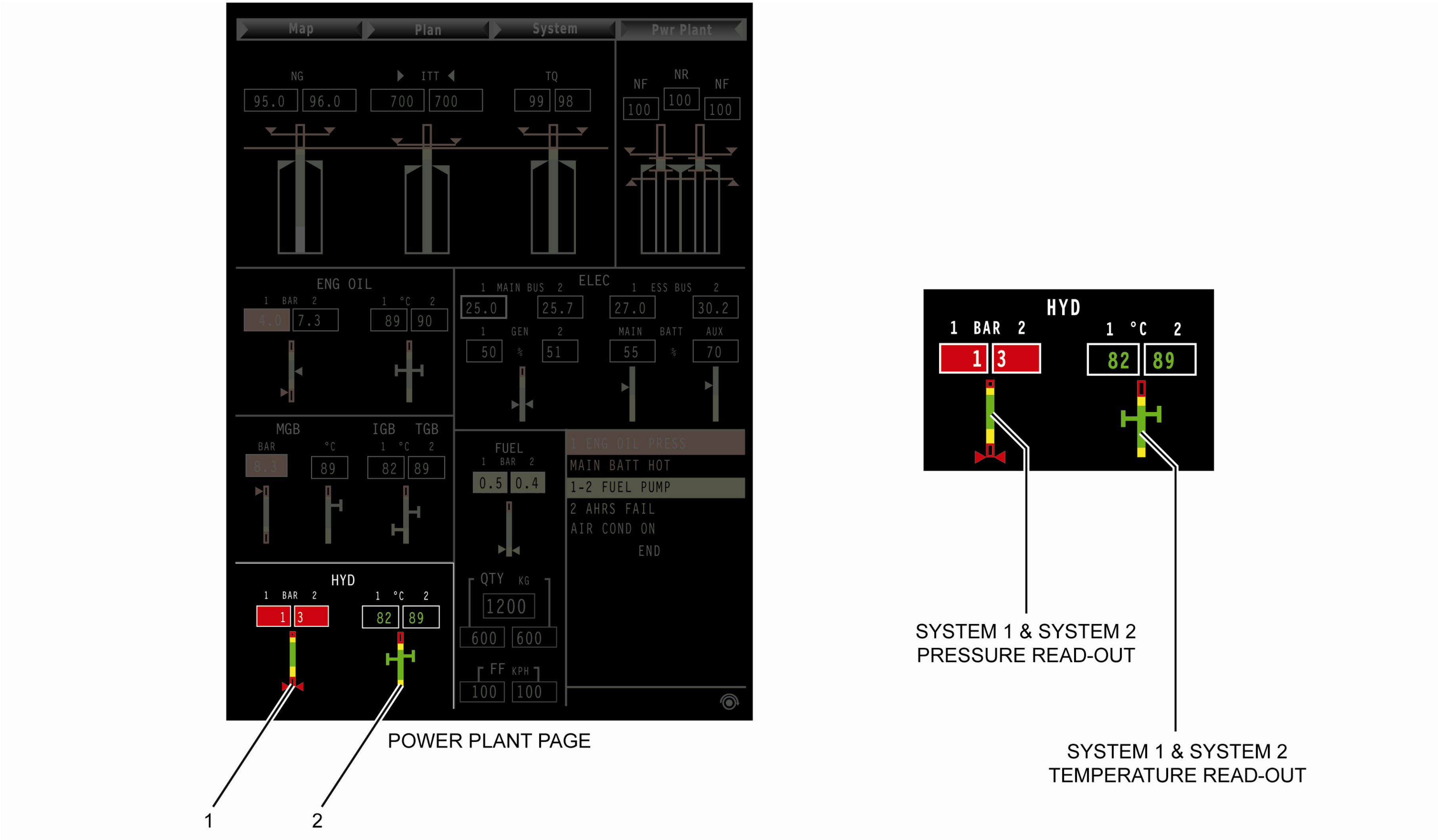
3. PRESSURE

   The values of pressure in the systems no.1 and no.2 are represented by digital readouts aside the label HYD 1 and HYD 2.

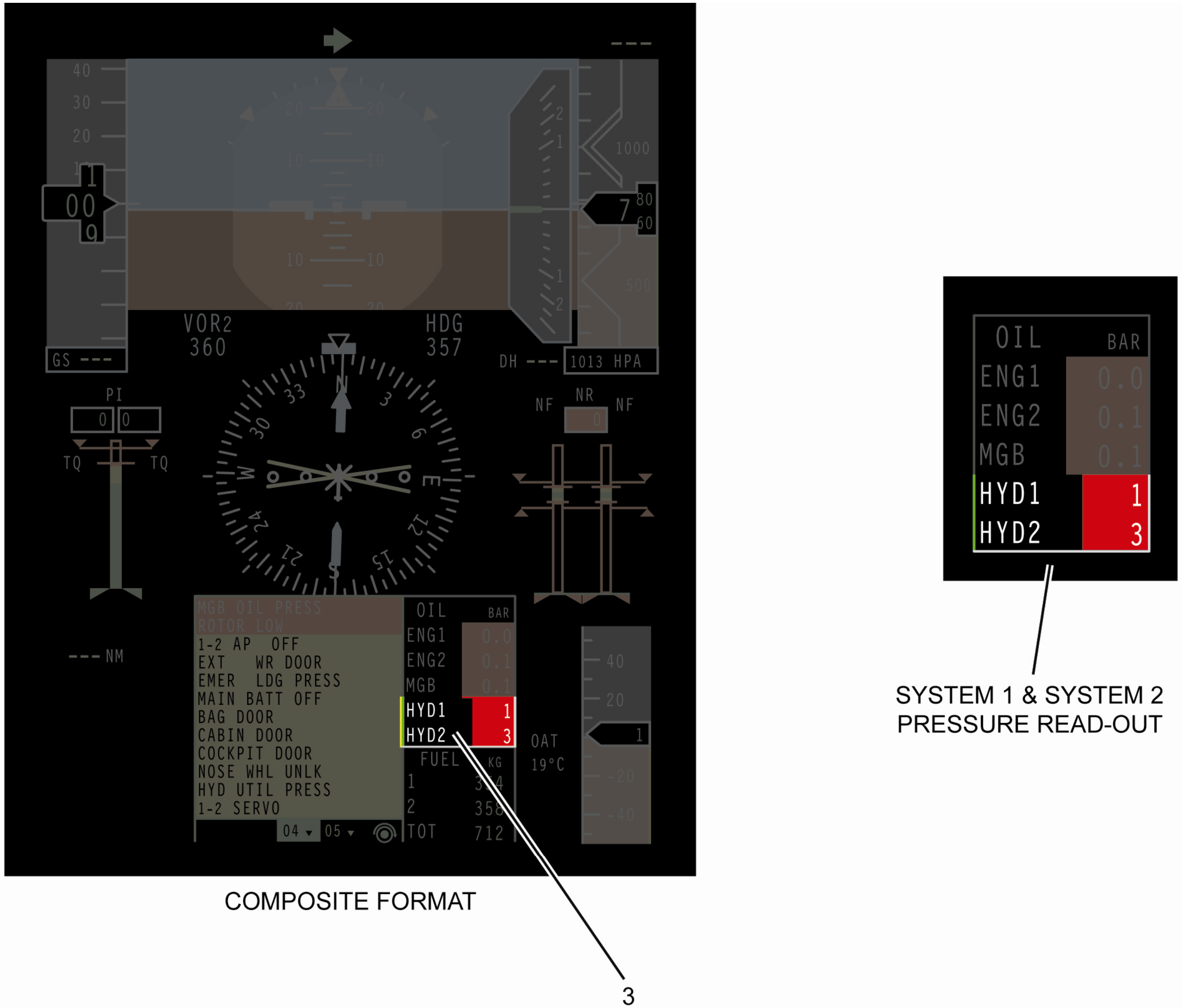
The generic synoptic format page displays the values of pressure in the HYD area.

4. PRESSURE

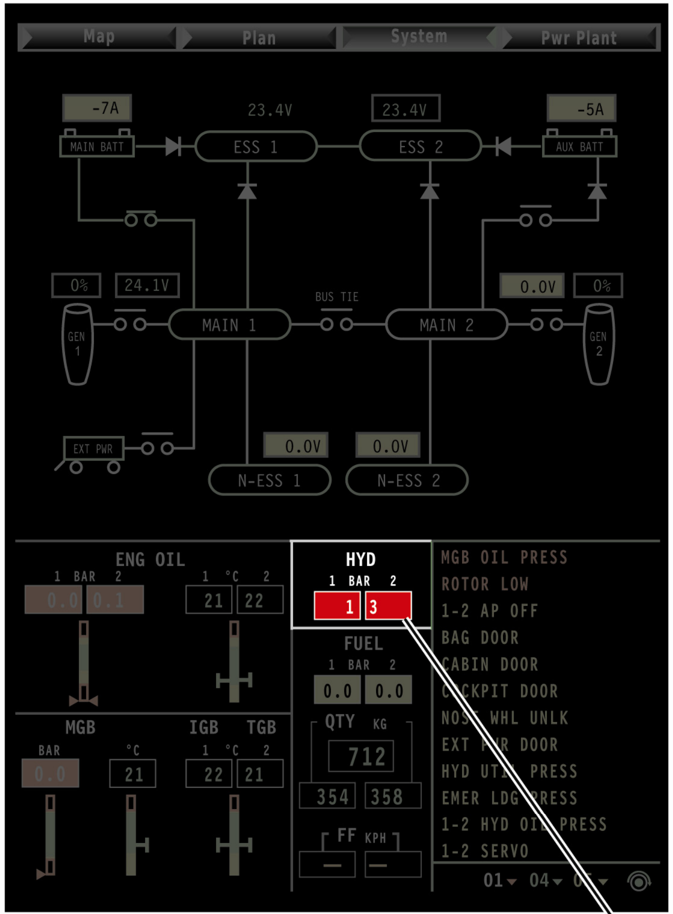
   The values of pressure in the systems no.1 and no.2 are represented by digital readouts inside the box 1 and 2 respectively.



HYDRAULIC POWER – INDICATIONS (1 OF 4)

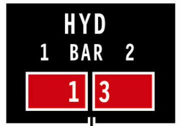


HYDRAULIC POWER – INDICATIONS (2 OF 4)



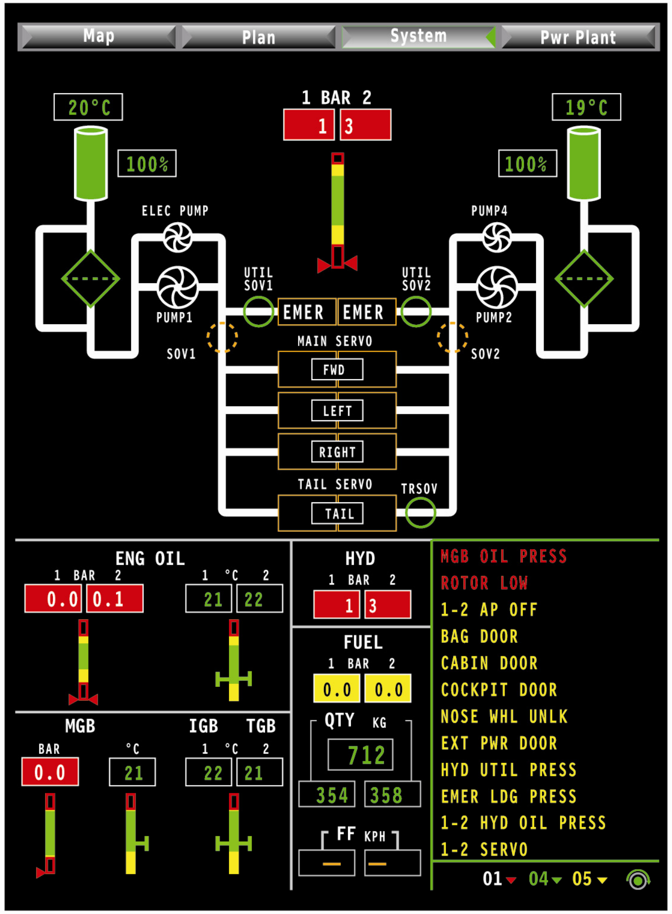
ELECTRICAL SYNOPTIC FORMAT

4



SYSTEM 1 & SYSTEM 2
PRESSURE READ-OUT

HYDRAULIC POWER – INDICATIONS (3 OF 4)



HYDRAULIC SYNOPTIC FORMAT

HYDRAULIC POWER – INDICATIONS (4 OF 4)

HYDRAULIC POWER – CONTROLS AND INDICATIONS

1. ELEC PUMP pushbutton

ON (green) lighted ..………	the electrical pump is operating for a maximum time of two minutes (controlled by the time relay)

2. HYD 1 indicator

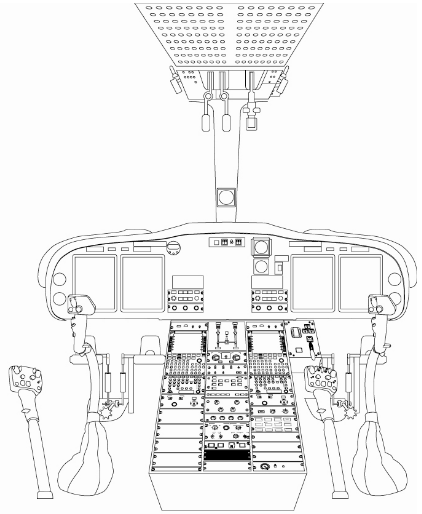
PRESS (amber) lighted …...	the hydraulic pressure in the system no.1 drops below 163 bar
TEMP (amber) lighted ….....	the temperature of the fluid in the system no.1 is above 134°C

3. SOV guarded switch

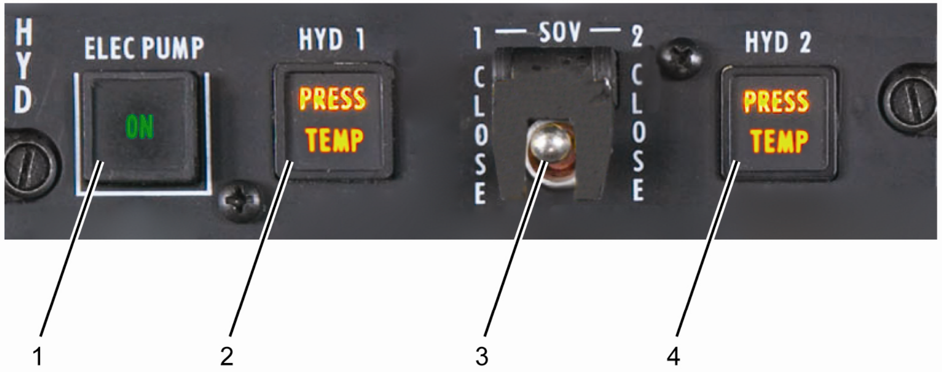
central position ……............	the shut-off valves are open and automatically controlled by the system
CLOSE …………......…….....	ON THE GROUND
	allows to select a single system (the no.1 or the no.2) during the ground check
	IN FLIGHT
	allows to isolate a system (the no.1 or the no.2) in case of anomalous conditions

4. HYD 2 indicator

PRESS (amber) lighted ..….	the hydraulic pressure in the system no.2 drops below 163 bar
TEMP (amber) lighted ….....	the temperature of the fluid in the system no.2 is above 134°C



HYD CONTTROL PANEL



HYDRAULIC POWER – CONTROLS AND INDICATIONS

HYDRAULIC POWER – SYNOPTIC DIAGRAM

On each Multi-Function Display (MFD) a synoptic diagram representing the status of the hydraulic power operation can be displayed via the "System / Hydraulic" menu.

The synoptic diagram shows:

- the pressure values in the systems no.1 and no. 2
- the temperature values in the systems no.1 and no. 2
- the level of the hydraulic fluid in the reservoirs no.1 and no.2 (in %)
- the status of the return by-pass filters in the systems no.1 and no.2
- the status of the hydraulic pumps no.1, no.2 and no.4
- the status of the electrical pump
- the status of landing gear UTIL SOV1 and UTIL SOV2
- the status of the flight controls SOV1, SOV2 and TRSOV

CAUTION: In case of MAU 1(2) failure, do not use synoptic pages information.

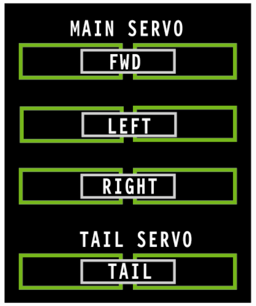
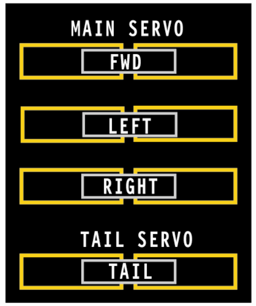
SYMBOLS USED IN THE SYNOPTIC

The following table shows the symbols used to represent the major components of the hydraulic system on the hydraulic synoptic page and the relevant states.

PUMP STATE	PUMP GRAPHIC
OFF	
FAILED OFF	
ON	
FAILED ON	
UNDETERMINED	

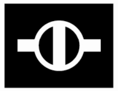
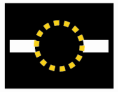
LEVEL STATE OF THE RESERVOIR	RESERVOIR GRAPHIC
100%	
50%	
28%	
22%	
UNDETERMINED	

ITEM	NORM	FAILED
LANDING GEAR ACTUATOR GRAPHIC		

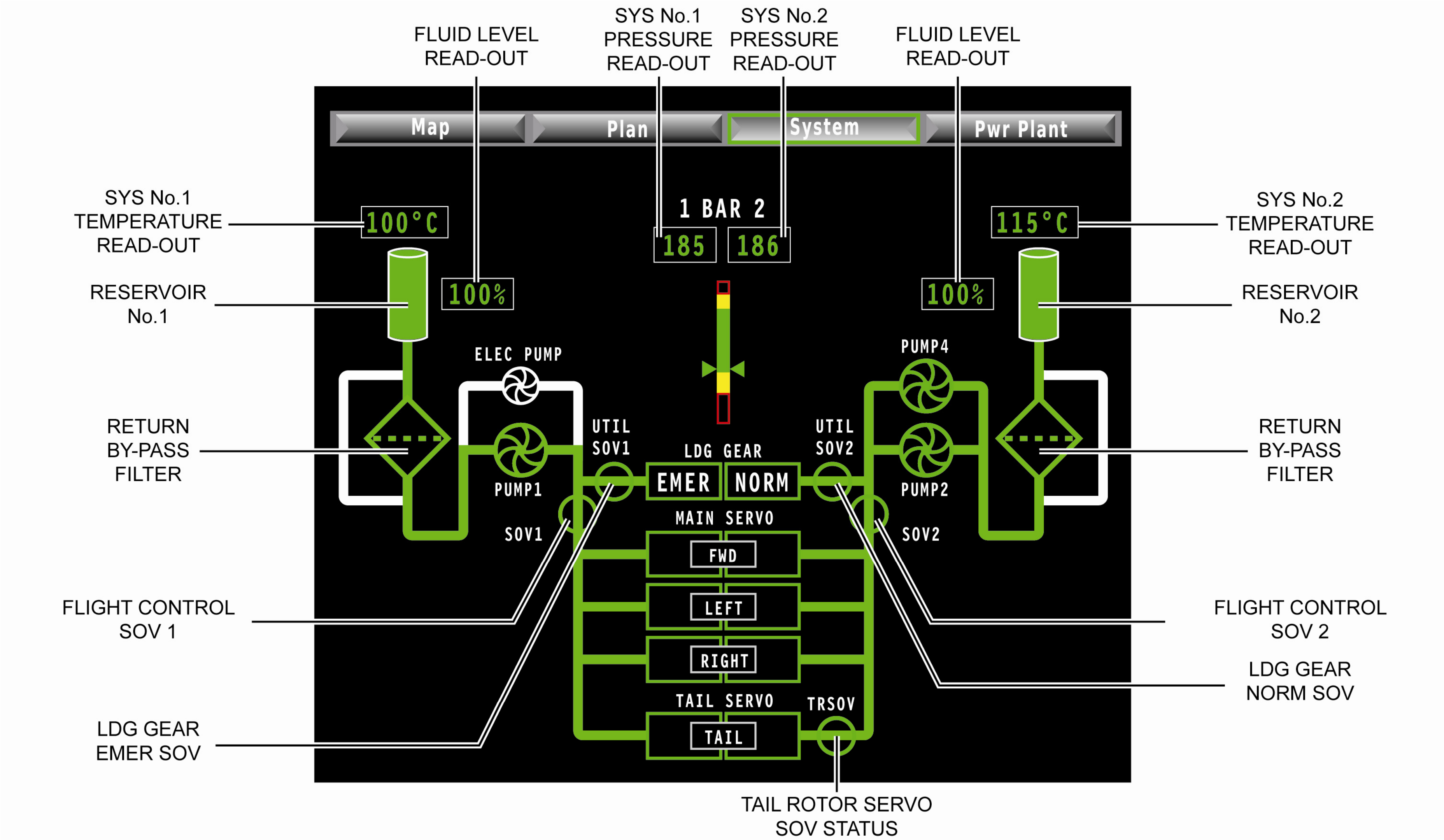
ITEM	NORM	FAILED
FLIGHT CONTROL ACTUATOR GRAPHIC		

FILTER STATE	FILTER GRAPHIC
NORMAL	
CLOGGED	
UNDETER-MINED	

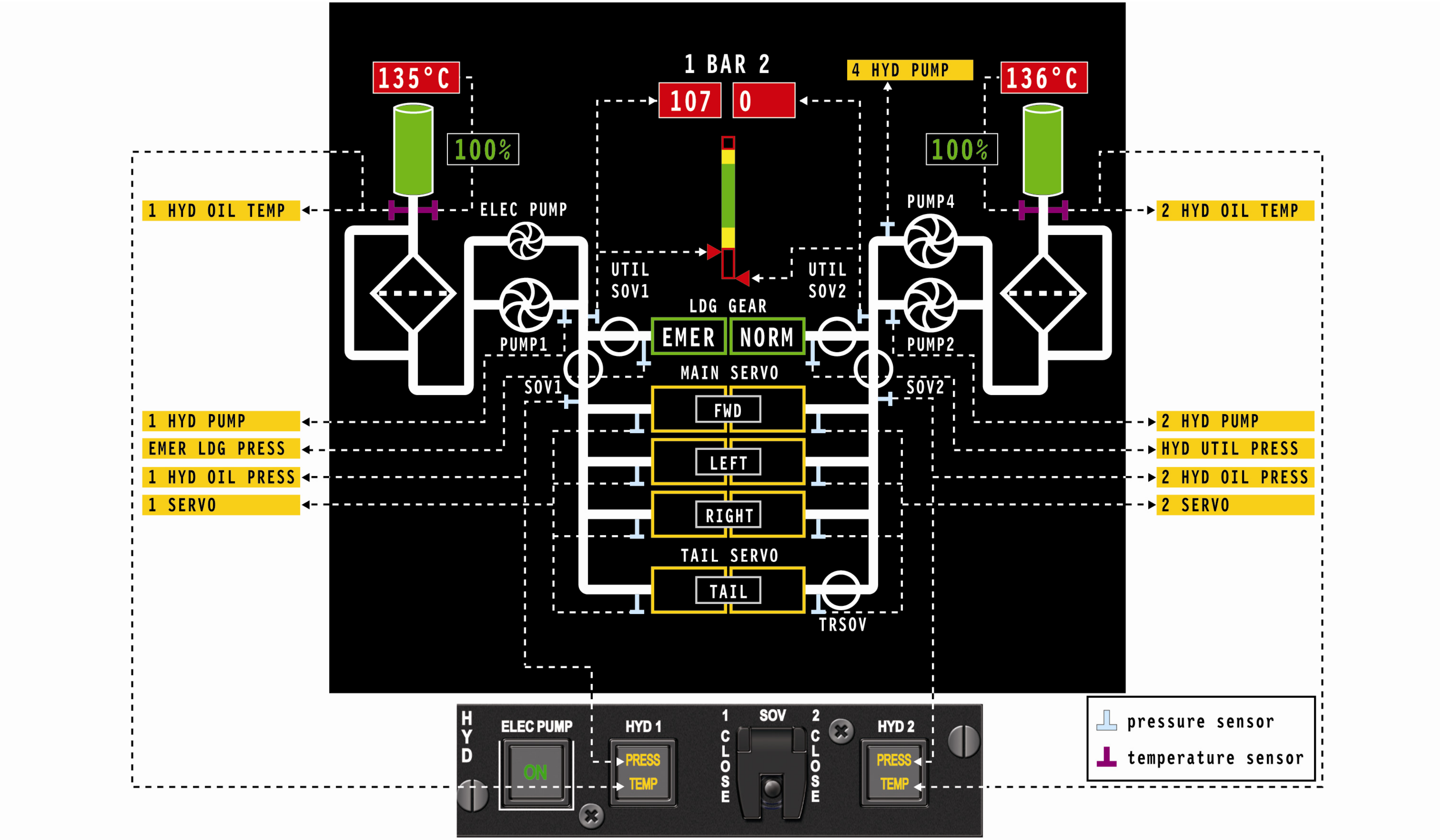
FLOW LINE STATE	FLOW LINE GRAPHIC
NO FLOW	
FLOW	
UNDETERMINED	

VALVE STATE	VALVE GRAPHIC	VALVE STATE	VALVE GRAPHIC
CLOSED		FAIL OPEN	
OPEN		UNDETERMINED	

SYMBOLS USED IN THE SYNOPTIC DIAGRAM



HYDRAULIC POWER – SYNOPTIC DIAGRAM



PRESSURE / TEMPERATURE SENSORS LOCATION AND RELATED CAUTIONS

HYD SYS NO.1 – PRINCIPLE OF OPERATIONS

The hydraulic system no.1

- supplies hydraulic power to the main rotor and tail rotor servo actuators
- supplies power to the landing gear extension and retraction system in normal conditions

The hydraulic system no.1 is controlled by a combination of

- automatic controls
- manual controls

AUTOMATIC CONTROLS

Automatic controls are provided by the fluid level microswitches located inside the PCM 1 reservoir.

- if the fluid in the PCM 1 reaches the minimum level (22%), the UTIL SOV1 of the LDG GEAR EMER circuit will close to stop further drop in fluid pressure
- if the fluid in the PCM 2 reservoir reaches the minimum level (22%), SOV1 will be inhibited from closing to avoid that, if TRSOV closes, the tail rotor will lose power completely

MANUAL CONTROLS

The pilot operates the hydraulic system no. 1 acting on

- the ELEC PUMP push button switch
- the flight control SOV switch

In flight, the SOV switch is used to close the system no.1 if a hydraulic over temperature condition occurs.

HYDRAULIC SYSTEM NO.2 – PRINCIPLE OF OPERATIONS

The hydraulic system no.2

- supplies hydraulic power to the main rotor and tail rotor servo actuators
- supplies power to the landing gear extension and retraction system in normal conditions

The hydraulic system no.2 is controlled by a combination of

- automatic controls
- manual controls

AUTOMATIC CONTROLS

Automatic controls are provided by the fluid level microswitches located inside the PCM 2 reservoir.

- if the fluid in the utility circuit reaches the level of 50%, the UTIL SOV2 will close to stop further drop in fluid pressure
- if the fluid in the PCM 2 reaches the level of 28%, the TRSOV will close and the UTIL SOV2 is re-opened
- if the fluid in the PCM 2 reaches the minimum level (22%), the UTIL SOV2 will be closed again (TRSOV still closed)

MANUAL CONTROLS

The pilot operates on the hydraulic system no. 2 acting on

- the Flight Control SOV switch

In flight the SOV switch is used to close the system no.2 if a hydraulic over temperature condition occurs.

On the ground, the SOV switch is used to check the hydraulic system no.2.

PRE-FLIGHT CHECK OPERATIONS

FLIGHT CONTROLS

The system no.1 permits to test the flight controls on ground only.

The pilot pressing the ELEC PUMP push-button on the HYD control panel can carry out cyclic, collective and yaw pedals full and free check.

The full and free check should be carried out with slow displacement of the controls and one control at a time in order not to overload the electric pump.

The electrical pump disengages automatically after 2 minutes.

HYDRAULIC synoptic page

- ELEC PUMP pressurises the hydraulic system no.1 at 105 bars. This value of pressure implies a fail condition for the main rotor, tail rotor and emergency landing gear servo-actuators
- SOV1 and SOV2 are displayed in an undetermined status
- cautions are displayed in the CAS window

  1-2-4 HYD PUMP

  1-2 SERVO

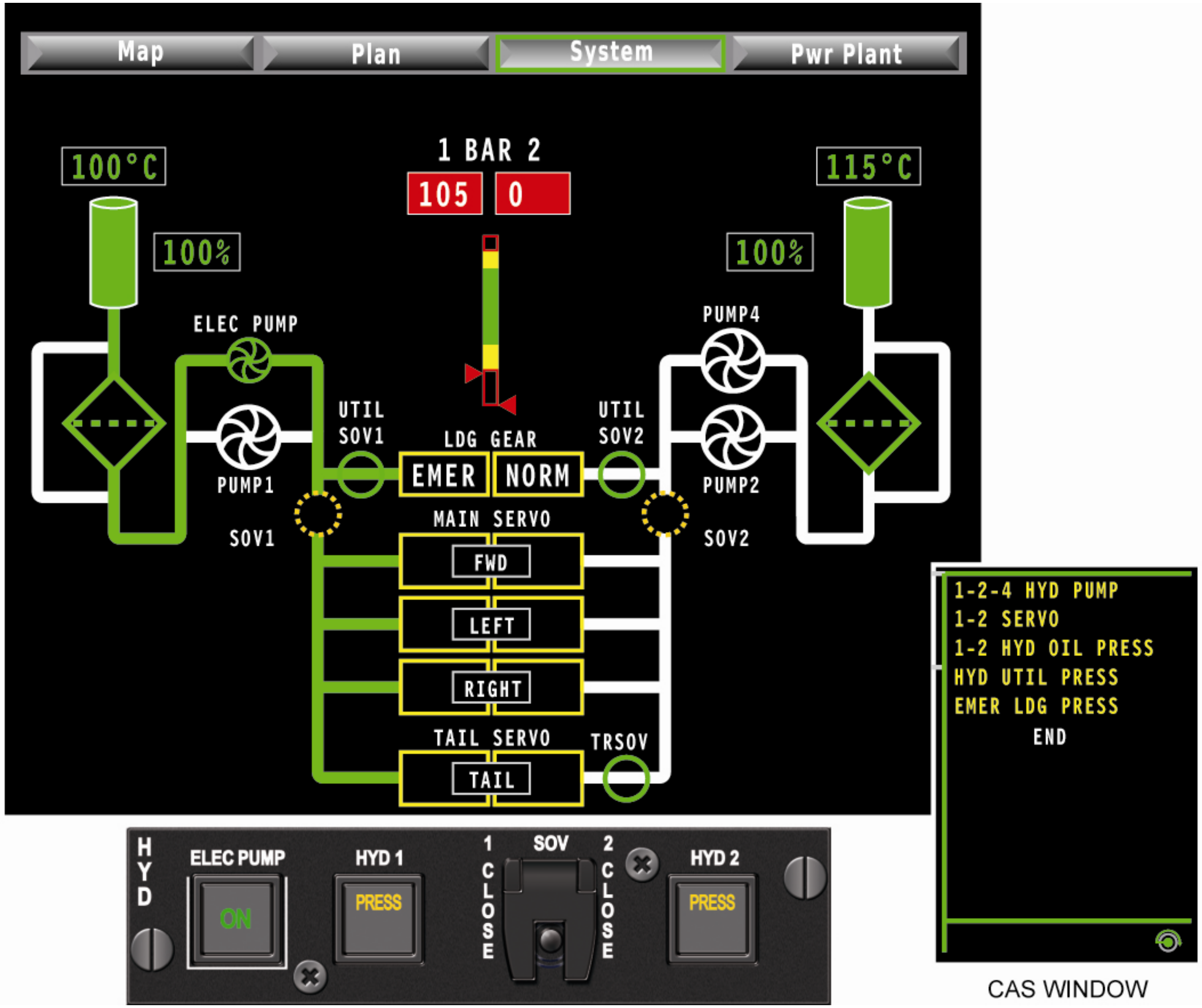
  1-2 HYD OIL PRESS

  HYD UTIL PRESS

  EMER LDG PRESS

on HYD control panel

- ELEC PUMP: ON lighted (green)
- HYD1 and HYD2: PRESS lighted (amber)



CAS WINDOW

PRE–FLIGHT CHECK OPERATIONS

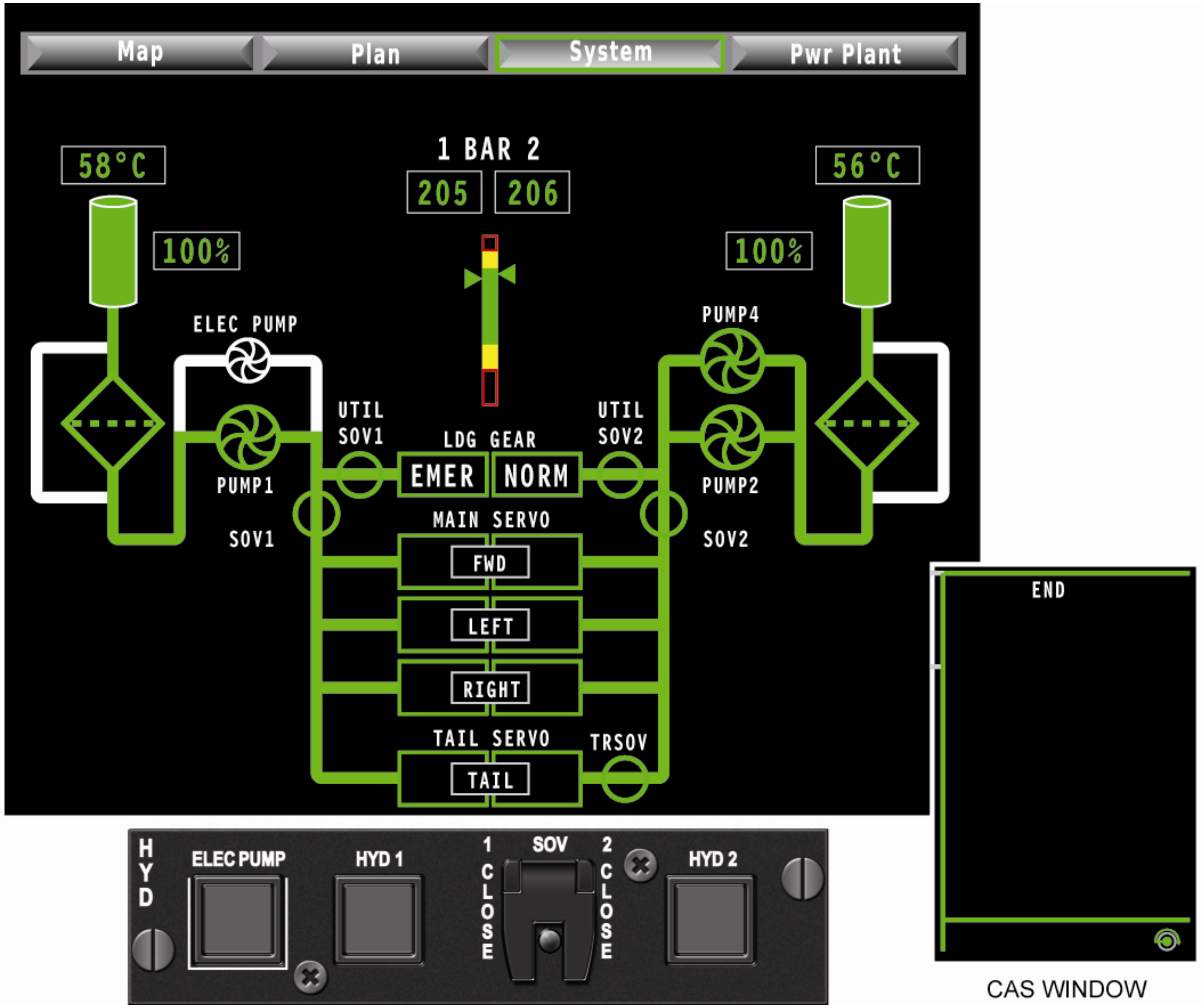
NORMAL OPERATIONS

HYDRAULIC synoptic page

- PUMP1 pressurizes the system no.1
- PUMP2 and PUMP4 pressurize the system no.2
- UTIL SOV1 and UTIL SOV2 are opened and the landing gear is pressurized at 207 bar
- SOV1, SOV2 and TRSOV are opened and flight controls are pressurized at 207 bar by both hydraulic systems

on HYD control panel

- HYD1 and HYD2: blank



CAS WINDOW

NORMAL OPERATIONS

FAILURE - PUMP NO.2

In case of failure of one pump (for example PUMP2), the remaining pump (in this example PUMP4) is able to supply the operating pressure to the system no.2.

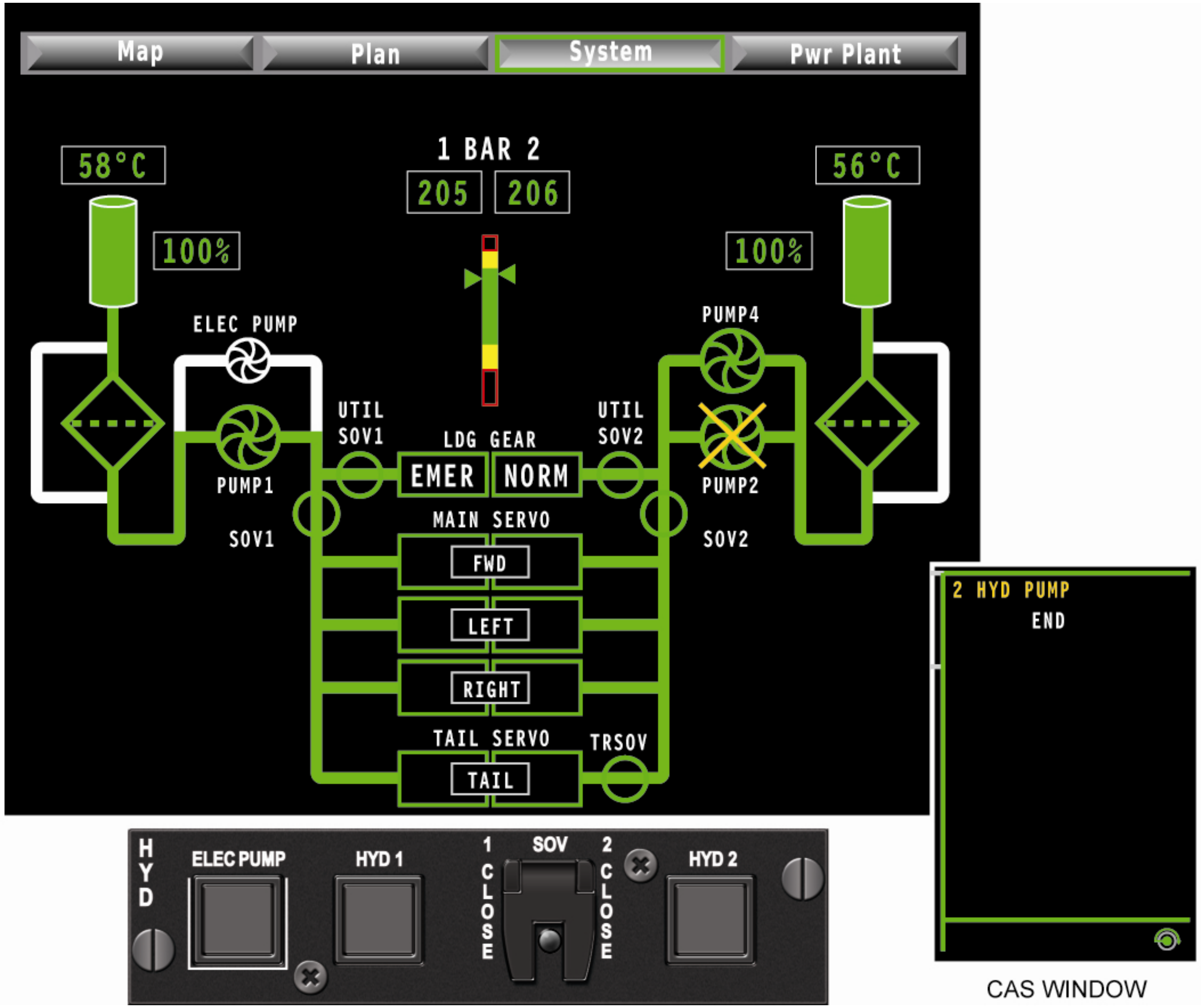
HYDRAULIC synoptic page

- PUMP1 pressurizes the system no.1
- PUMP2 fails; PUMP4 pressurize the system no.2
- UTIL SOV1 and UTIL SOV2 are opened and the landing gear is pressurized at 207 bar
- SOV1, SOV2 and TRSOV are opened and flight controls are pressurized at 207 bar by both hydraulic systems
- a caution is displayed in the CAS window and the pilot has to follows the relevant malfunction procedure

  2 HYD PUMP

on HYD control panel

- HYD1 and HYD2: blank



CAS WINDOW

FAILURE – PUMP NO.2

FAILURE - PUMP NO.1

PUMP1 failure implies the total loss of system no.1 operations. In this case hydraulic power is supplied by the system no.2 only.

HYDRAULIC synoptic page

- PUMP1 fails
- PUMP2 and PUMP4 pressurize the system no.2
- SOV1 and UTIL SOV1 are in an undetermined status. The flight controls are considered failed for the system no.1. The landing gear emergency operation is not available
- UTIL SOV2 is opened and the landing gear normal operations is pressurized at 207 bar
- SOV2 and TRSOV are opened and flight controls are pressurized at 207 bar only by the system no.2
- cautions are displayed in the CAS window and the pilot has to follows the relevant malfunction procedure

  1 HYD PUMP

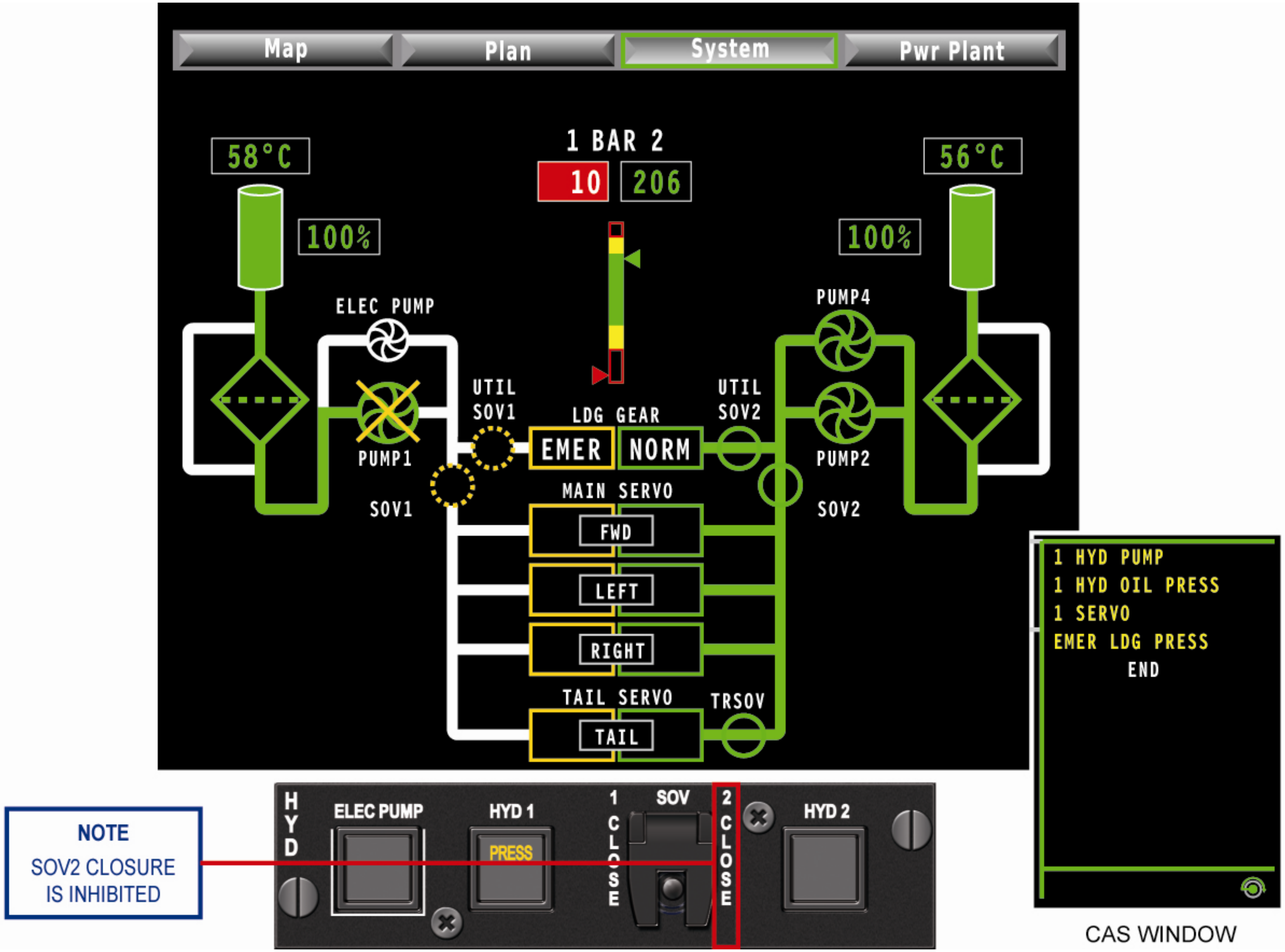
  1 HYD OIL PRESS

  1 SERVO

  EMER LDG PRESS

on HYD control panel

- HYD 1: PRESS lighted (amber)
- HYD 2: blank



FAILURE – PUMP NO.1

FAILURE – HYD 2 FLUID LEVEL AT 50%

Hereafter are described some failure relevant different conditions of leakage in the hydraulic system no.2. In this case, the three level microswitches installed inside the reservoir, control the UTIL SOV and the TRSOV to contain the leakage.

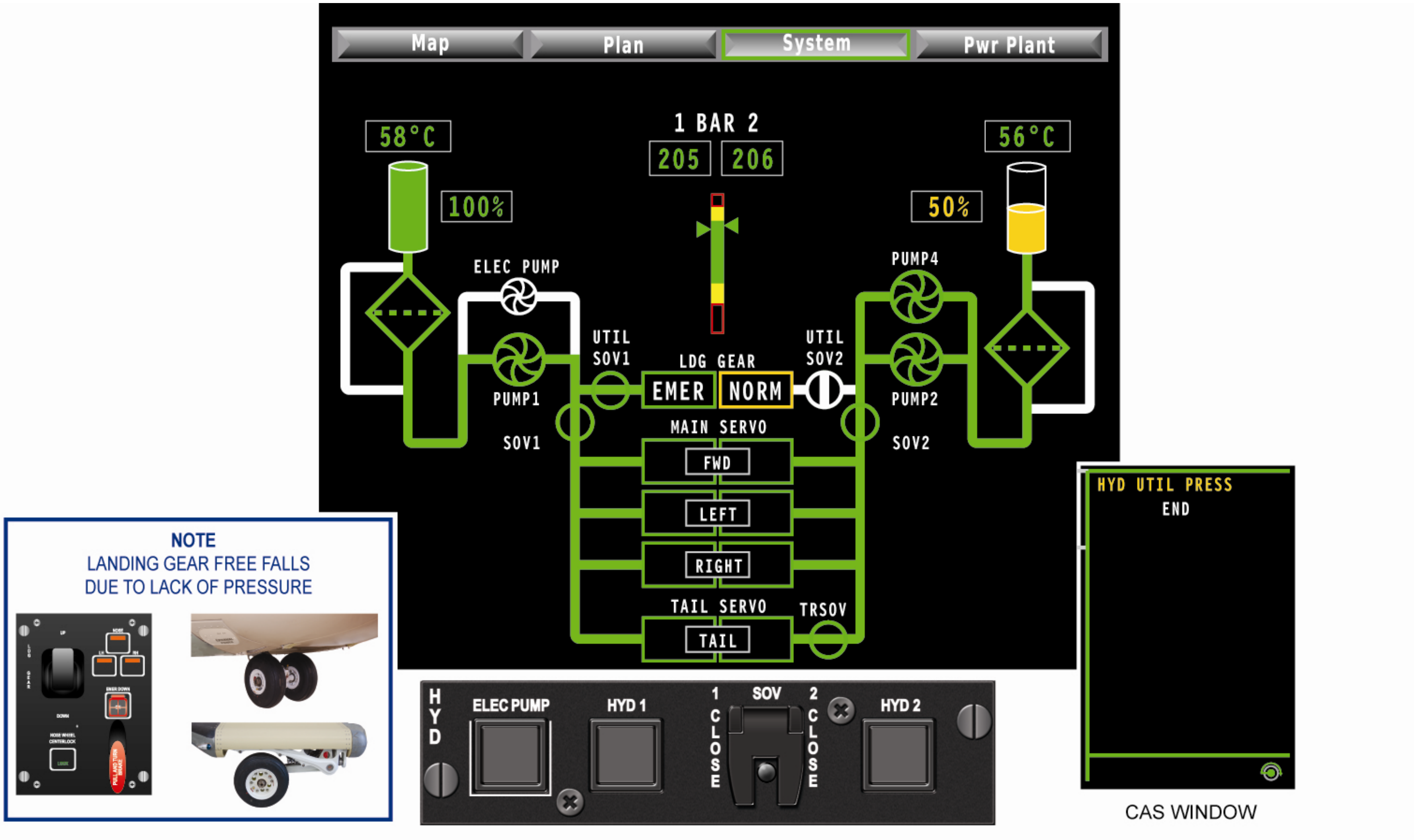
HYDRAULIC synoptic page

- PUMP1 pressurizes the system no.1
- PUMP2 and PUMP4 pressurize the system no.2
- UTIL SOV1 is opened and the landing gear EMER circuit is pressurized at 207 bar
- UTIL SOV2 is automatically closed by the level microswitch and the landing gear normal operation is not available. The landing gear free falls due to lack of pressure
- SOV1, SOV2 and TRSOV are opened and the flight controls are pressurized at 207 bar by the system no.1 and no.2
- a caution is displayed in the CAS window and the pilot has to follows the relevant malfunction procedure

  HYD UTIL PRESS

on HYD control panel

- HYD1 and HYD 2: blank



FAILURE – HYD 2 FLUID LEVEL AT 50%

FAILURE – HYD 2 FLUID LEVEL AT 28%

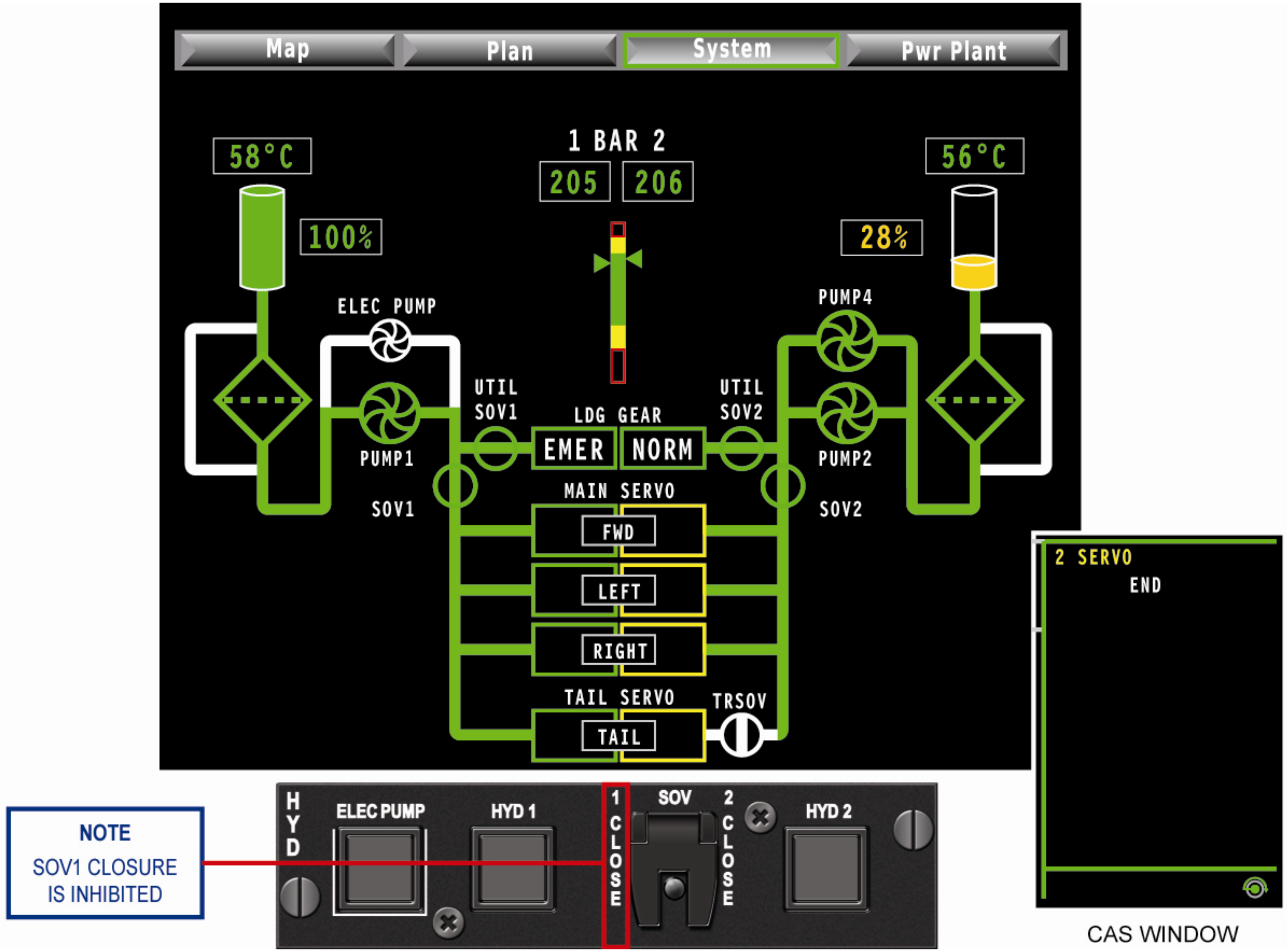
HYDRAULIC synoptic page

- PUMP1 pressurizes the system no.1
- PUMP2 and PUMP4 pressurize the system no.2
- UTIL SOV1 and UTIL SOV2 are opened. UTIL SOV2 is automatically re-opened by the 28% level microswitch
- SOV1 and SOV2 are opened and flight controls are pressurized at 207 bars by the system no.1
- TRSOV is automatically closed by the level microswitch
- a caution is displayed in the CAS window and the PLT has to follows the relevant malfunction procedure

  2 SERVO

on HYD control panel

- HYD1 and HYD 2: blank



CAS WINDOW

FAILURE – HYD 2 FLUID LEVEL AT 28%

FAILURE – HYD 2 FLUID LEVEL AT 22% (MIN)

HYDRAULIC synoptic page

- PUMP1 pressurizes the system no.1
- PUMP2 and PUMP4 pressurize the system no.2
- UTIL SOV 1 is opened and the landing gear emergency circuit is pressurized at 207 bar
- UTIL SOV2 and TRSOV are automatically closed by the minimum level microswitch (22%). The landing gear normal operation is not available and the landing gear free falls due to a lack of pressure
- cautions are displayed in the CAS window and the pilot has to follows the relevant malfunction procedure

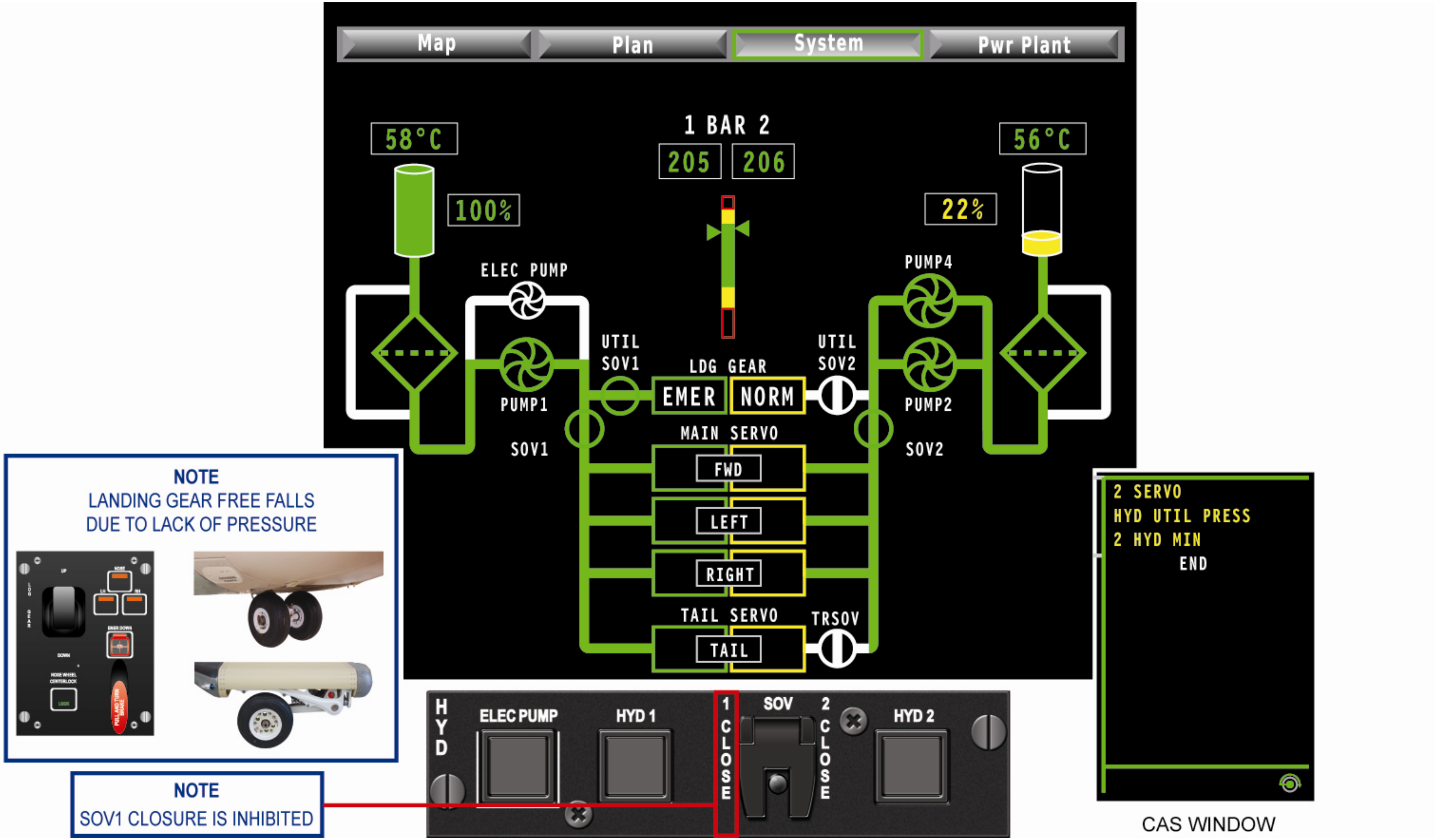
  2 SERVO

  HYD UTIL PRESS

  2 HYD MIN

on HYD control panel

- HYD1 & HYD 2: blank



FAILURE – HYD 2 FLUID LEVEL AT 22% (MIN)

FAILURE – HYD 1 FLUID LEVEL AT 50%

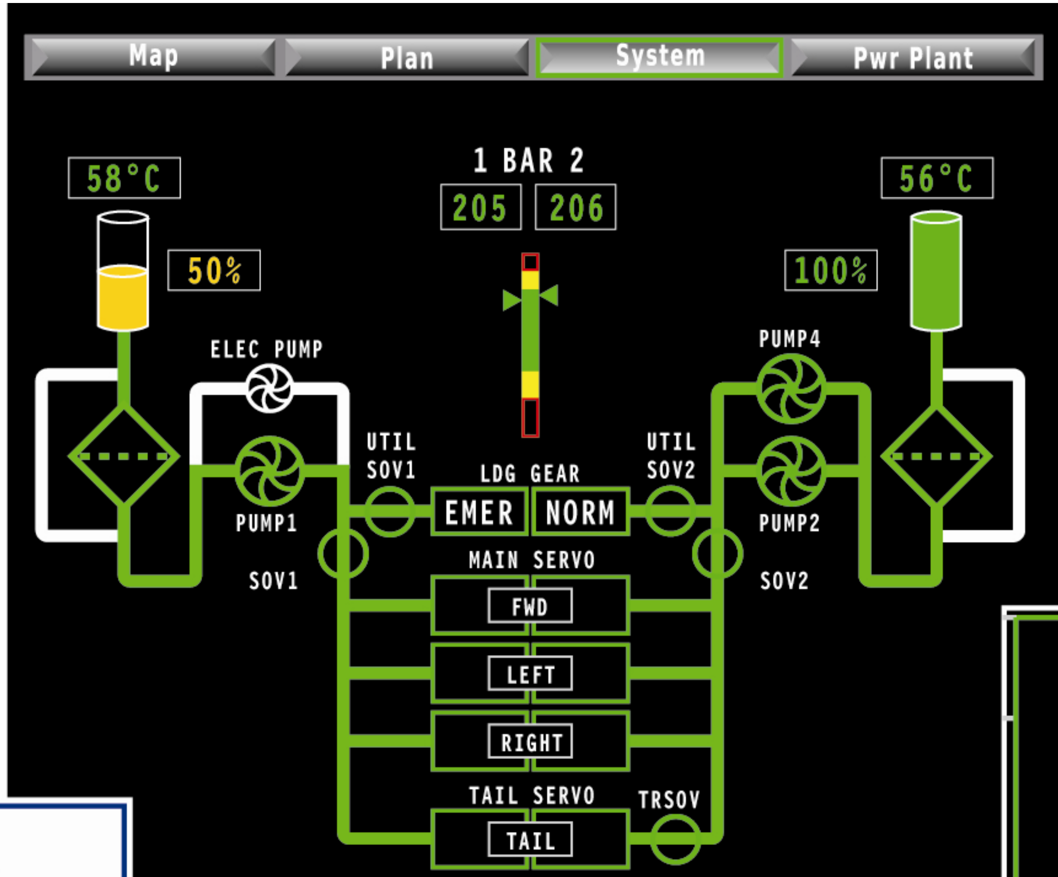
When the level of the fluid inside the PCM1 reservoir reaches the 50%, there will be no effects on the system operations. Only the indications in the synoptic page (reservoir and read-out amber) are provided.

<u>HYDRAULIC synoptic page</u>

- PUMP1 pressurizes the system no.1
- PUMP2 and PUMP4 pressurize the system no.2
- UTIL SOV1 and UTIL SOV2 are opened and the landing gear circuit is pressurized at 207 bar
- SOV1, SOV2 and TRSOV are opened and the flight control circuit is pressurized at 207 bar

<u>on HYD control panel</u>

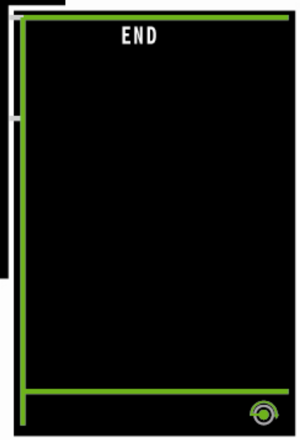
- HYD1 and HYD 2: blank



NOTE

IF No.1 HYD RESERVOIR LEVEL= 50%:

- ONLY THE INDICATION IN THE SYNOPTIC PAGE IS PROVIDED
- NO EFFECT IN THE SYSTEM OPERATION



CAS WINDOW

FAILURE – HYD 1 FLUID LEVEL AT 50%

FAILURE – HYD 1 FLUID LEVEL AT 28%

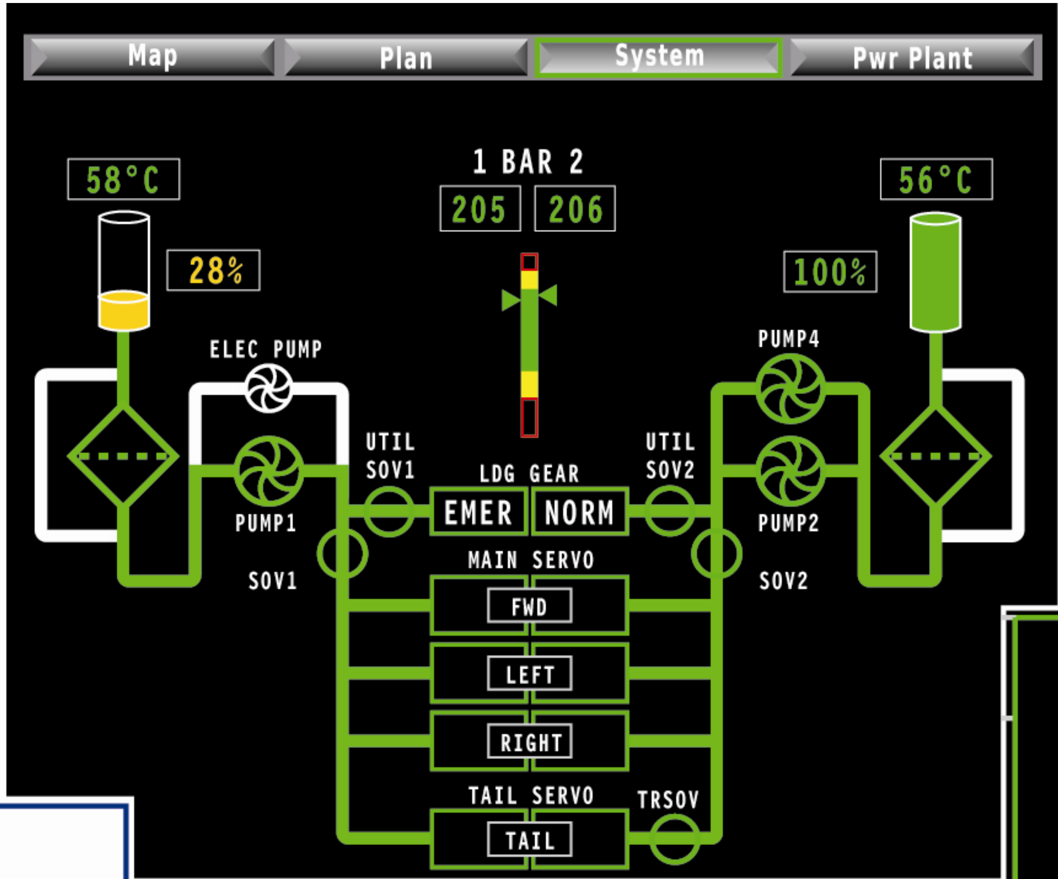
When the level of the fluid inside the PCM1 reservoir reaches the 28%, there will be no effects on the system operations. Only the indications in the synoptic page (reservoir and read-out amber) are provided.

HYDRAULIC synoptic page

- PUMP1 pressurizes the system no.1
- PUMP2 and PUMP4 pressurize the system no.2
- UTIL SOV1 and UTIL SOV2 are opened and the landing gear circuit is pressurized at 207 bar
- SOV1, SOV2 and TRSOV are opened and the flight control circuit is pressurized at 207 bar

on HYD control panel

- HYD1 and HYD 2: blank



CAS WINDOW

NOTE

IF No.1 HYD RESERVOIR LEVEL= 28%:

- ONLY THE INDICATION IN THE SYNOPTIC PAGE IS PROVIDED
- NO EFFECT IN THE SYSTEM OPERATION

FAILURE – HYD 1 FLUID LEVEL AT 28%

FAILURE – HYD 1 FLUID LEVEL AT 22% (MIN)

When the level of the fluid inside the PCM1 reservoir reaches the 22%, there will be no effects on the system operations.

HYDRAULIC synoptic page

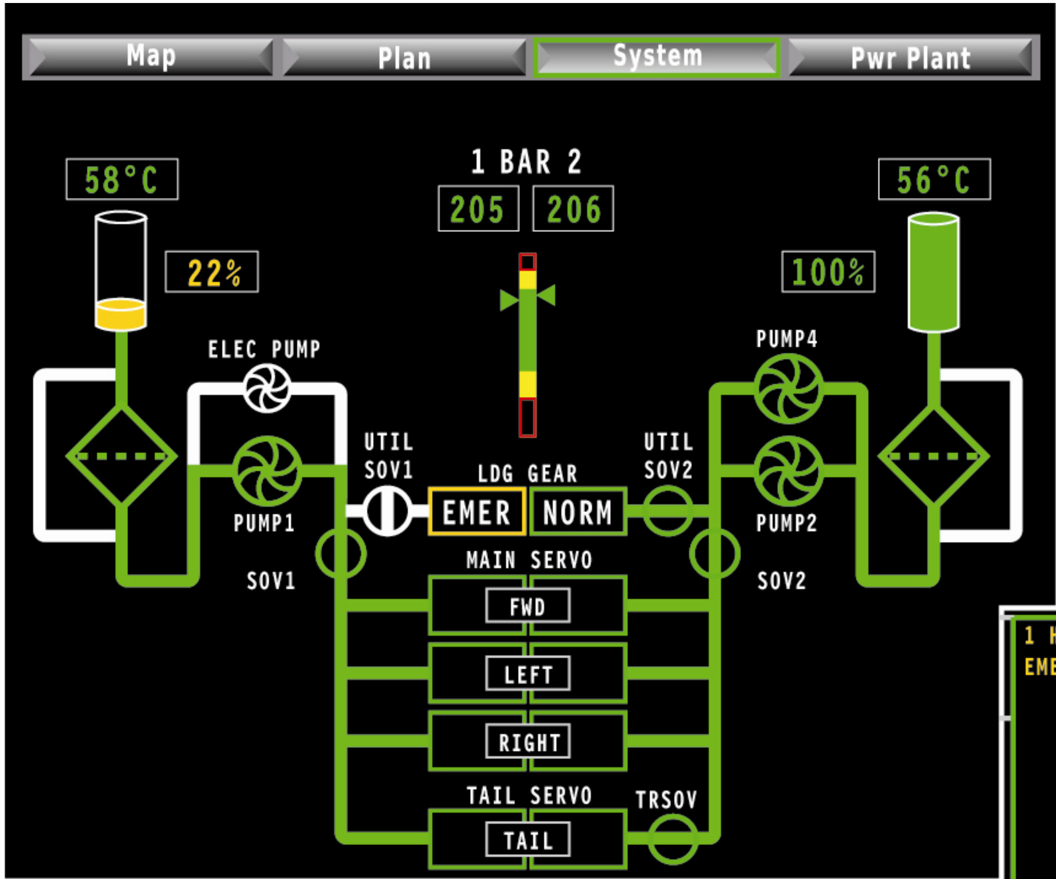
- PUMP1 pressurizes the system no.1
- PUMP2 and PUMP4 pressurize the system no.2
- UTIL SOV1 automatically closed by the minimum level microswitch (22%) and the landing gear emergency operation is not available
- UTIL SOV2 and TRSOV are opened.
- SOV1, SOV2 and TRSOV are opened and the flight control circuit is pressurized at 207 bar
- cautions are displayed in the CAS window and the pilot has to follows the relevant malfunction procedure

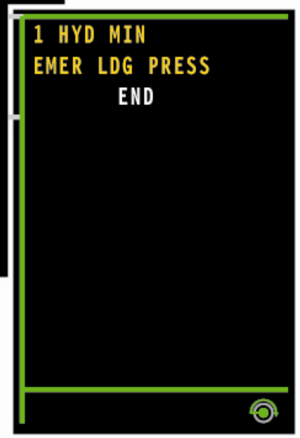
  1 HYD MIN

  EMER LDG PRESS

on HYD control panel

- HYD1 and HYD 2: blank





CAS WINDOW

NOTE

SYSTEM No.1 ACTIVATES THE CLOSURE OF UTIL SOV1 ONLY AT 22% OF No.1 RESERVOIR LEVEL

FAILURE – HYD 1 FLUID LEVEL AT 22% (MIN)

PAGE INTENTIONALLY LEFT BLANK

HYD SYS No.1	HYD FLUID LEVEL		HYD SYS No.2
NORMAL OPERATION		100%	NORMAL OPERATION
NORMAL OPERATION		50%	• UTIL SOV 2 = CLOSED* ⇨ LDG GEAR FREE FALL ↓ HYD UTIL PRESS
NORMAL OPERATION		28%	• UTIL SOV 2 = OPEN* ⇨ LDG CIRCUIT REPRESSURIZED • TRSOV = CLOSED* ↓ 2 SERVO
• UTIL SOV 1 = CLOSED* ↓ 1 HYD MIN EMER LDG PRESS		22%	2 HYD MIN ↓ • UTIL SOV 2 = CLOSED* ⇨ LDG GEAR FREE FALL ↓ HYD UTIL PRESS • TRSOV = CLOSED ↓ 2 SERVO

* Automatic action

FLUID LEVEL LOGIC – SUMMARY

FAILURE – HYDRAULIC FLUID OVERHEATING

When the hydraulic fluid reaches an overheating condition (the temperature is greater than 134°C), the affected hydraulic system (no.2 in this example) must be isolated.

HYDRAULIC synoptic page

- check to confirm the system no.2 over temperature
- lower the LDG GEAR following the normal procedure
- lift the cover of the SOV switch on the HYD control panel and move the switch to 2 CLOSE
- cautions are displayed in the CAS window and the pilot has to follows the relevant malfunction procedure

  2 HYD OIL TEMP

  2 HYD OIL PRESS

  2 SERVO

on HYD control panel

- HYD 1: blank
- HYD 2: TEMP and PRESS lighted (amber)

NOTE 1.

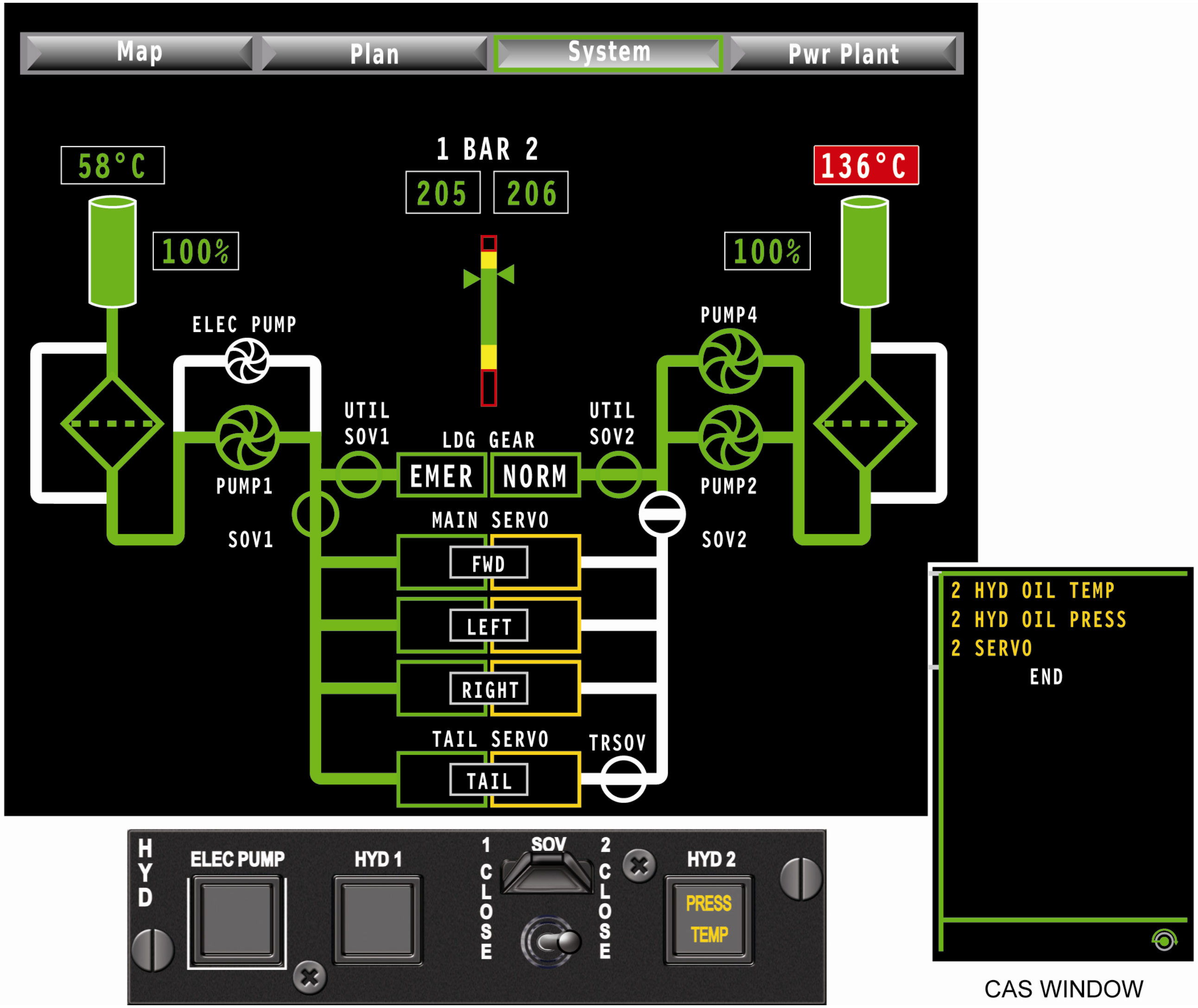
Overheating in hydraulic system no.1 is similar.

NOTE 2.

With one hydraulic system SOV shut off, a subsequent drop of pressure in the other system will override the SOV selection and reinstate pressure to the servo's. In these conditions the SOV switch will not be automatically reset.

WARNING

A subsequent 1(2) SERVO caution in the opposite system (if due to servoactuator jamming, see chapter 67-00) will not override the SOV selection resulting in a total loss of the affected servoactuator.



CAS WINDOW

FAILURE – HYDRAULIC FLUID OVERHEATING

FAILURE – OPPOSITE SERVO AND HYDRAULIC FLUID OVERHEATING

When a SERVO and a HYD OIL TEMP caution messages for opposite systems are both illuminated (2 SERVO and 1 HYD OIL TEMP in this example), the SOV switch on the HYD control panel is not inhibited.

Moving to 1 CLOSE will cause loss of control in the no.1 servoactuator.

NOTE.

The SOV switch is ineffective when 1(2) HYD OIL PRESS caution is active.

WARNING

Do not switch SOV to CLOSE on the unaffected system since this will cause loss of control in the affected servo jack.

HYDRAULIC synoptic page

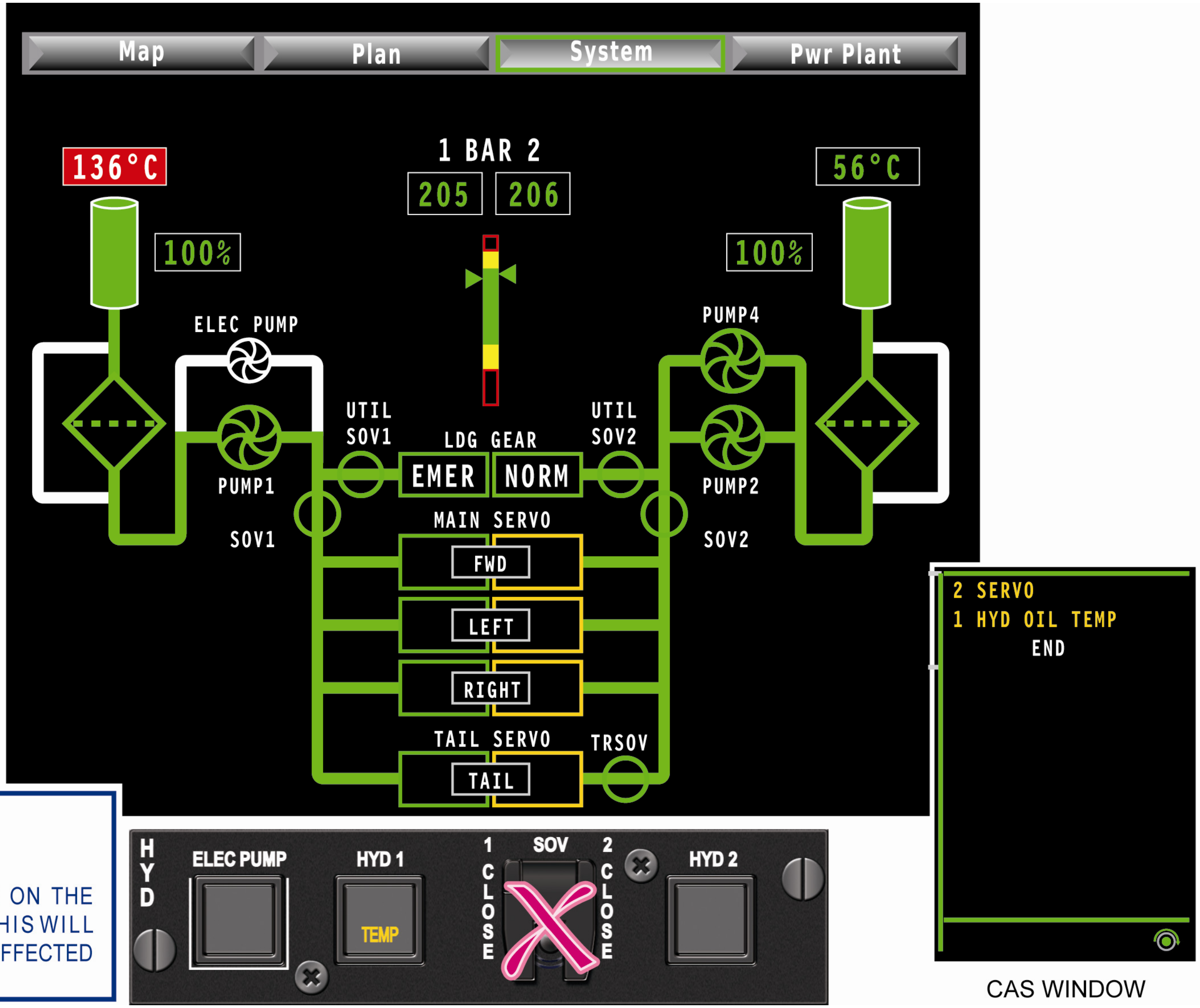
- check to confirm the system no.1 over temperature
- cautions are displayed in the CAS windows and the pilot has to follow the relevant malfunction procedure

  2 SERVO

  1 HYD OIL TEMP

on HYD control panel

- HYD 1: TEMP lighted (amber)
- HYD 2: blank



CAS WINDOW

WARNING

DO NOT SWITCH SOV TO CLOSE ON THE UNAFFECTED SYSTEM SINCE THIS WILL CAUSE LOSS OF CONTROL IN THE AFFECTED SERVO JACK

FAILURE – OPPOSITE SERVO AND HYDRAULIC FLUID OVERHEATING

CAS CAUTION MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
1 HYD OIL PRESS **1 HYD PUMP** **1 SERVO** **EMER LDG PRESS**	Loss of pressure in associated hydraulic system (less than 163 bar)	HYDRAULIC PRESSURE LOW	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES HYDRAULIC SYSTEM
2 HYD OIL PRESS **2–4 HYD PUMP** **2 SERVO** **HYD UTIL PRESS**	Loss of pressure in associated hydraulic system (less than 163 bar)	HYDRAULIC PRESSURE LOW	
HYD UTIL PRESS	Low pressure in landing gear NORM hydraulic system	NORMAL LANDING GEAR PRESSURE LOW	
EMER LDG PRESS	Low pressure in emergency landing gear hydraulic system	EMERGENCY LANDING GEAR PRESSURE LOW	
1(2) HYD OIL TEMP	Associated hydraulic system overheat (greater than 134°C)	HYDRAULIC FLUID OVERHEATING	

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
1(2) HYD MIN	Associated system low hydraulic fluid level	HYDRAULIC FLUID LEVEL LOW	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES HYDRAULIC SYSTEM
1(2)(4) HYD PUMP	Low pressure at pump outlet	HYDRAULIC PUMP 1, 2 OR 4 FAILURE	

LIMITATIONS

Refer to AW139-RFM-4D Section 1.

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CHAPTER 30

ICE AND RAIN PROTECTION

SECTION 00 – GENERAL

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ICE AND RAIN PROTECTION – GENERAL

The ice and rain protection system comprises:

- Pitot Heating system
- Windshield Wiper system

PITOT HEATING SYSTEM – GENERAL

The Pitot Heating system prevents the accretion of ice on the Pitot-static probes.

The pitot-static system collects air data for cockpit instrumentation and consists of two similar systems, one for the pilot (RH) and one for the copilot (LH).

The pitot-static system is part of the Air Data System (ADS) and is described in Ch. 34-00-00.

The two pitot static probes are protected against ice formation by means of electrical heating elements.

Two independent pitot heating systems (pilot and copilot) are installed, each consisting of:

- One Pitot-Static Probe Heating Element
- One Current Monitor
- One control switch

PITOT HEATING SYSTEM – MAIN COMPONENTS

PITOT-STATIC PROBE HEATERS

Each Pitot-Static Probe is provided with an integral electrical heating element (resistor).

LH and RH Pitot-Static Probe heater resistors are supplied by MAIN 1 bus via the PITOT HTR CPLT circuit breaker and by ESS 2 bus via the PITOT HTR PLT circuit breaker, respectively.

CURRENT MONITORS

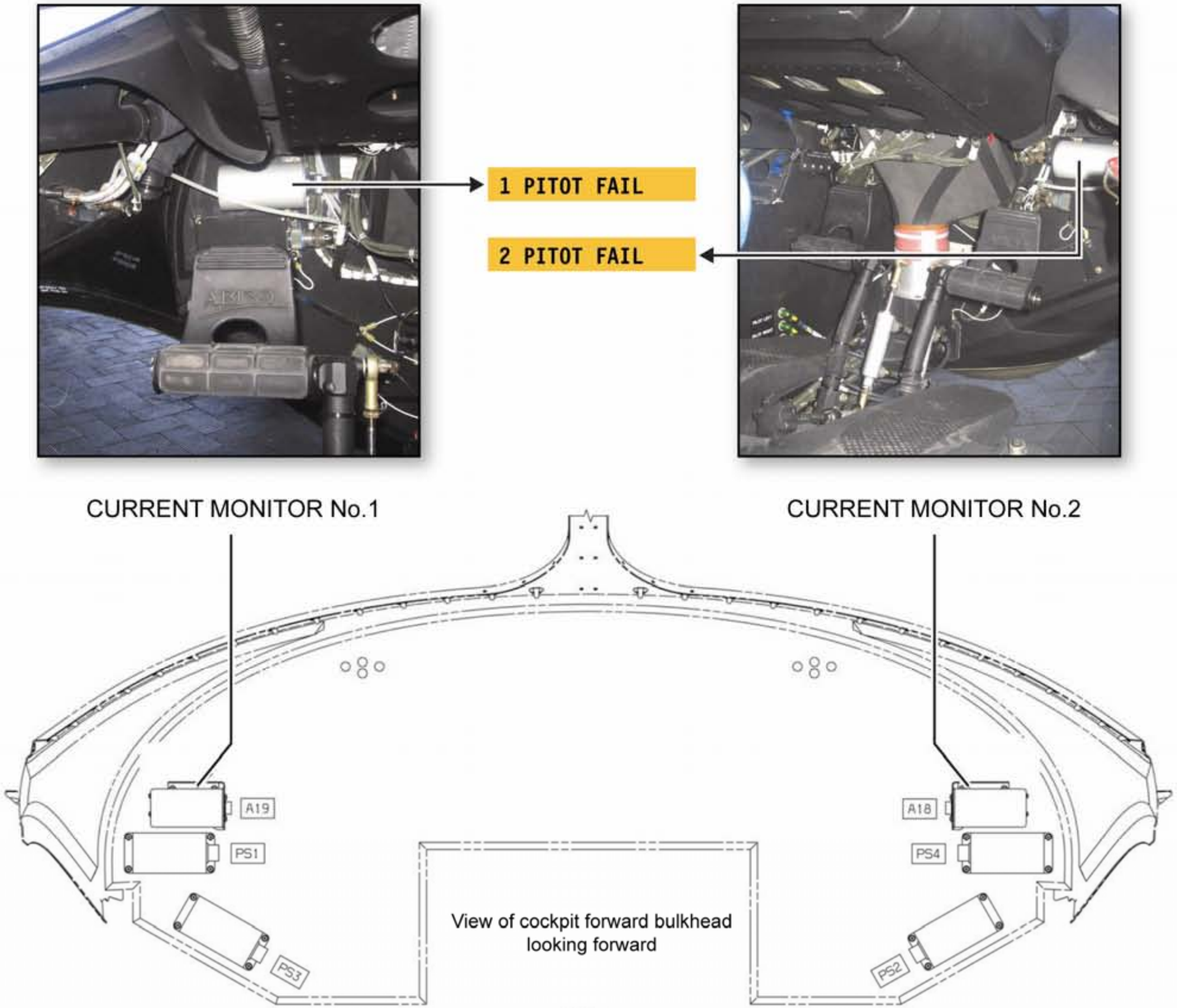
Each Current Monitor monitors the current flowing to the relevant Pitot-Static Probe heating element.

If the PITOT HEATER switch is at ON but the current decreases below a set value, the Current Monitor triggers the 1(2) PITOT FAIL caution message in the CAS window.

The two current monitors are installed forward of copilot (No.1) and pilot (No.2) pedals, below the instrument panel, and are supplied by MAIN 1 bus via the PITOT FAIL CPLT circuit breaker and by ESS 2 bus via the PITOT FAIL PLT circuit breaker, respectively.

NOTE. RH side components are
arranged symmetrically

PITOT SYSTEM – LOCATION OF MAIN COMPONENTS



PITOT SYSTEM – LOCATION OF MAIN COMPONENTS

PITOT SYSTEM - CONTROLS AND INDICATORS

1. PITOT HEATER PILOT switch

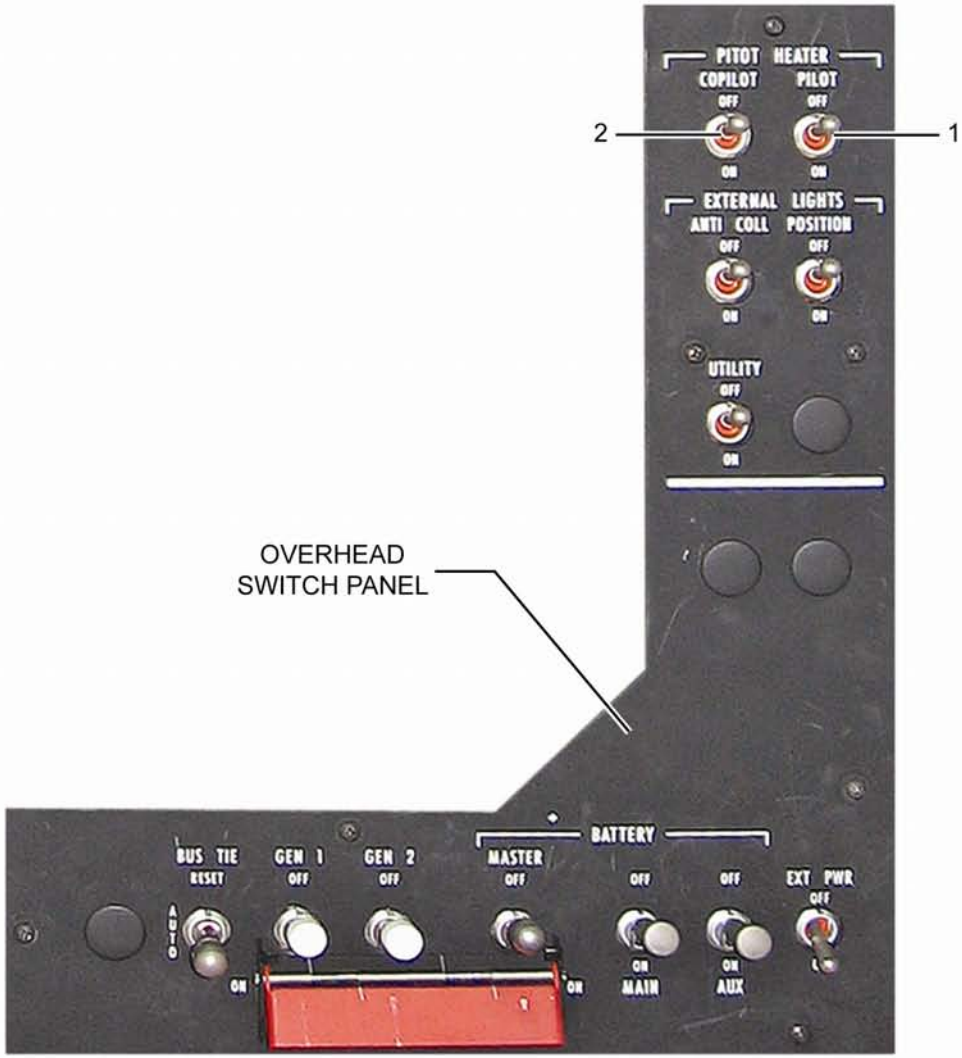
   OFF ………. the right pitot probe (pilot) is not heated

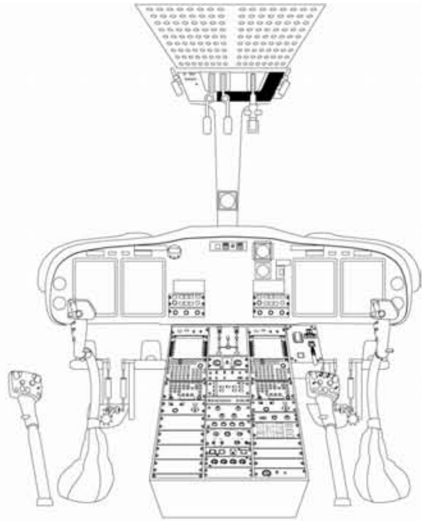
   ON ………… the right pitot probe (pilot) is heated

2. PITOT HEATER COPILOT switch

   OFF ………. the left pitot probe (copilot) is not heated

   ON ………… the left pitot probe (copilot) is heated





PITOT - CONTROLS AND INDICATORS

PITOT HEATING SYSTEM – PRINCIPLE OF OPERATION

When the PITOT HEATER COPILOT switch on the overhead console is at ON, electrical power is connected to the LH pitot-static probe resistor and the PITOT 1 HEAT ON advisory message is displayed in the CAS window.

When the PITOT HEATER PILOT switch on the overhead console is at ON, electrical power is connected to the RH pitot-static probe resistor and the PITOT 2 HEAT ON advisory message is displayed in the CAS window.

A Current Monitor on each line monitors for proper operation of the heating system and triggers the 1(2) PITOT FAIL caution message in the CAS window in case of failure.

PITOT SYSTEM - CAS CAUTION MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
1(2) PITOT FAIL	Associated pitot heater failure	PITOT HEATER FAILURE	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES
1(2) PITOT HEAT OFF	Associated pitot heater is selected OFF and OAT below 4°C	PITOT HEATER OFF	MISCELLANEOUS SYSTEMS

PITOT SYSTEM – CAS ADVISORY MESSAGES

CAS CAPTION	MESSAGE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
1(2) PITOT HEAT ON	Pitot heating ON		Section 2 NORMAL PROCEDURES

PITOT SYSTEM – LIMITATIONS

Refer to AW139-RFM-4D Section 1.

WINDSHIELDS WIPER SYSTEM – GENERAL

The purpose of the windshield wiper system is to keep the windshield surface clean from water, dirt, sand, dust or a thin coat of soft snow.

The system consists of two identical installations, one for the pilot windshield and the other for the copilot windshield and allows the operation of the pilot and copilot wiper blades separately or together.

WINDSHIELDS WIPER SYSTEM – MAIN COMPONENTS

The main components of the windshield wiper system are:

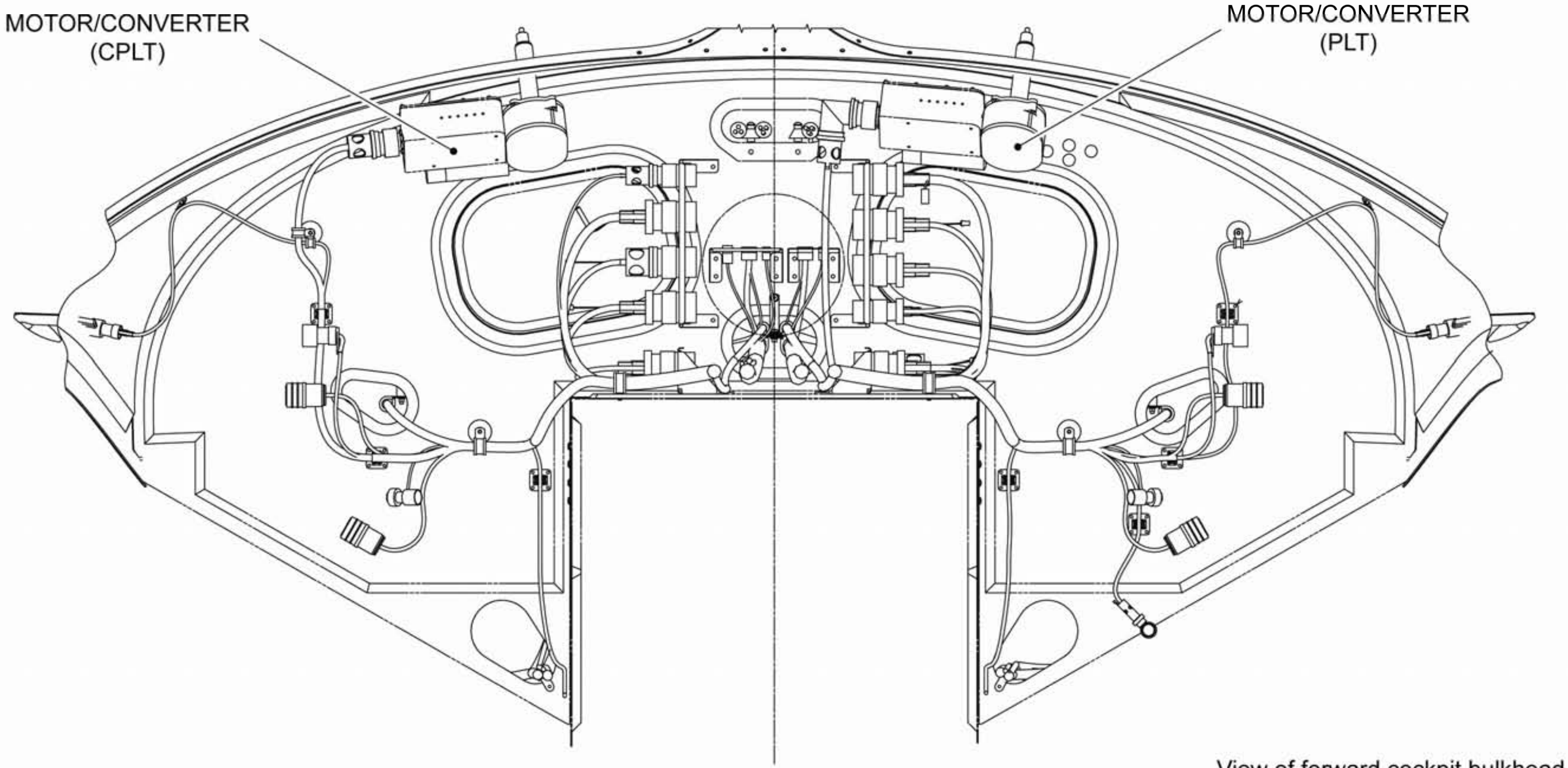
- the arms and wiper blades
- the wiper motor converter

WINDSHIELDS WASHER SYSTEM – GENERAL

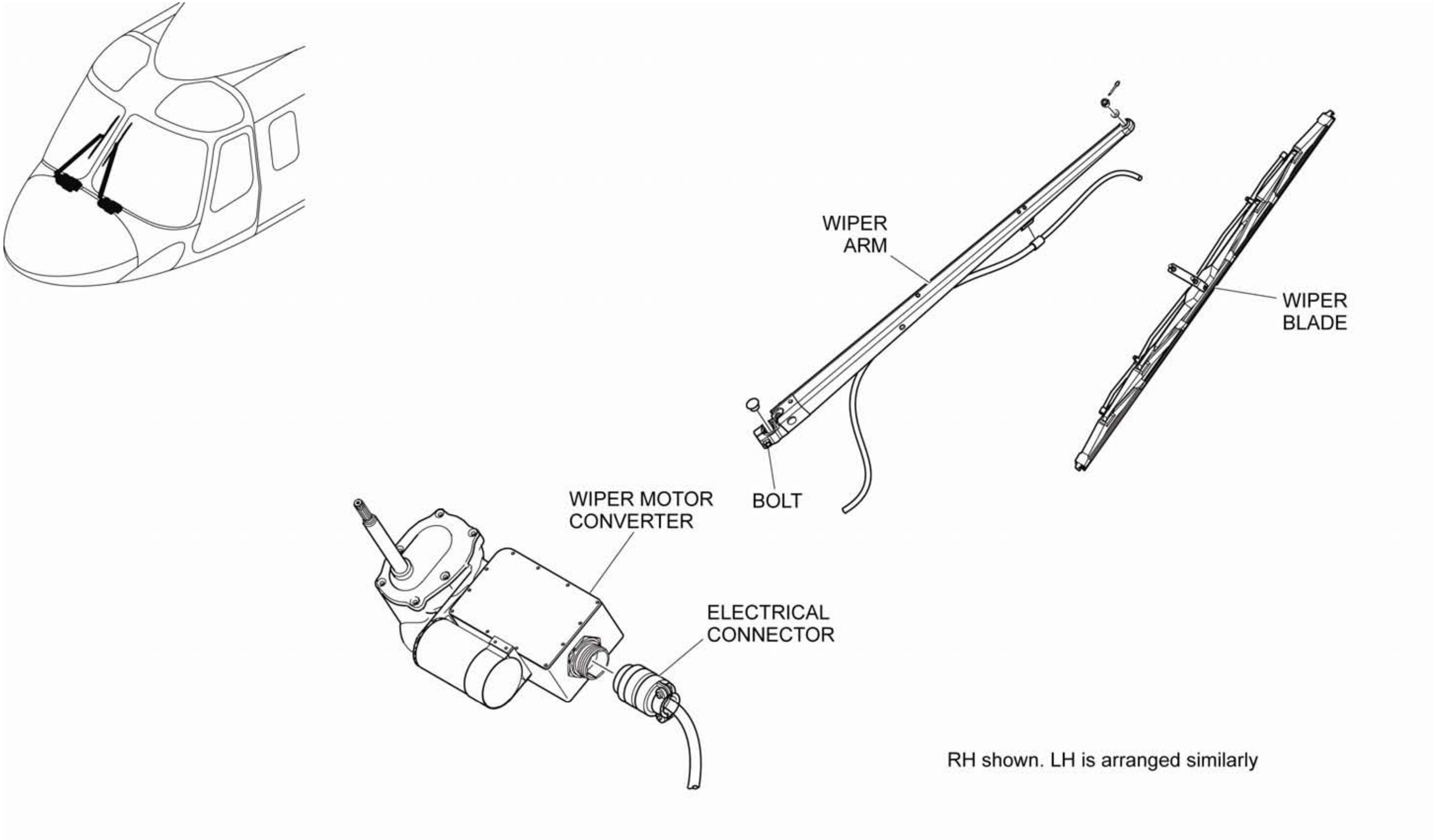
An optional windshield washer system can be installed to help cleaning the windshield and operates in conjunction with the windshield wiper system.

WINDSHIELDS WASHER SYSTEM – MAIN COMPONENTS

The system consists of a tank for the washing liquid, an electrical pump and sprayers on the wiper blades.



WINDSHIELDS WIPER SYSTEM – LOCATION OF MAIN COMPONENTS



WINDSHIELDS WIPER SYSTEM – LOCATION OF MAIN COMPONENTS

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WINDSHIELDS WIPER SYSTEM - CONTROLS AND INDICATORS

1. WINDSHIELD WASHER push-button (optional)

pressed < 1 sec …. the light washing cycle is actuated: a complete low speed cycle with washing and two complete low speed cycles to clean up the windscreen faces

pressed > 1 sec …. the heavy washing cycle is activated: a continuous low speed cycle with washing up to command disengage and two complete low speed cycles to clean up the windscreen faces

NOTE.
The washing mode has precedence against all the other operational modes, unless for the OFF position. It will override the SLOW/FAST rotary switch, the WIPER momentary switches and the SINGLE/DUAL rotary switch.
The washing cycle will engage always either the pilot and copilot systems according to the light/heavy washing cycle.

2. SINGLE/DUAL rotary switch

SINGLE ………….. when selected and the pilot or copilot WIPER momentary switch pressed, the corresponding system starts working

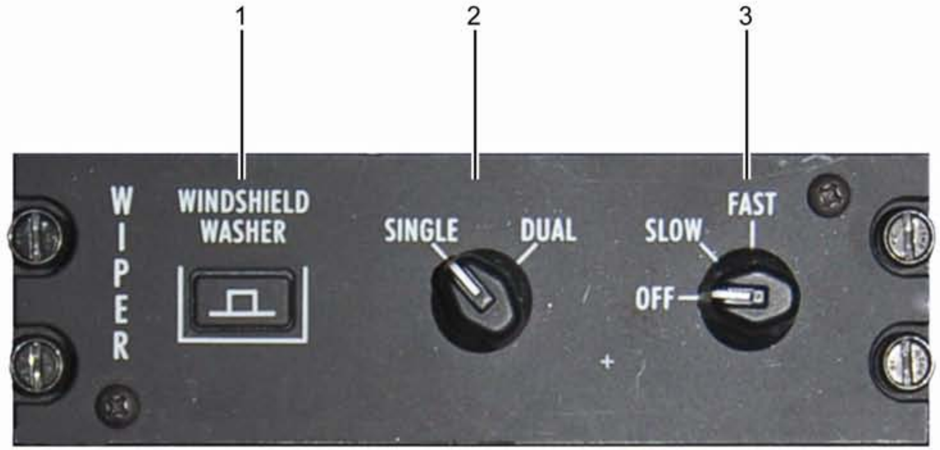
DUAL …………..…. when selected and the pilot or copilot WIPER momentary switch pressed, the systems starts working

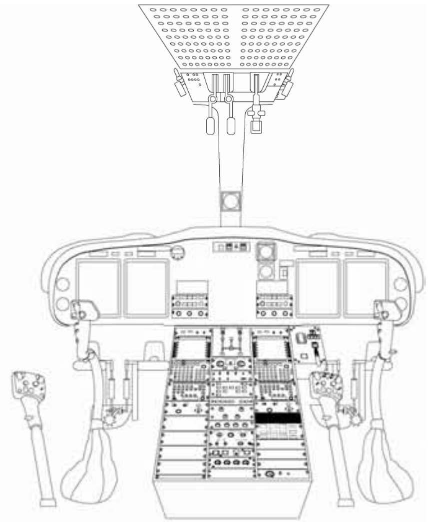
3. OFF/SLOW/FAST rotary switch

OFF ………………. the wipers are off once they reach the parking position

SLOW …………….. the wiping sequence operates with a frequency of 45 cycles per minute

FAST ……………… the wiping sequence operates with a frequency of 90 cycles per minute





WINDSHIELDS WIPER SYSTEM – CONTROLS AND INDICATORS

4. WIPER momentary switch

PRESSED ………. the corresponding wiping sequence starts. To stop the wiping sequence a further action is necessary

NOTE.

When the DUAL mode is selected, a further action on whichever momentary switch (pilot or copilot) will stop both the systems.

WINDSHIELDS WIPER SYSTEM – CONTROLS AND INDICATORS

WINDSHIELDS WIPER SYSTEM – PRINCIPLE OF OPERATIONS

The Windshield Wiper system mode of operation is selected via a three-position rotary switch OFF / LOW SPEED / HIGH SPEED and a two-position rotary switch DUAL / SINGLE located on the WIPER control panel.

The wiper operation (start/stop) is controlled via the WIPER button switch on each cyclic stick.

Whenever the wipers are turned off they automatically reach the park position.

The Pilot wiper motor converter is supplied by MAIN 2 bus via the WIPER PLT circuit breaker.

The Copilot wiper motor converter is supplied by SEC 1 bus via the WIPER CPLT circuit breaker.

The windshield washer system operation is controlled via a push button switch located on the WIPER control panel.

The system is able to supply a cleaning liquid to both wiper blades and includes a low level sensor to detect a minimum liquid quantity which triggers a maintenance event in the Central Maintenance Computer (CMC): the white MAINTENANCE message will be displayed automatically in the CAS.

CHAPTER
31
INDICATING / RECORDING

SECT. 00 - GENERAL

INDICATING/RECORDING SYSTEM – GENERAL

The Indicating/Recording system includes:

- Central Display System (CDS), which integrates on four Display Units (DU) all aircraft data for display, including flight instrumentation, navigational data, system monitoring instrumentation and status, and optional video images
- Central Warning System (CWS), which provides crew with alerts (visual and aural) for dangerous conditions and malfunctions, and with advisory annunciations
- Independent Instruments, i.e. stand alone indicators
- Multi-Purpose Flight Data Recorder (MPFDR) – or Flight Data Recorder / Cockpit Voice Recorder (FDR/CVR) – which automatically records flight data and audio into a crashworthy integrated recorder

CENTRAL DISPLAY SYSTEM – GENERAL

The Central Display System (CDS) is part of the Primus Epic® Integrated Avionics system and shows all the aircraft data to the pilots via four Display Units (DU).

The most important data provided by the CDS are:

- attitude and heading (HDG) from the Attitude and Heading Reference System (AHRS)
- airspeed, vertical speed, and altitude data from the Air Data System (ADS)
- navigation data (such as deviations, bearing, distance, etc.) from the navigational radios and from the Flight Management System (FMS)
- guidance data from the Automatic Flight Control System (AFCS)
- active COM and NAV radio frequencies and ATC Transponder code
- engine and aircraft system parameters (such as NG, NF, NR, torque, temperatures, pressures, etc.)
- engine and aircraft system failure and status indications
- digital map data from the FMS and from Digital Map Generator (if installed)
- weather and ground-map data from the weather radar (WXR) system (if installed)
- terrain data from the Enhanced Ground Proximity Warning System (EGPWS) (if installed)
- traffic data from the Traffic Alert and Collision Avoidance System (TCAS)(if installed)
- video images from FLIR and/or cameras (if installed)

CDS – MAIN COMPONENTS

The CDS main components are:

- four Display Units (DU)
- one DU Dimming control panel (DIM)
- two Cursor Control Devices (CCD)
- two Display Controllers (DC)
- two Remote Instrument Controllers (RIC)
- one Reversion Control Panel (RCP)

DISPLAY UNIT (DU)

The DU is a Line Replaceable Unit (LRU) provided with an 8”×5” color flat-panel Active Matrix Liquid-Crystal Display (AMLCD).

Each DU receives digital data from the Avionics-Standard Communication-Bus version-D (ASCB-D), generates the graphic information and displays them.

Four DU’s are installed on the instrument panel, two in front of the pilot and two in front of the co-pilot and are numbered DU 1 to DU 4 from left to right.

The outboard DU’s (DU 1 and DU 4) automatically operate as Primary Flight Displays (PFD).

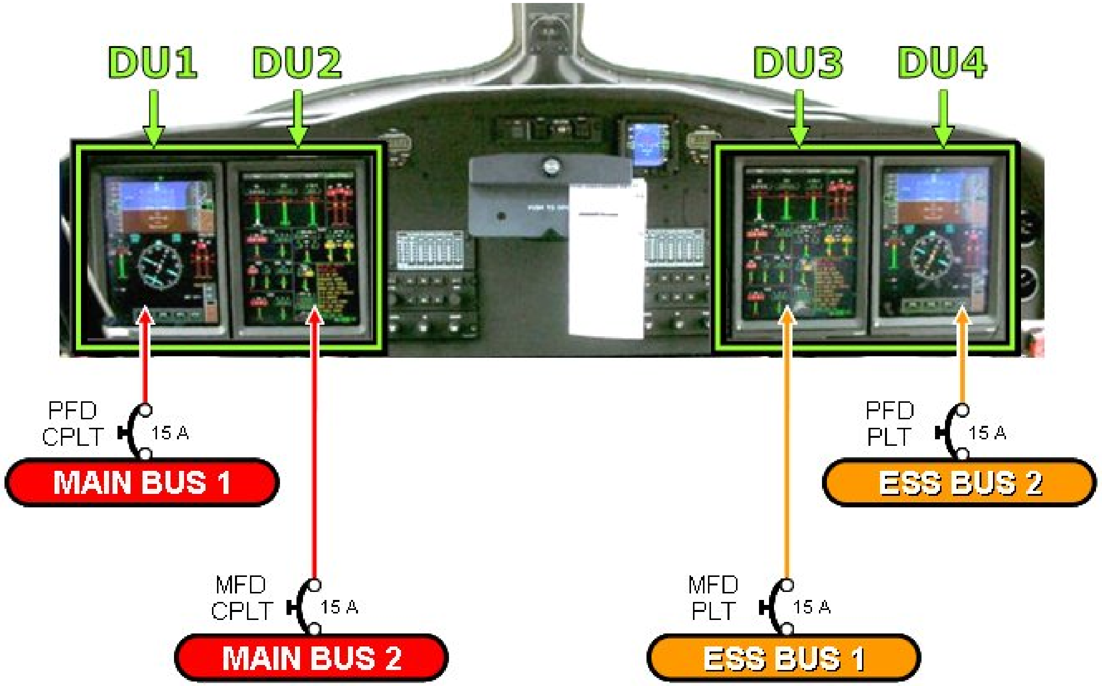
The inboard DU’s (DU 2 and DU 3) automatically operate as Multi-Function Displays (MFD).

The DUs are electrically supplied by different buses:

- DU 1 from MAIN 1 bus via the DISPLAY – PFD CPLT circuit breaker
- DU 2 from MAIN 2 bus via the DISPLAY – MFD CPLT circuit breaker
- DU 3 from ESS 1 bus via the DISPLAY – MFD PLT circuit breaker
- DU 4 from ESS 2 bus via the DISPLAY – PFD PLT circuit breaker

NOTE: Copilot’s DUs power inputs are connected to ESS 1 and ESS 2 buses when the aircraft is on the ground or when starting an engine.

The brightness of each DU is manually adjusted via the DU Dimming control panel located on the central console. Each DU is also provided with an ambient light sensor to automatically adjust the brightness of the LCD as ambient light varies.



DISPLAY UNITS – GENERAL LAYOUT

PRIMARY FLIGHT DISPLAY UNIT (PFD)

The PFD has a single format and its main function is to provide the following:

- ADI (Attitude Director Indicator) display
- HSI (Horizontal Situation Indicator) display
- Airspeed Indicator
- Vertical Speed Indicator
- Barometric Altimeter
- Power Index Indicator (PI)
- Triple Tachometer (NR, NF)
- Navigational data (Source, Ident, Course, Distance, etc)
- Radio Altimeter (RADALT)
- OAT and Wind Vector Indicators
- COMM / NAV / XPDR Frequencies and Code

MULTI FUNCTION DISPLAY (MFD)

The MFD can display different formats that are selectable via the Cursor Control Device (CCD):

- Power Plant (default at power up)
- System
- Plan
- Map

POWER PLANT FORMAT

The main function of the MFD Power Plant format is to provide the following:

Engine primary data (NG, ITT, TQ)

- Triple Tachometer (NR, NF)
- Engine secondary indications (Oil Press and Temp)
- Gearboxes indications (Oil Press and Temp)
- Electrical power system indications (Voltage, Load)
- Hydraulic power system indications (Oil Press and Temp)
- Fuel system indications (Fuel Press, Quantity and Flow)
- Crew Alerting System (CAS) messages

SYSTEM FORMAT

Different sub-formats are available to display different system data; selection is performed via the CCD.

Standard System sub-formats are:

- “Electrical” (Electrical power system synoptic diagram)
- “Hydraulic” (Hydraulic power system synoptic diagram)
- “Flt Contr” (AFCS indications)
- “Maintenance” (Central Maintenance Computer data)
- “Sys Config” (Configuration management data)
- Time/Date
- Config

Note: "Maintenance" and "Sys Config" formats are designed for maintenance purposes and can only be selected on the ground.

In case the helicopter is fitted with optional systems that output a video signal, additional sub-formats will be available (Eg: Cameras, FLIR, Digital Map, etc.).

All the sub-formats that are selectable during flight show the relevant system data in the top half of the MFD; the bottom half shows reduced Power Plant format data.

PLAN FORMAT

The main function of the MFD Plan format is to display a North-Up synthetic map based on FMS data in the top half of the MFD. The bottom half of the MFD shows reduced Power Plant format data.

MAP FORMAT

The main function of the MFD Map format is to display a Heading-Up synthetic map based on FMS data in the top half of the MFD. The bottom half of the MFD shows reduced Power Plant format data.

COMPOSITE FORMAT

Any DU shows the Composite format whenever the paired DU is off, so as to provide pilot or co-pilot with minimum essential information for safe flight with a single DU.

The Composite format can either be automatically or manually selected.

Automatic reversion to Composite format occurs when the paired DU becomes inactive on the ASCB-D and latches to prevent flashing due to intermittent failures.

Manual reversion can be selected via the Reversion Control Panel (RCP) PLT or CPLT selectors: this causes the non-selected paired DU to turn off.

When the PLT or CPLT selectors are returned to NORM, the automatic reversion operation is reset.

Composite format is a reduced PFD format with the addition of the following:

- Crew Alerting System (CAS) messages
- Pressure readouts (Engines, Main Gearbox, Hydraulic power systems)
- Fuel Quantity indicator

DU DIMMING CONTROL PANEL (DIM)

DU Dimming control panel allows adjusting the LCD brightness of the two PFD and two MFD individually by means of four potentiometers.

CURSOR CONTROL DEVICES (CCD)

CCD is used to manage data and control the DU's. The control panel has two DISPLAY SELECT buttons, the SET control (dual concentric knobs), a joystick and an ENTER button.

The joystick is used to move the cursor through the MFD and operate the MFD designator and radio tuning on the PFD.

REMOTE INSTRUMENT CONTROLLER (RIC)

The two RIC are used to set Course, Decision Height (DH) and Heading and to perform Radio Altimeter self-test by means of three control knobs with associated buttons.

All RIC selections except Heading Set are independent for pilot and co-pilot.

DISPLAY CONTROLLER (DC)

The two DC provide pilot/co-pilot independent selection of:

- Barometric pressure reference
- Selected Altitude
- Short-range and long-range navigation sources
- On-side/cross-side bearing sources
- HSI format for display on the associated PFD

The DC also provides for interface of the on-side Cursor Control Device (CCD) and Remote Instrument Controller (RIC) to the MAU's.

Three buttons labelled HSI, MAP, and WX (or WX/TERR) operate together to control the format of the HSI.

The HSI formats that can be selected are the following:

FULL (Default format)	Displays a conventional HSI with a full 360° compass rose
ARC	Displays an expanded arc of the compass rose extending to ±45° across heading
HOV (*)	Displays the HSI HOV mode with a velocity vector
HOV+360° MAP (*)	Displays HOV mode with velocity vector with 360° FMS map overlay
ARC+MAP	Displays the ARC mode with an FMS map overlay
ARC+WX (**)	Displays the arc format with a WX overlay
ARC+TERR (***)	Displays the arc format with EGPWS terrain overlay
ARC+MAP+WX (**)	Displays the arc mode with FMS map and WX overlays
ARC+MAP+TERR (***)	Displays the arc format with FMS map and EGPWS terrain overlays

Notes:
(*) Available only if Enhanced AFCS is installed.
(**) Available only if Weather Radar (WX) is installed.
(***) Available only if Enhanced Ground Proximity Warning System (EGPWS) is installed.

CCD, RIC AND DC INTERFACE

CCD, RIC and DC panels constitute a set of control panels for the DUs.

CCD and RIC are interfaced with the DC panel; the DC interfaces the set of panels to the MAUs.

The co-pilot set of panels is supplied by MAIN 1 bus and protected by DISPLAY – PFD CPLT CONTR circuit breaker.

The pilot set of panels is supplied by ESS 1 bus and protected by DISPLAY – PFD PLT CONTR circuit breaker.

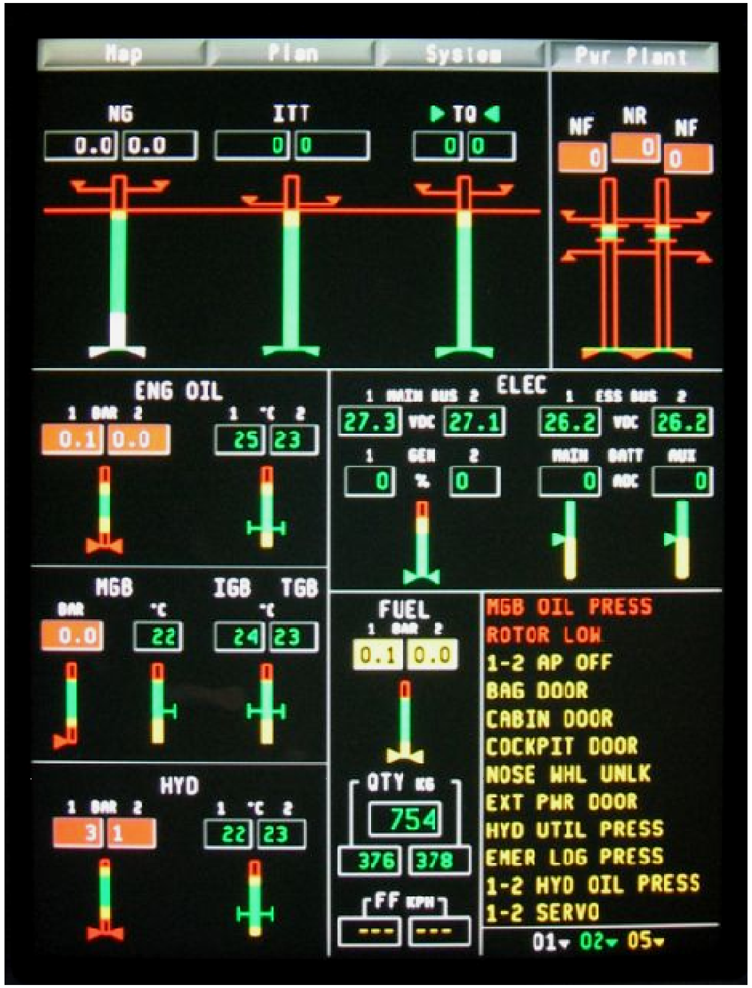
REVERSION CONTROL PANEL (RCP)

RCP permits manual reversion to single-unit operation for DU's in pilot's and co-pilot's stations, and manual reversion to single-source display for the Air Data System (ADS) and the Attitude and Heading Reference System (AHRS).

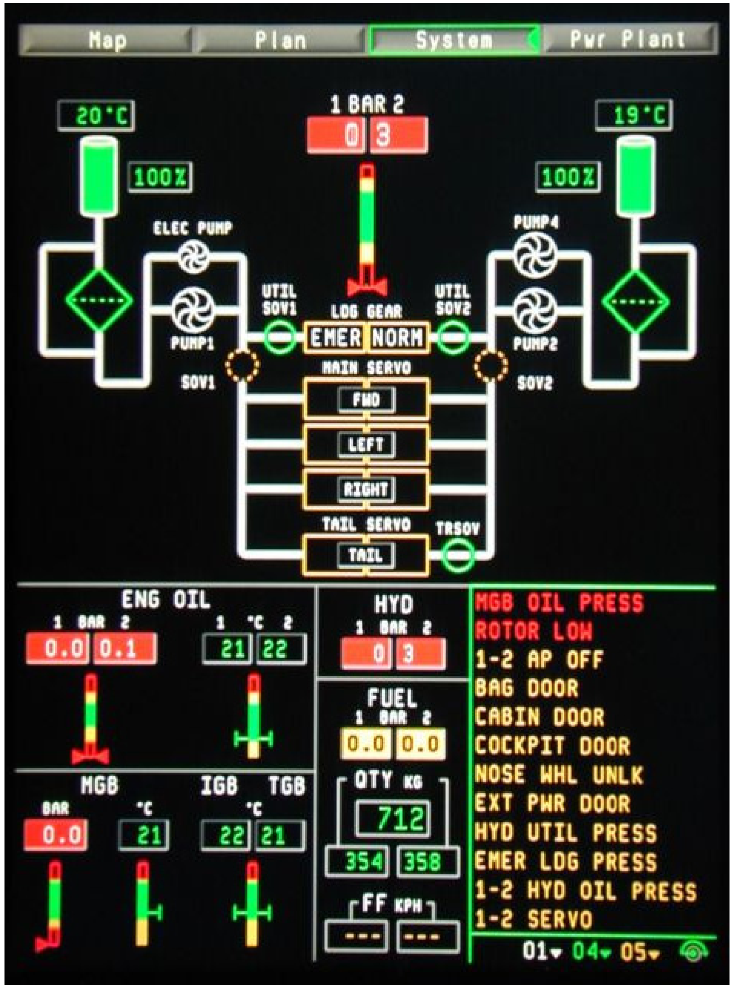
Such manual reversion actions permit recovering from single unit/system failure where loss of important data on displays occurs.

DISPLAY UNIT (DU)

PRIMARY FLIGHT DISPLAY (PFD)

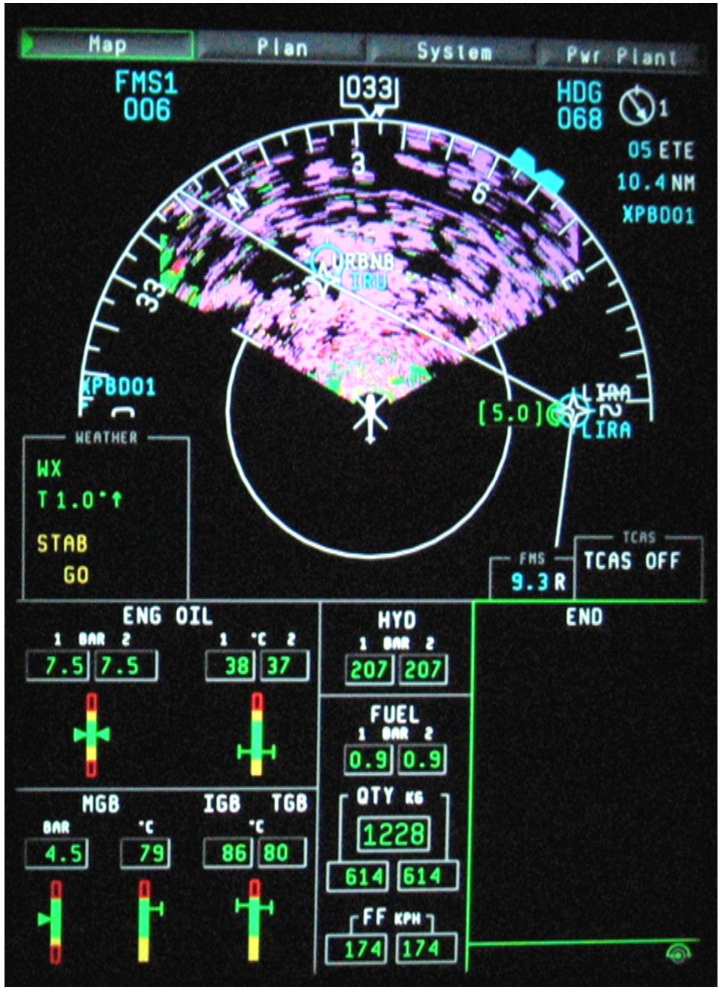


POWER PLANT

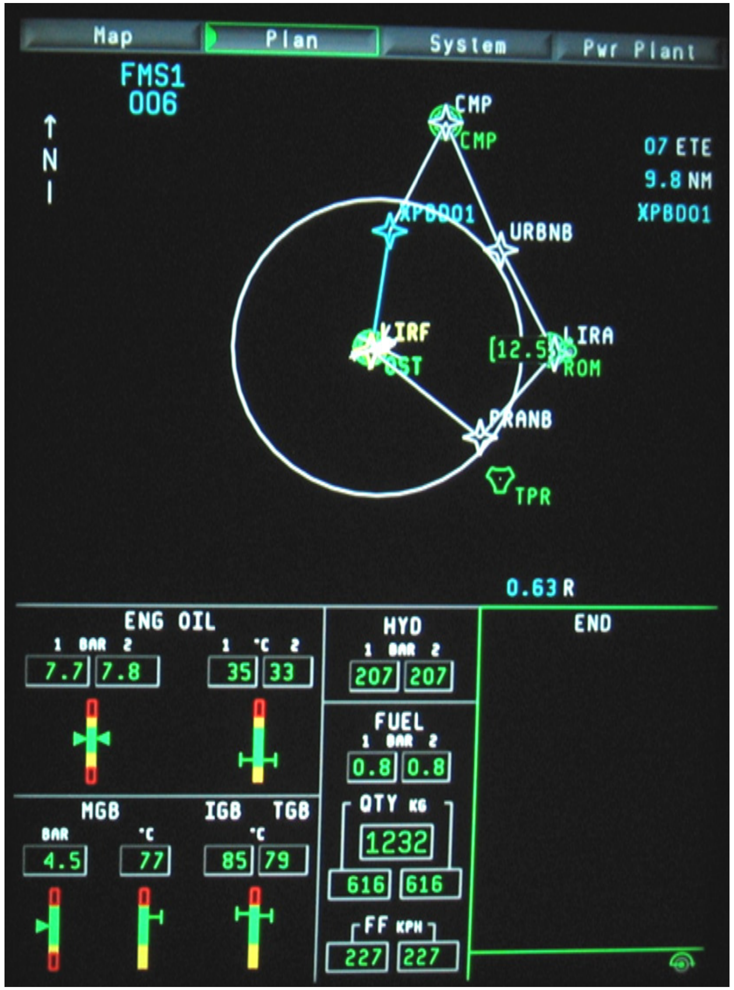


SYSTEM SYNOPTIC DIAGRAM

MULTI-FUNCTION DISPLAY (MFD) FORMATS



MAP (FMS, WX, TCAS)



PLAN (FMS)

MULTI-FUNCTION DISPLAY (MFD) FORMATS

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COMPOSITE FORMAT

CENTRAL DISPLAY SYSTEM - CONTROLS AND INDICATORS

CURSOR CONTROL DEVICE (CCD)

1. DISPLAY SELECT pushbutton

   LH pressed ... activates the cursor(s) on the LH display of the relevant pilot's station (PLT = MFD; CPLT = PFD)

   RH pressed ... activates the cursor(s) on the RH display of the relevant pilot's station (PLT = PFD; CPLT = MFD)

2. ENTER pushbutton

   MFD selected. executes the highlighted menu selection or toggles between highlighted menu options; deactivates the designator (Map or Plan formats)

   PFD selected.. toggles the highlighted active/standby values

3. JOYSTICK

   MFD selected. controls cursor position in the drop-down menu or the designator position (when active)

   PFD selected.. controls cursor position in the COMM/IDENT window

4. SET inner knob

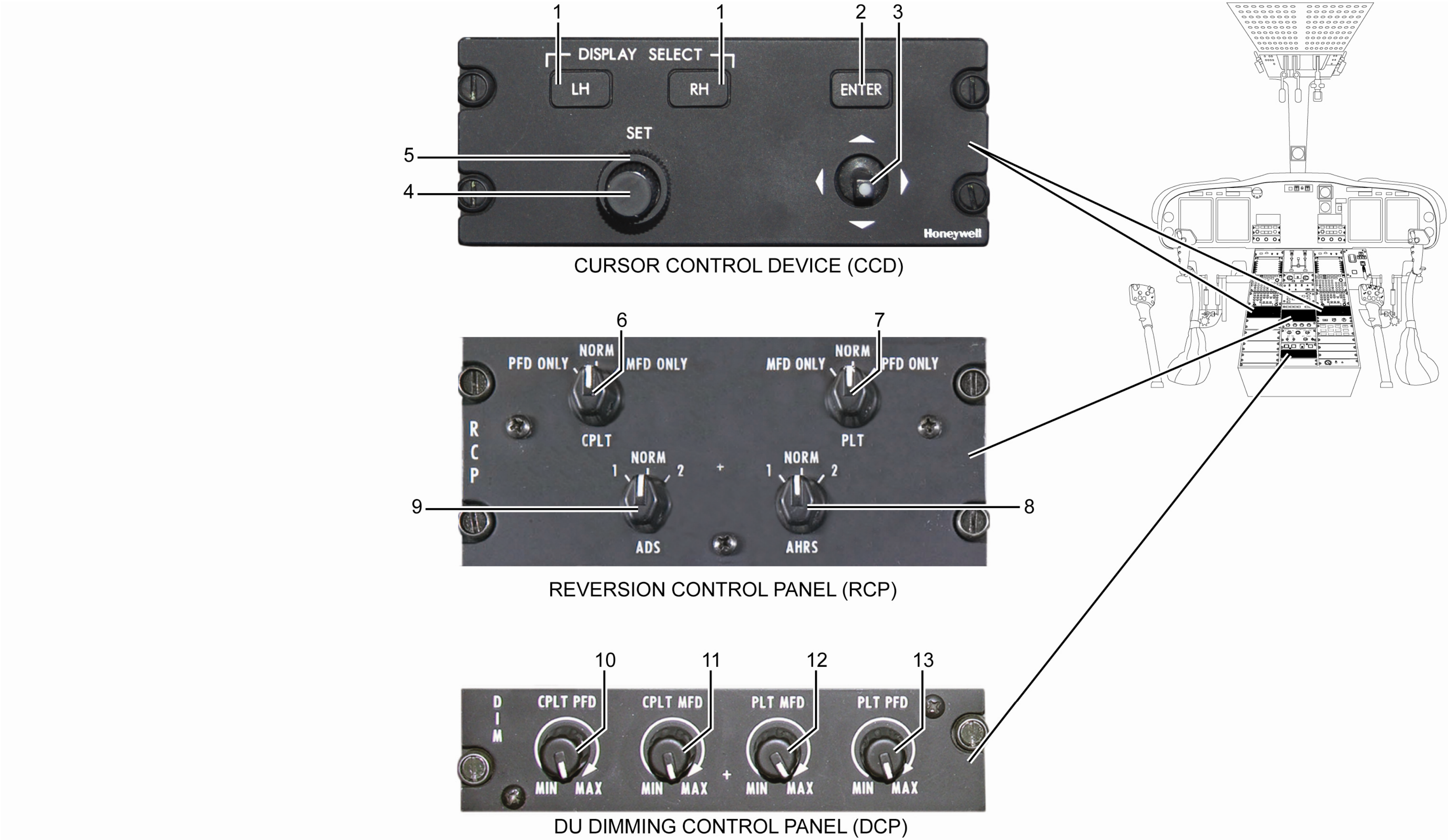
   MFD selected. scrolls the CAS window messages

   PFD selected.. adjusts decimals of the highlight standby frequency or the last two digits of XPDR code

5. SET outer knob

   MFD selected. sets the scale range (Map or Plan formats) or the highlighted value in the drop-down menu

   PFD selected.. adjusts units of the highlight standby frequency or the first two digits of XPDR code



CENTRAL DISPLAY SYSTEM – CONTROLS AND INDICATORS (1 OF 6)

REVERSION CONTROL PANEL (RCP)

6. CPLT selector

NORM ………	automatic reversion selected
PFD ONLY ….	the copilot MFD is turned off; the copilot PFD is set to composite format
MFD ONLY …	the copilot PFD is turned off; the copilot MFD is set to composite format

7. PLT selector

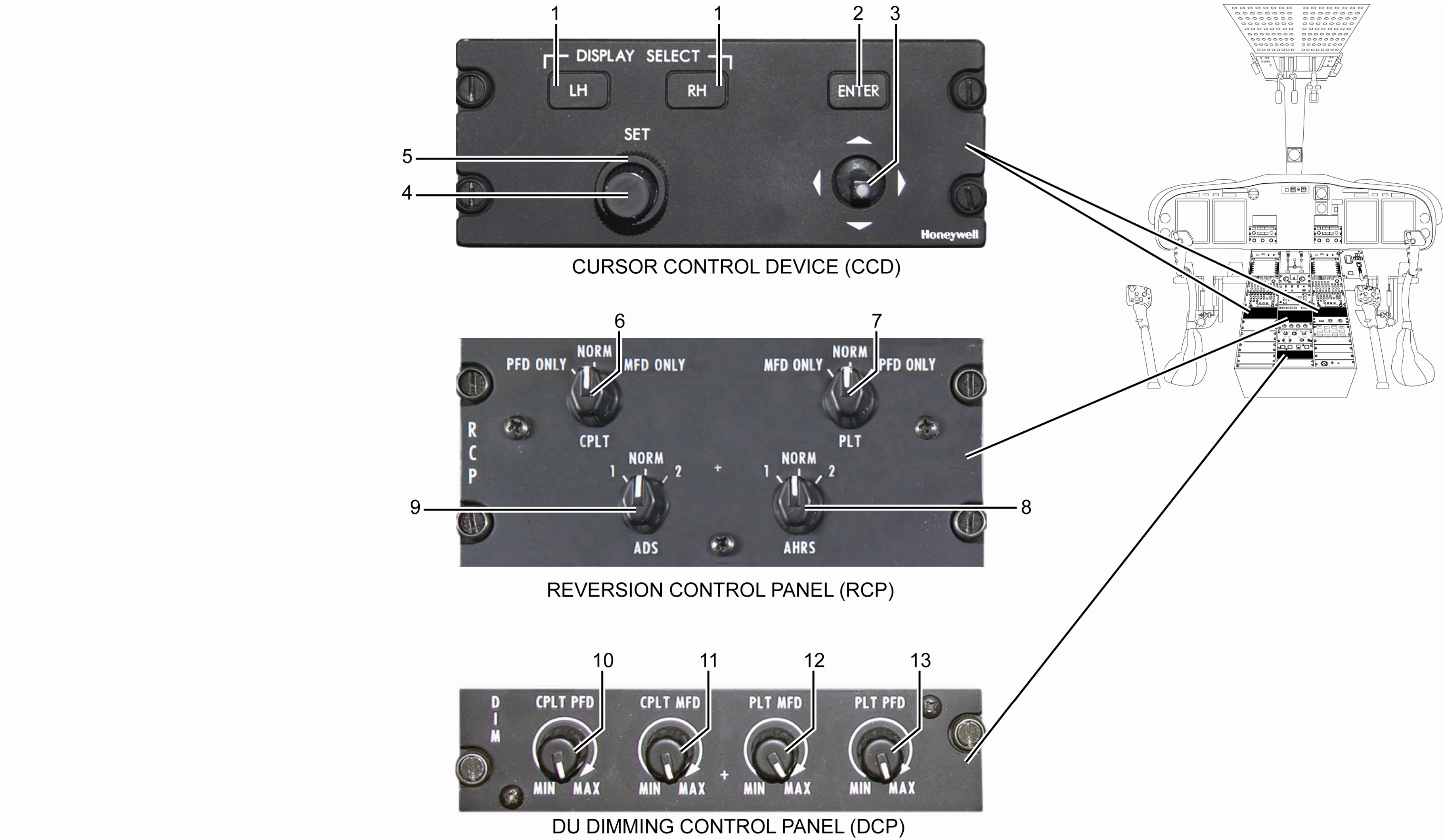
NORM ………	automatic reversion selected
PFD ONLY ….	the pilot MFD is turned off; the pilot PFD is set to composite format
MFD ONLY …	the pilot PFD is turned off; the pilot MFD is set to composite format

8. AHRS selector

NORM ………	each pilot's station displays the on-side AHRS data
1 ……………..	both pilot's stations display the AHRS 1 data
2 ……………..	both pilot's stations display the AHRS 2 data

9. ADS selector

NORM ………	each pilot's station displays the on-side ADS data
1 ……………..	both pilot's stations display the ADS 1 data
2 ……………..	both pilot's stations display the ADS 2 data



CURSOR CONTROL DEVICE (CCD)

REVERSION CONTROL PANEL (RCP)

DU DIMMING CONTROL PANEL (DCP)

CENTRAL DISPLAY SYSTEM – CONTROLS AND INDICATORS (2 OF 6)

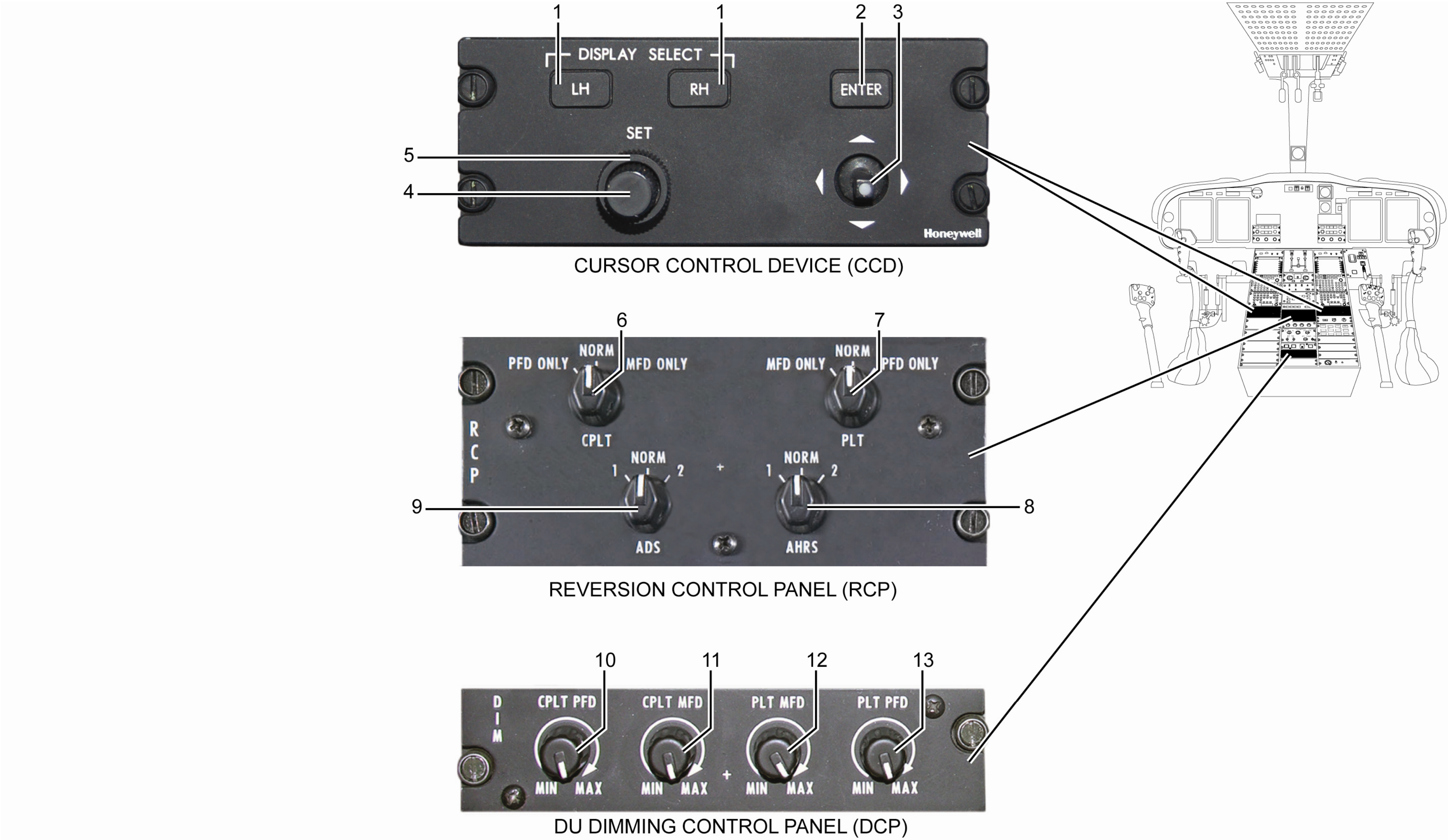
DU DIMMING CONTROL PANEL

10. CPLT PFD potentiometer
    MIN ………. set the copilot PFD to the minimum brightness value
    MAX ………. set the copilot PFD to the maximum brightness value

11. CPLT MFD potentiometer
    MIN ………. set the copilot MFD to the minimum brightness value
    MAX ………. set the copilot MFD to the maximum brightness value

12. PLT MFD potentiometer
    MIN ………. set the pilot MFD to the minimum brightness value
    MAX ………. set the pilot MFD to the maximum brightness value

13. PLT PFD potentiometer
    MIN ………. set the pilot PFD to the minimum brightness value
    MAX ………. set the pilot PFD to the maximum brightness value



CENTRAL DISPLAY SYSTEM – CONTROLS AND INDICATORS (3 OF 6)

REMOTE INSTRUMENT CONTROLLER (RIC)

14. COURSE select knob

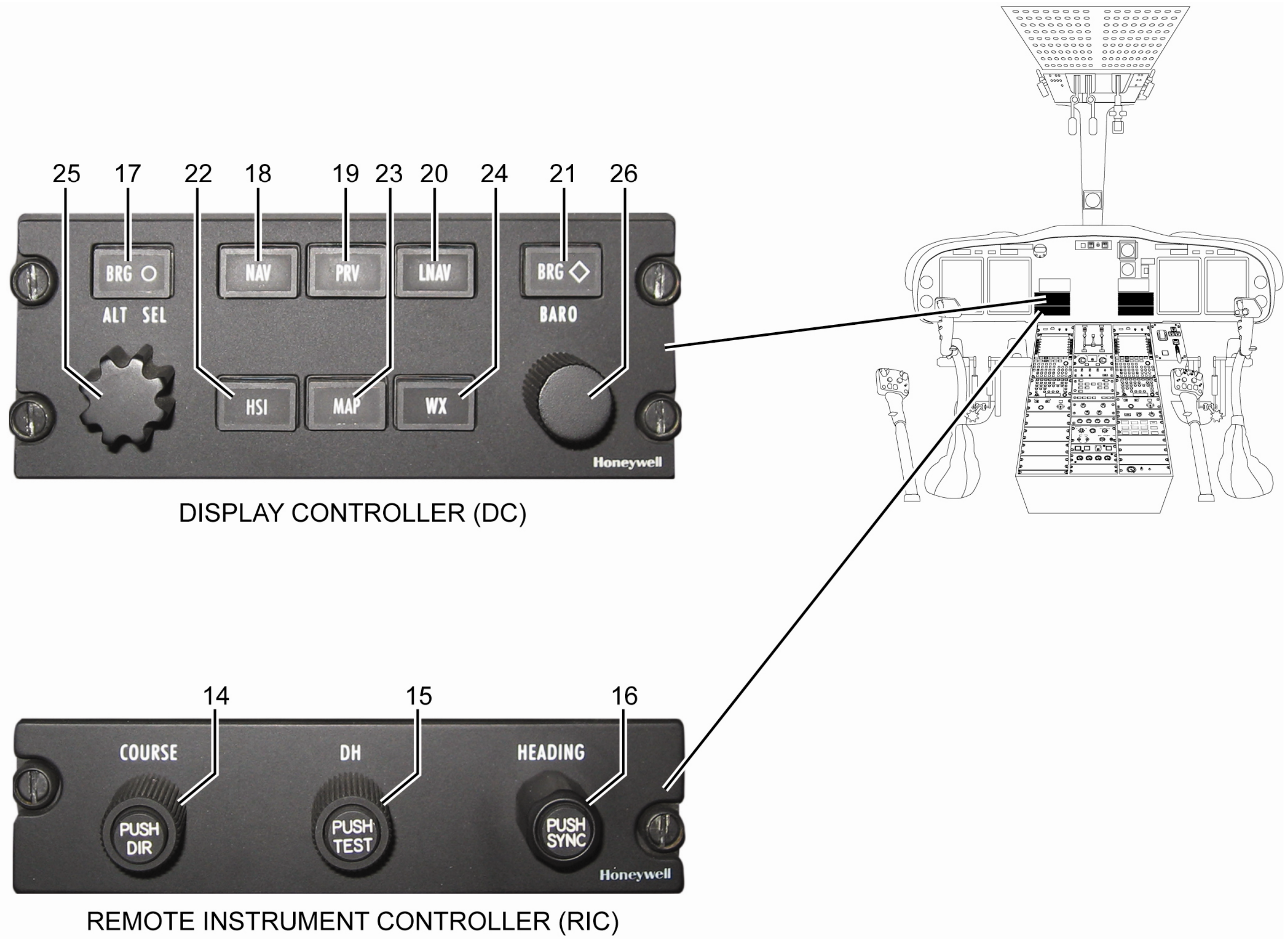
rotated …….	selects the desired course when VOR/LOC is selected as the Primary Navaid or as the PRV (Preview) on the on-side HSI
PUSH DIR button pressed …...	selects the Direct Course, i.e. synchronizes the selected course to the VOR bearing, when VOR is selected as the Primary Navaid or as the PRV (Preview) on the on-side HSI

15. DH select knob

rotated ……	selects the value of decision height which is displayed on the on-side PFD
PUSH TEST button pressed ……	activates the test of the radio altimeter (in case of single radio altimeter installation) or of the on-side radio altimeter (in case of dual radio altimeter installation)

16. HEADING select knob

rotated ……	controls the Heading Set bug on both HSI's.
PUSH SYNC button pressed……	synchronizes the selected Heading Set bug on both HSI's to the current aircraft heading of the coupled attitude and heading reference system (AHRS) or to the heading of the on-side AHRS when the flight director is not coupled



CENTRAL DISPLAY SYSTEM – CONTROLS AND INDICATORS (4 OF 6)

DISPLAY CONTROLLER (DC)

17. Circle BRG (LH) pushbutton

pressed ….. selects VOR1 for display on circle (white) bearing pointer on the on-side HSI. Subsequent presses toggle between the different sources following the sequence: OFF → VOR1 → ADF1 → DF (if installed) → FMS1 → OFF

18. NAV pushbutton

pressed ….. selects the on-side short-range navigation source (VOR/LOC) as the Primary Navaid on the on-side PFD. Subsequent presses toggle between the on-side and cross-side VOR/LOC.

19. PRV (Preview) pushbutton

pressed...…. when LNAV is selected as the Primary Navaid allows to preview the on-side VOR/LOC deviation to be displayed on a secondary CDI on the on-side HSI. Subsequent presses toggle between the different sources following the sequence: PRV OFF → on-side VOR/LOC → cross-side VOR/LOC → PRV OFF.

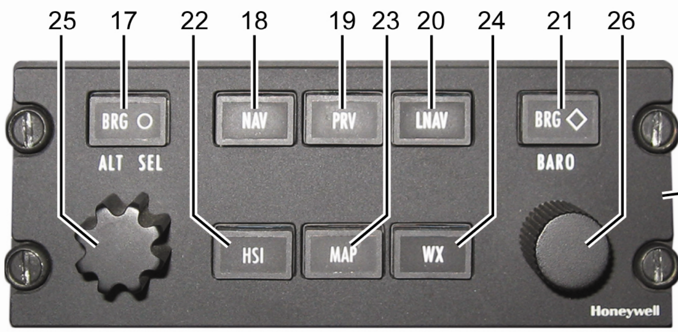
Note: ineffective if pressed when short-range navigation source (VOR/LOC) is selected as the Primary Navaid

20. LNAV pushbutton

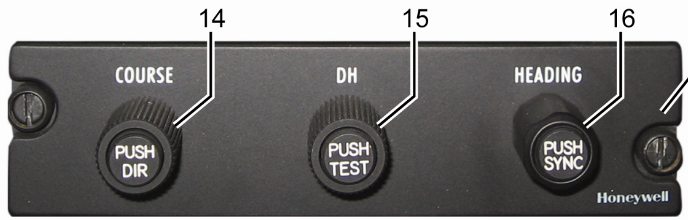
pressed...…. selects the on-side long-range navigation source (FMS) as the Primary Navaid on the on-side PFD. Subsequent presses toggle between the on-side and cross-side FMS.

21. Diamond BRG (RH) pushbutton

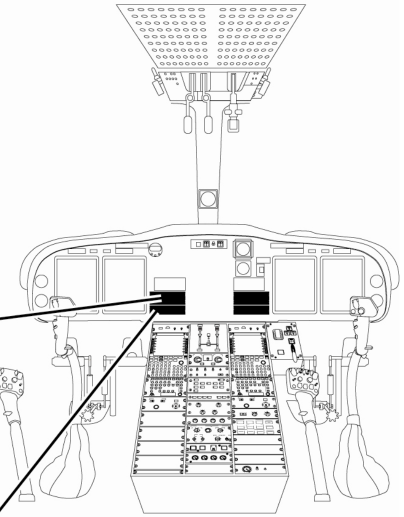
pressed...…. selects VOR2 for display on the diamond (green) bearing pointer on the on-side HSI. Subsequent presses toggle between the different sources following the sequence: OFF → VOR2 → ADF2 → DF (if installed) → FMS2 → OFF



DISPLAY CONTROLLER (DC)



REMOTE INSTRUMENT CONTROLLER (RIC)



CENTRAL DISPLAY SYSTEM – CONTROLS AND INDICATORS (5 OF 6)

22. HSI pushbutton

pressed....... toggles between FULL Compass and ARC modes on the on-side HSI.
If FD HOV mode is available, it toggles between HSI modes in the following sequence:
FULL Compass → ARC → HOV → FULL Compass
Note: if HOV mode is engaged, the HSI button is ineffective

23. MAP pushbutton

pressed....... selects ARC+MAP mode on the on-side HSI.
Subsequent presses toggle MAP on and off.

24. WX or WX/TERR pushbutton

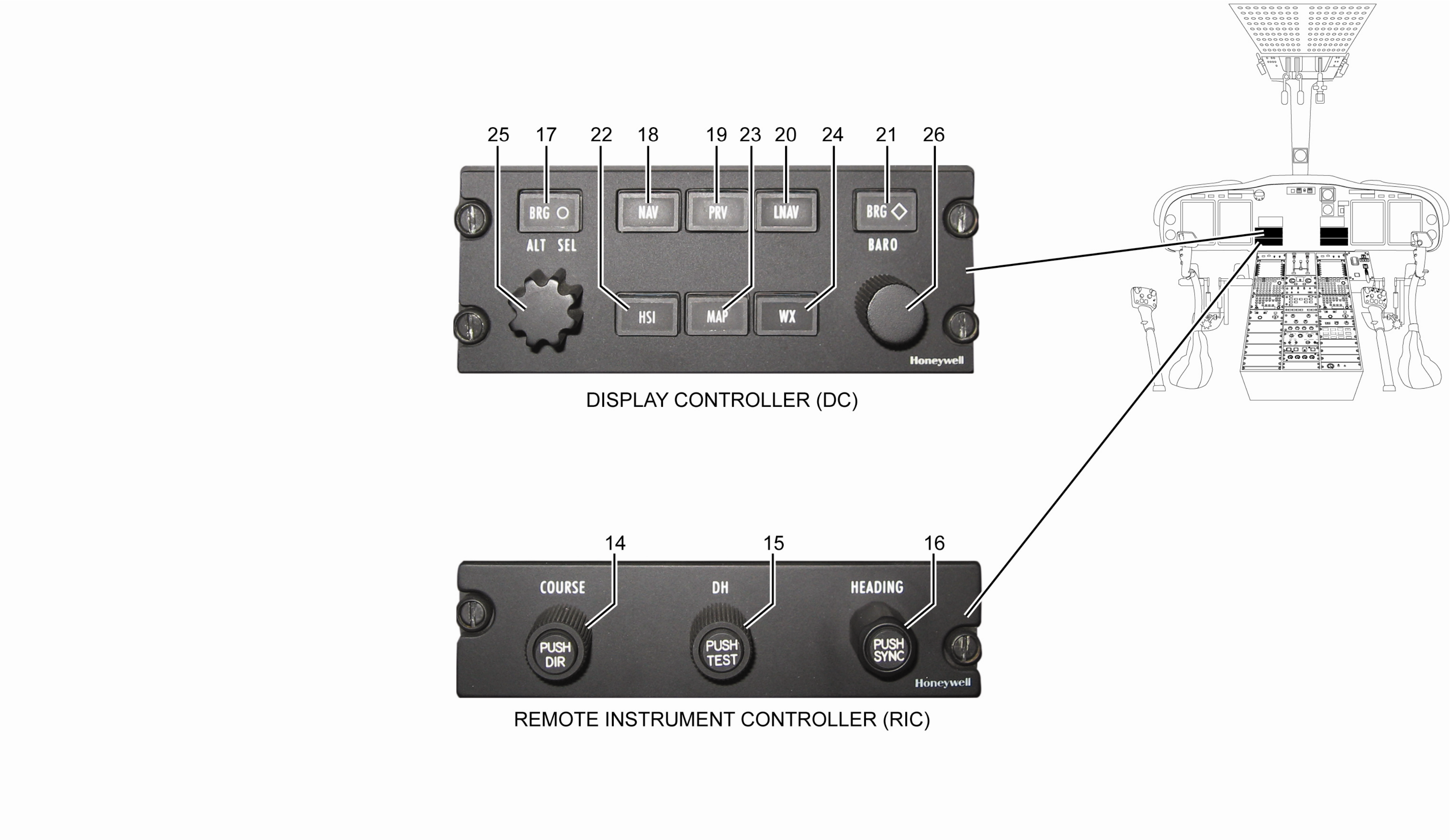
pressed....... (if Weather Radar is installed) selects ARC+WX mode on the on-side HSI.
Subsequent presses toggle WX on and off, or between HSI modes in the following sequence:
ARC → ARC+WX → ARC+TERR (is EGPWS is installed) → ARC

25. ALT SEL knob

rotated....... sets the Selected Altitude reference (bug + digital readout) on both barometric altimeters

26. BARO knob

rotated....... sets barometric pressure reference for the on-side barometric altimeter



CENTRAL DISPLAY SYSTEM – CONTROLS AND INDICATORS (6 OF 6)

CDS – OPERATION

CURSORS

Up to three different cursors can be displayed on one DU in both pilots' stations at a time.

Pilot and co-pilot can independently control cursors using their own CCD.

At power up cursors are automatically selected on the two MFD's.

Cursors can be toggled between PFD and MFD by using the DISPLAY SELECT buttons.

Cursors are typically shown as green boxes and respond either to the joystick and the associated ENTER button or to any of the SET concentric knobs.

MFD FORMAT SELECTION

The top line of the MFD shows four virtual pushbuttons where a green box cursor can be displayed.

The virtual pushbuttons are labelled as follows:

- Map
- Plan
- System
- Pwr Plant

At power-up, the Pwr Plant format is automatically selected.

The cursor can be positioned on any of the virtual pushbuttons using the joystick on the CCD and then, pressing the ENTER pushbutton on the CCD, the associated format is selected for the MFD: the corresponding pushbutton is shown as "pressed".

MFD DROP-DOWN MENUS

Each MFD virtual pushbutton has an associated drop-down menu that can be displayed by moving the joystick down.

Joystick is then used to highlight the desired option and the ENTER pushbutton to select it.

Quick escape from drop-down menu display is obtained by moving the joystick to the left or to the right.

Some drop-down menu options operate as toggle switches: a black box adjacent to the option label can be checked or unchecked by pressing the ENTER pushbutton (e.g.: PwrPlant/Analog).

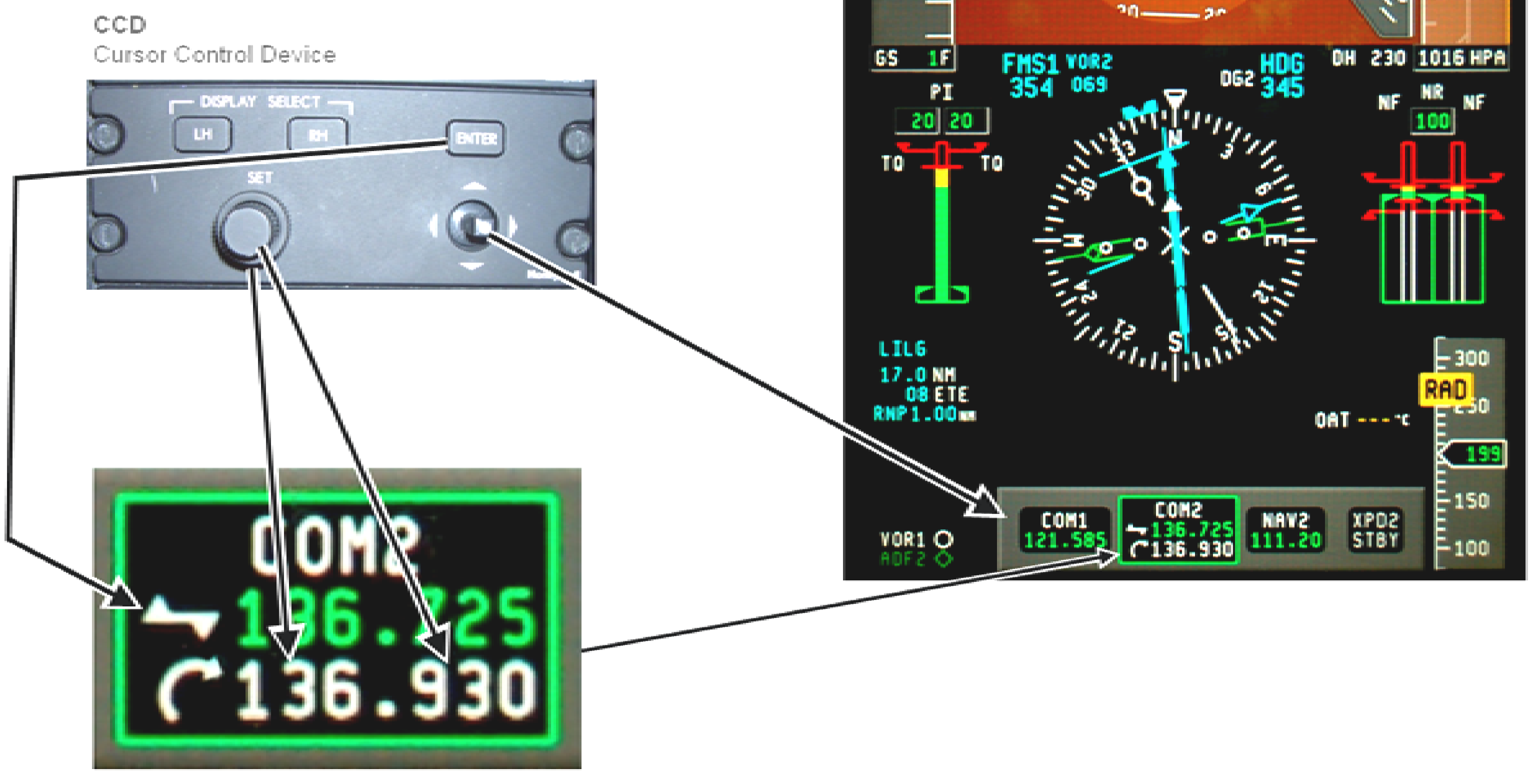
Some drop-down menu options are mutually exclusive selections: a black dot is displayed adjacent to each option label involved. Pressing the ENTER pushbutton to select an option (black dot changes into green dot), deselects the others (e.g.: Map/TAWS, Map/Weather, Map/Off).

Some drop-down menu options show a LH or RH white arrow: moving the cursor in the direction of the arrow a submenu opens to show additional options (e.g.: System/Config, Map/Traffic).

Cursor Control Device

Eg: Pilot's Station

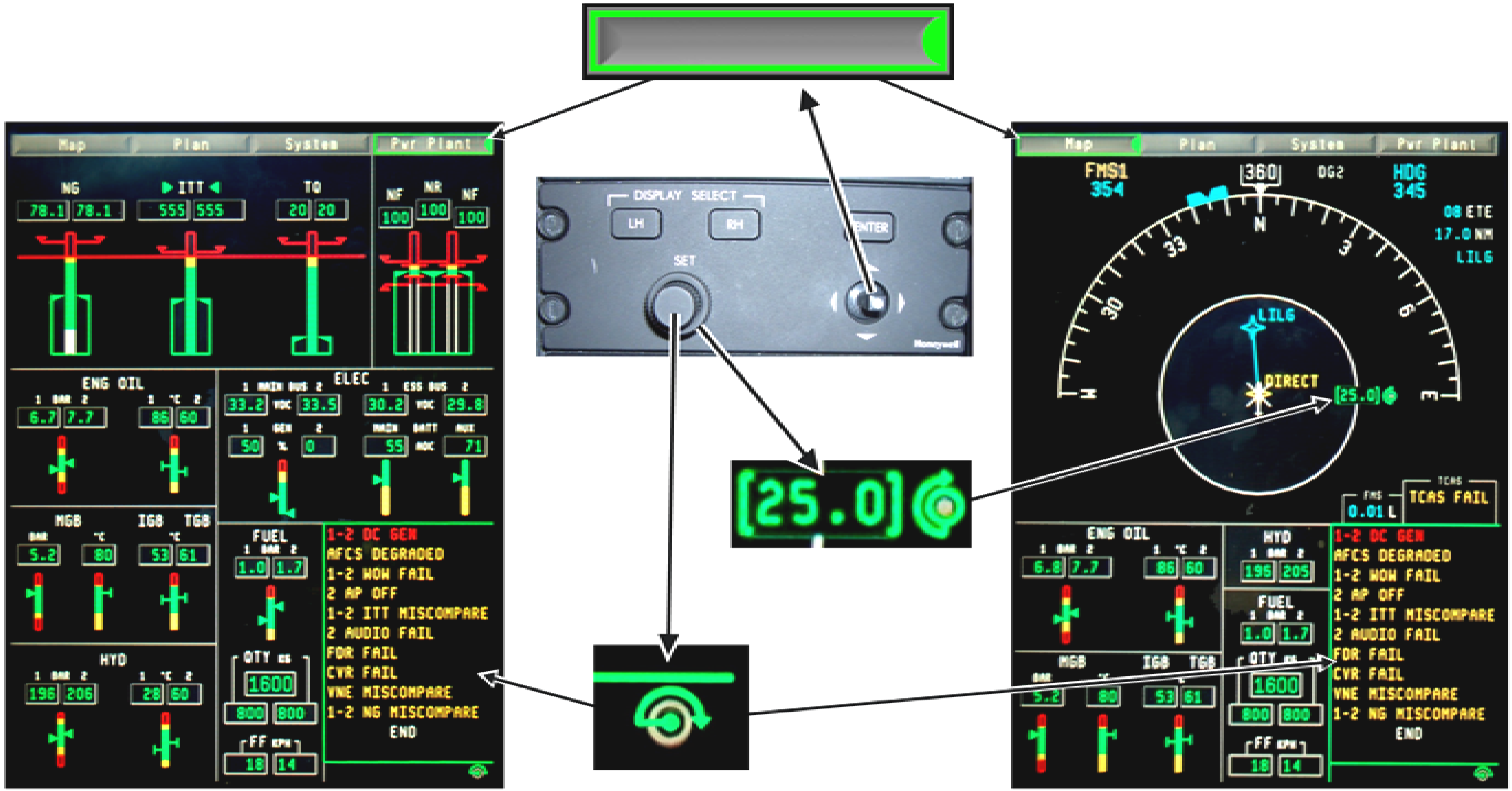
CURSORS DISPLAY SELECT


CCD
Cursor Control Device
DISPLAY SELECT
LH
RH
ENTER
SET
COM2
136.725
136.930
FD FAIL
TCAS FAIL
TAWS FAIL
FMS1 354
VOR2 069
HDG 345
DH 230
1016 HPA
COM1 121.585
NAV2 111.20
XPD2 STBY
VOR1
ADF2
RAD
OAT

PFD CURSORS


CCD
Cursor Control Device
DISPLAY SELECT
LH
RH
ENTER
SET
[12.5]
FMS2 VOR2
084 039
079
HDG
257
PI
TQ
NF
NR
NF
SRN
9.0 NM
04 ETE
RNP1.00
RAD1
OAT 21 °C
30.04 IN
GS ---
1500
1000
COM1
118.100
COM2
124.22
NAV2
111.20
XPDR
4320

PFD CURSORS


Map
Plan
System
Pwr Plant
NG
ITT
TQ
NF
NR
ENG OIL
ELEC
MGB
IGB
TGB
HYD
FUEL
QTY
FF
1-2 DC GEN
AFCS DEGRADED
1-2 WOW FAIL
2 AP OFF
1-2 ITT MISCOMPARE
2 AUDIO FAIL
FDR FAIL
CVR FAIL
VNE MISCOMPARE
1-2 NG MISCOMPARE
END
DISPLAY SELECT
LH
RH
SET
CENTER
FMS1
354
360
DG2
HDG
345
08 ETE
17.0 NM
LIL6
DIRECT
[25.0]
TCAS FAIL

MFD CURSORS

MFD DROP-DOWN MENUS

SYSTEM / TIME/DATE

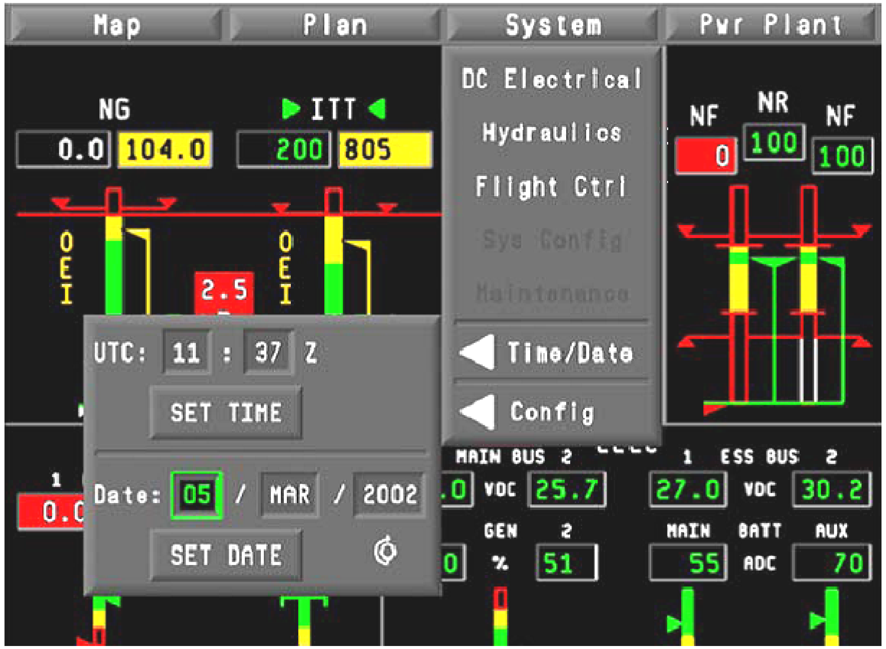
Moving the joystick in the direction of the arrow when cursor is on the Time/Date option of the System drop-down menu opens a sub-menu that shows the Primus EPIC® real time clock; the Time/Date submenu also permits manual setting of time and date.

When the GPS receiver is valid and is receiving satellite data, the system clock automatically synchronizes with the time and date from the GPS receiver.

When the system time is synchronized with the GPS, both SET TIME and SET DATE prompts are displayed in gray to indicate that pilots cannot manually update the time and date.

When the system time and date are not synchronized with the GPS, the pilot (or co-pilot) can manually set them by:

- moving the cursor to the parameter that needs modification
- rotating the SET knob on the CCD to change values
- pressing the ENTER on the CCD with cursor on SET TIME or SET DATE, as appropriate, to save the value



When a pilot selects the Time/Date submenu, the submenu is grayed out on the other MFD.

SYSTEM / CONFIG

Moving the joystick in the direction of the arrow when cursor is on the Config option of the System drop-down menu opens a sub-menu that permits setting of the following:

- Metric Altimeter checkbox
- Fuel – Unit for Weight data (KG or LBS)
- Baro – Unit for Barometric Pressure Reference (IN or HPA)
- ITT – Unit for ITT Indicator (°C or %)

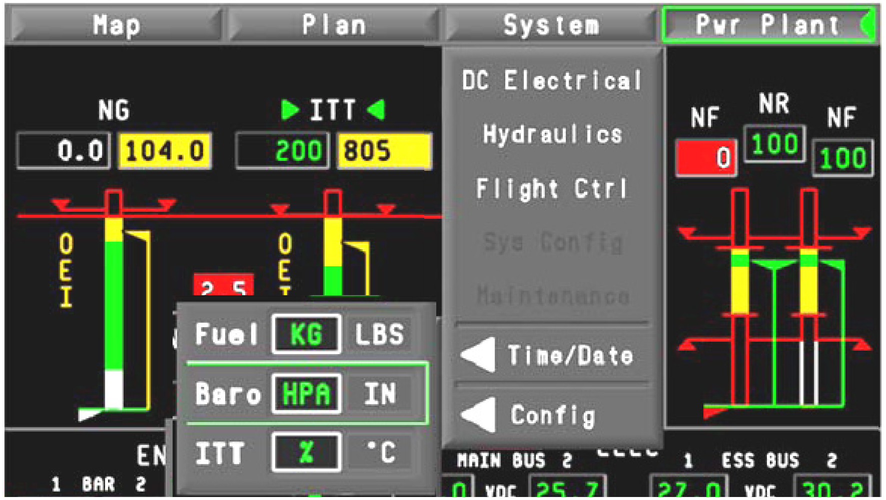
The Metric Altimeter is displayed as an additional digital readout in the top portion of the Barometric Altimeter when the option box is checked.

The selection of Weight data unit affects the following:

- Fuel Quantity Indicator
- Fuel Flow Indicator: unit will be KPH (kg per hour) or PPH (pounds per hour)
- Weight data readouts on the MCDU (see FMS in chapter 34-00)

Changes to the selections are made by using the cursor to highlight the parameter to be configured, and by using the ENTER button on the CCD to toggle between the available states.

When a pilot selects the Config submenu, the submenu is grayed out on the other MFD.



The selections made by a pilot affect all DUs and are memorized (i.e. the configuration selected before removing power is shown again at the next power-up).

SYSTEM / SYS CONFIG

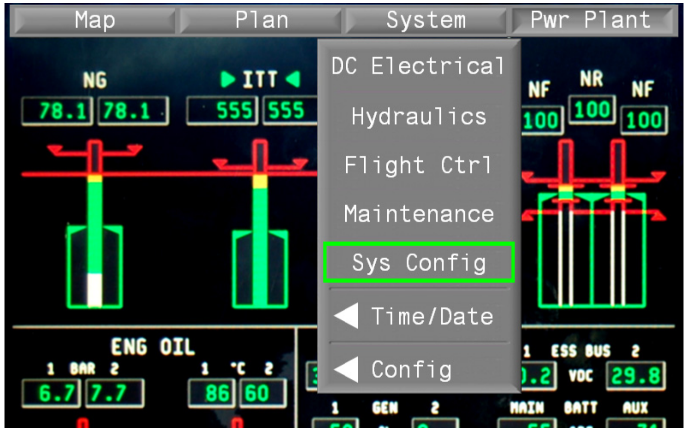
The Software Configuration Monitoring System (SCMS) is a function of the MAU that performs automatic monitoring of the hardware and software configuration items that are connected to the Avionics Standard Communication Bus (ASCB).

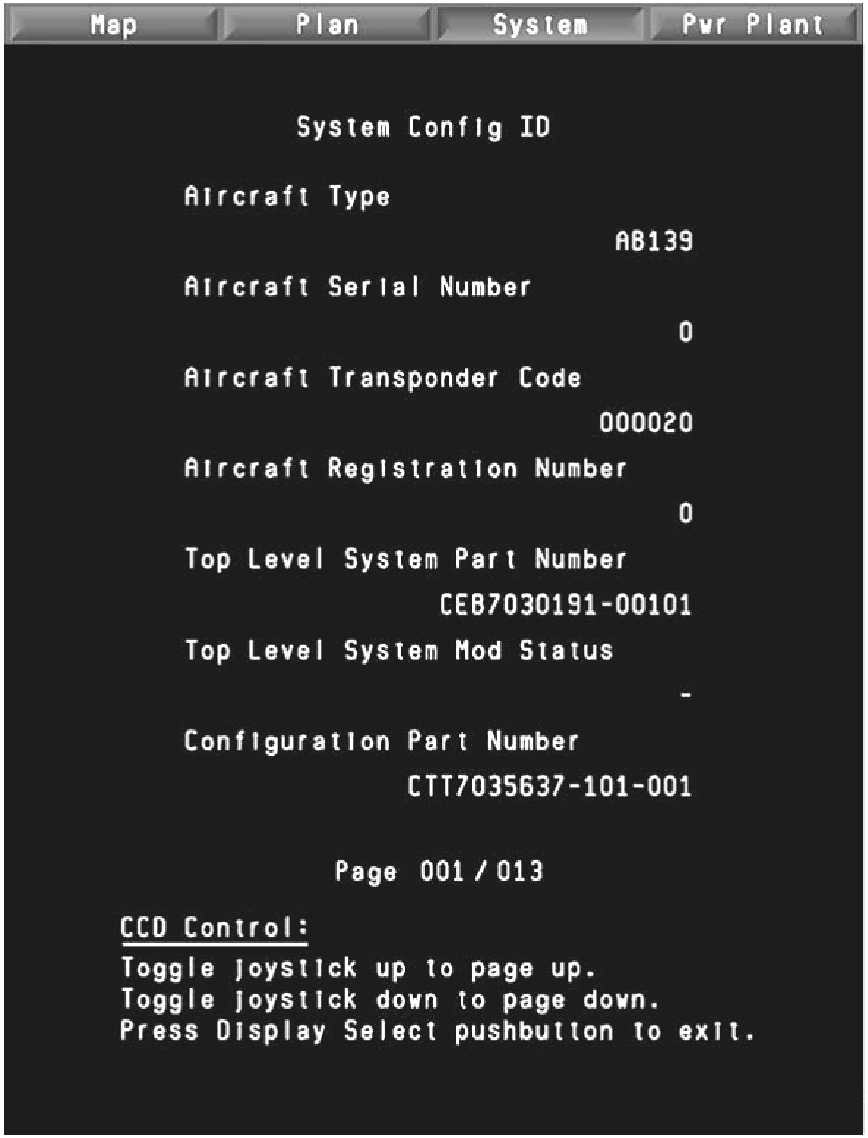
In case new software or hardware has been loaded in the system and the new configuration has not been validated, the VALIDATE CONFIG caution message is displayed in the CAS window.

Configuration validation is then required.

Pilot or copilot selects the Configuration window via the System / Sys Config drop-down menu to display the current configuration for verification.

When a pilot selects the Configuration submenu, the submenu is grayed out on the other MFD.



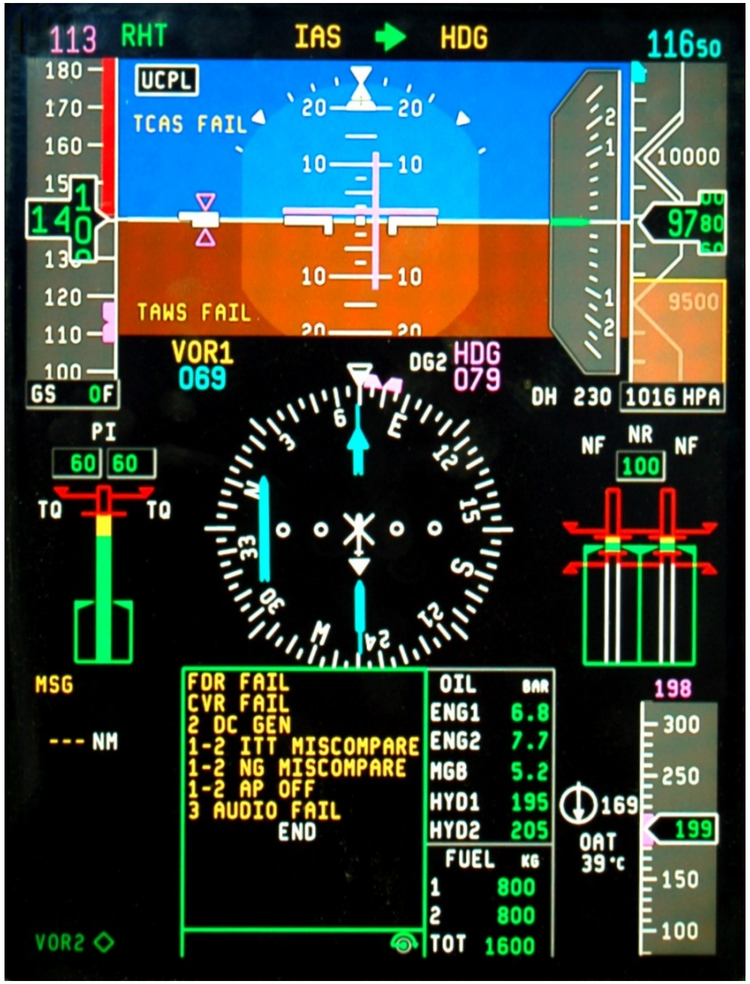


DISPLAY UNIT COLOR USAGE

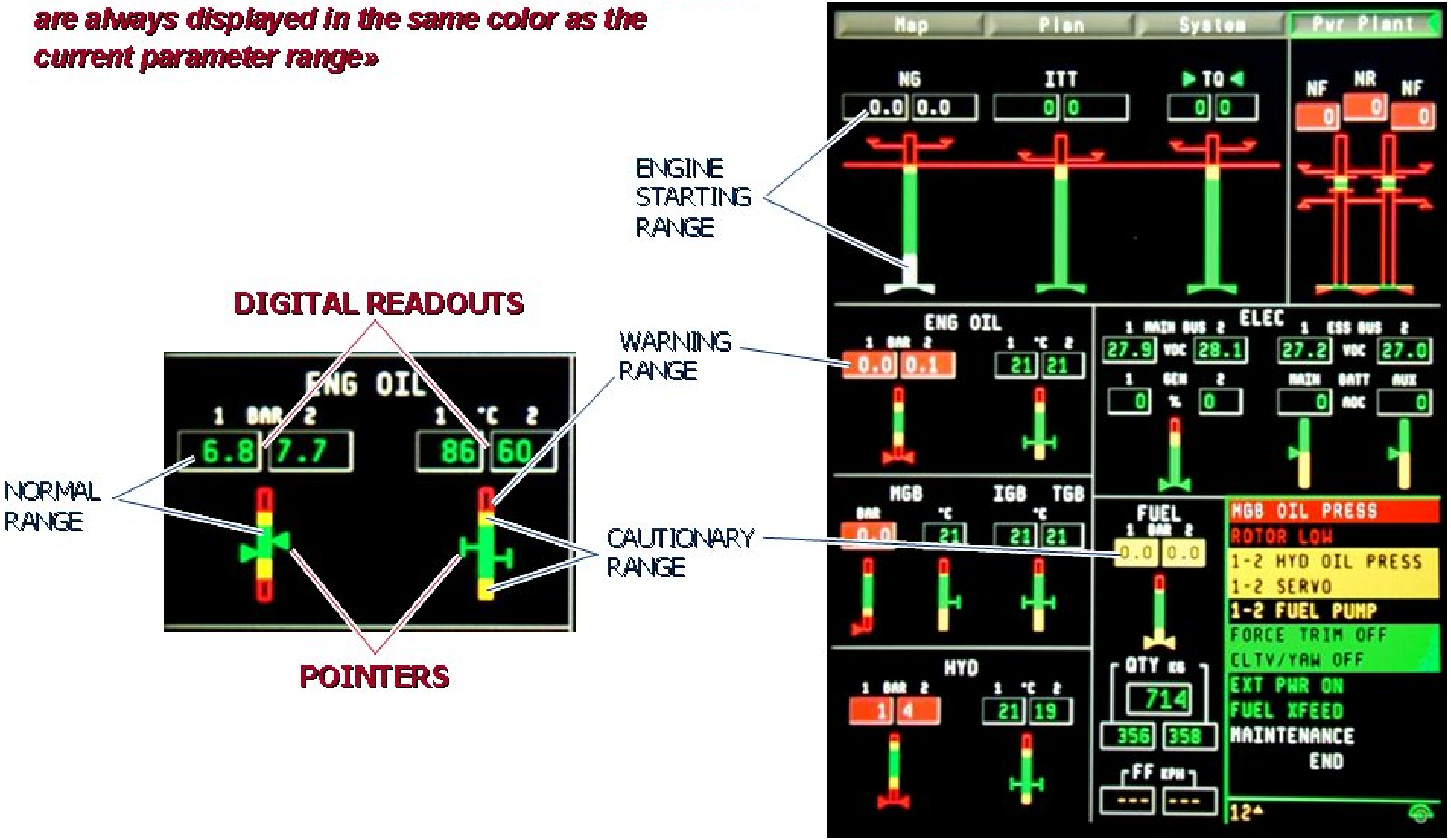
Colors used in the MFD and PFD displays follow the rules listed in the table.

Normally many of the graphic and digital displays change color when limits are reached or exceeded (usually from white or green to amber or red).

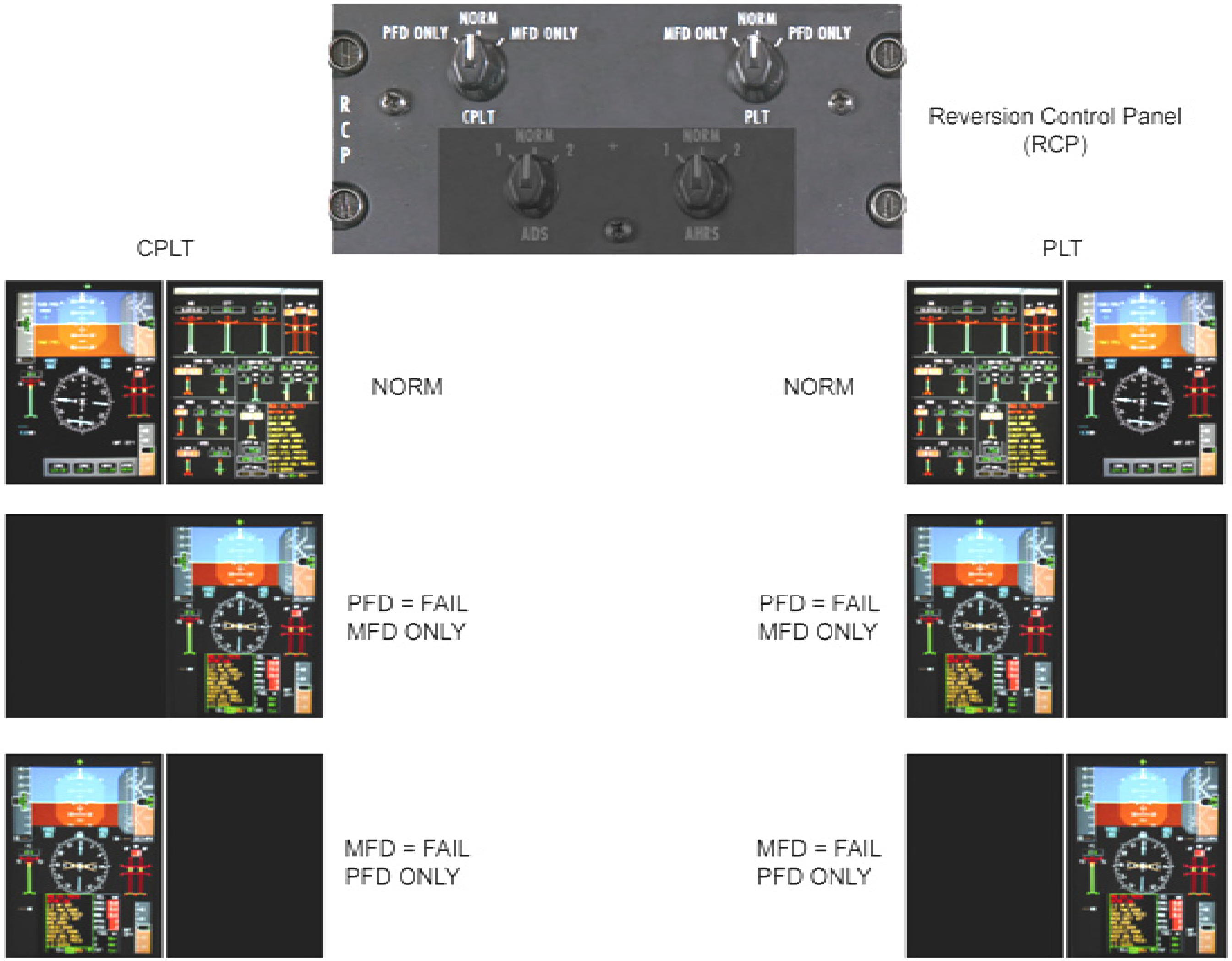
NORMAL VIDEO	REVERSE VIDEO	
RED	RED	Warning conditions & messages Flight envelope and system limits Invalid primary instruments
AMBER	AMBER	Caution conditions & messages Invalid data Miscompare annunciators
GREEN	GREEN	Normal operating conditions & data Advisory messages Engaged (Active) FD modes ON condition (synoptic displays)
CYAN	CYAN	Crew selected reference data and cues Primary Navaid data and cues
MAGENTA	MAGENTA	Crew selected reference data and cues when active with Flight Director Primary Navaid data and cues when active with Flight Director
WHITE		Labels, scales and reference symbols Status messages & data Armed FD modes OFF condition (synoptic displays)



«Pointers and digital readouts of system parameters are always displayed in the same color as the current parameter range»



POINTER / READOUT COLOR USAGE



DU – REVERSION LOGIC

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CENTRAL WARNING SYSTEM – GENERAL

The Central Warning System (CWS) provides system alerts to the aircrew when unsatisfactory aircraft conditions occur.

The system alerts are provided by the Monitor Warning Function (MWF) software running in each Modular Avionic Unit (MAU) that monitors continuously the aircraft systems.

The CWS provides the crew with visual indications —including Crew Alerting System (CAS) messages displayed in the CAS window of the Display Units— and aural warning messages.

CWS – MAIN COMPONENTS

The CWS main components are:

- two Monitor Warning Function (MWF) software, part of MAU 1 and MAU 2
- two Master Warning Lights (MWL)
- two Master Caution Lights (MCL)
- two CAS RST pushbuttons
- four Input/Output modules, part of MAU 1 and MAU 2
- one Aural Warning Generator (AWG), part of MAU 2

MONITOR WARNING FUNCTION (MWF)

The MWF is software running in the NIC/PROC module of each MAU that monitors continuously the aircraft systems and determines what alerts and advisories must be triggered and provided to the pilots.

MASTER WARNING LIGHTS (MWL)

Two red Master Warning Lights are installed in front of the PLT and CPLT seats. When a warning occurs, the MWLs start to flash until any pilot resets by either pressing his MWL or his CAS RST pushbutton.

MASTER CAUTION LIGHTS (MCL)

Two amber Master Caution Lights are installed in front of the PLT and CPLT seats. When a caution occurs, the MCLs start to flash until any pilot resets by either pressing his MCL or his CAS RST pushbutton.

CAS RST PUSHBUTTON

A CAS RST (CAS Reset) pushbutton is provided on each Collective Lever to reset both MWLs and MCLs

MAU INPUT/OUTPUT MODULES

The Control Input/Output (CIO) module and the Custom Input/Output (CSIO) module of both MAUs receive the input signals from all aircraft system sensors and drive the Master Warning Lights and the Master Caution Lights.

AURAL WARNING GENERATOR (AWG)

The Aural Warning Generator (AWG) supplies tones and aural messages (tones and voice) to alert pilots about possible dangerous conditions.

The CIO module of MAU 2 contains the AWG circuitry and supplies the aural warning output to the audio panels.

The AWG is controlled by the MWF software.

CWS – PRINCIPLE OF OPERATION

Each MWF generates a list of active messages (CAS List 1 in MAU 1 and CAS List 2 in MAU 2). In normal conditions, the CAS window in each MFD (or on a DU in Composite format) shows the messages generated by MAU 2, i.e. CAS List 2. Messages are grouped with the following priorities and formats and are identified by their color:

- Priority 1 — WARNING
- Priority 2 — CAUTION
- Priority 3 — ADVISORY
- Priority 4 — STATUS

The CAS window has 12 lines, 18 characters long.

The end of the message stack is indicated by the END line in white normal video.

Warning messages are shown at the top of the list (top priority) and cannot be scrolled. When a new warning message is activated, it is displayed in red reverse video (white text on red background) until it is acknowledged. After acknowledgement it is displayed in red normal video.

When a new caution message is activated, it is displayed as amber reverse video (black text on amber background) until it is acknowledged. After acknowledgement it is displayed in amber normal video.

When a new advisory message is activated it is displayed as green reverse video (black text on green background) for 5 seconds from when it is displayed in the CAS window, then it automatically changes into green normal video.

The status messages are only displayed when the helicopter is on the ground and are shown in white normal video.

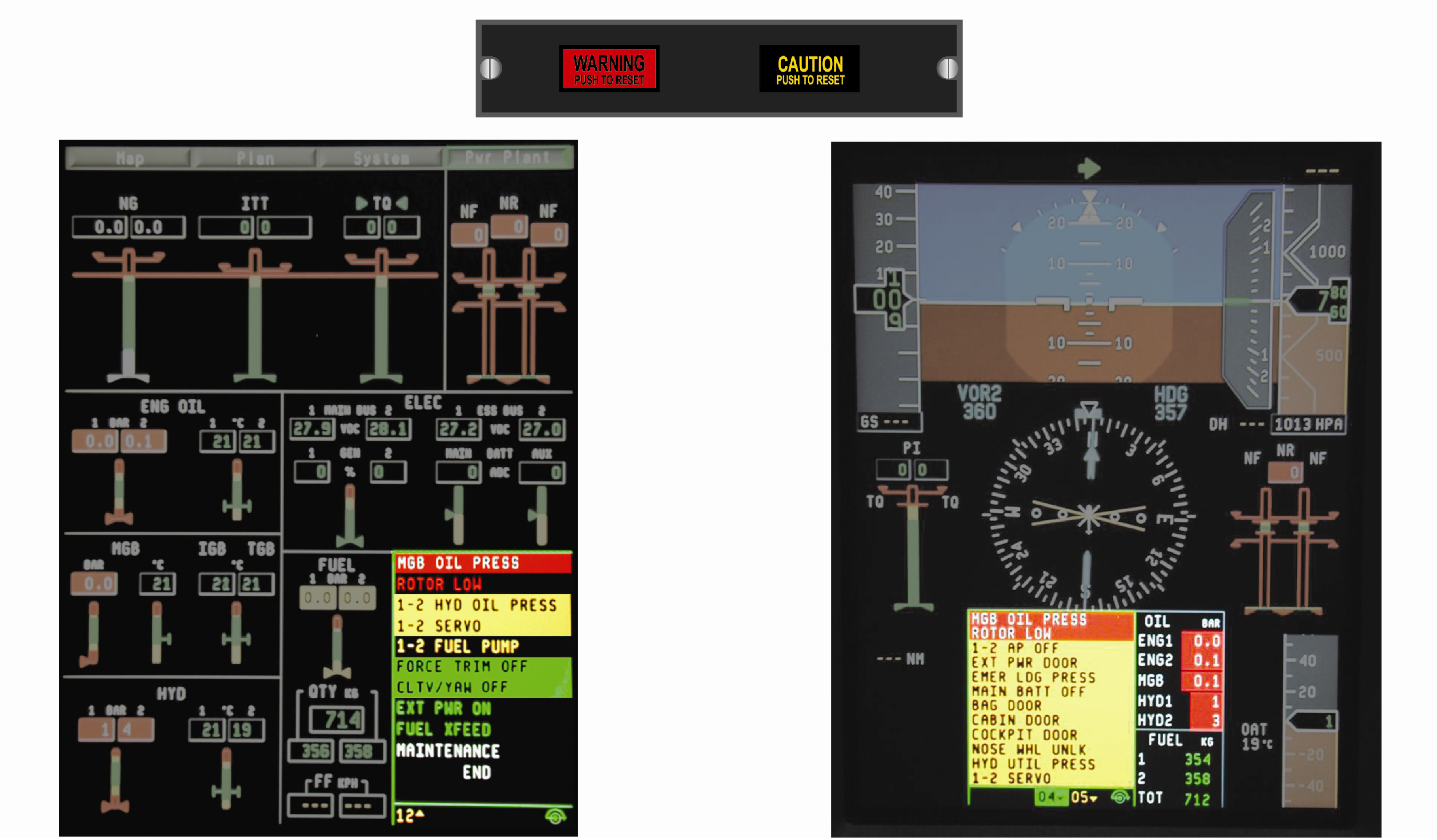
Within each group, messages are shown in the chronological sequence they have been triggered, with the newest message on the top.

Pilot and copilot can use the inner knob of their Cursor Control Device (CCD) to scroll up or down caution, advisory and status messages out of view, if acknowledged.

When any active message is out of view, annunciators are displayed below the CAS window showing how many messages are out of view and whether they are above or below the displayed stack.

When a new caution message actives while the list has been scrolled down, the list is brought back to the top automatically to show the new caution.

When a pilot acknowledges a message or scrolls the list, he affects both stations.



CENTRAL WARNING SYSTEM – GENERAL

CENTRAL WARNING SYSTEM - CONTROLS AND INDICATORS

1. CAS RST pushbutton

PRESSED .. when pressed the pilot (or copilot) acknowledges the warning/caution. The MWL and/or MCL extinguish and the aural message stops (if continuous)

2. MASTER WARNING LIGHTS

PRESSED .. when pressed the pilot (or copilot) acknowledges the warning; the lights stop flashing and the aural message removed (if applicable)

3. MASTER CAUTION LIGHT

PRESSED .. when pressed the pilot (or copilot) acknowledges the caution; the lights stop flashing

4. AWG switch on MISC panel

NORM ……. the tone and the 150 FEET aural message is normally heard when the aircraft is descending below 150 feet

REGRADE .. the tone and the 150 FEET aural message is suppressed

5. AWG switch on TEST panel

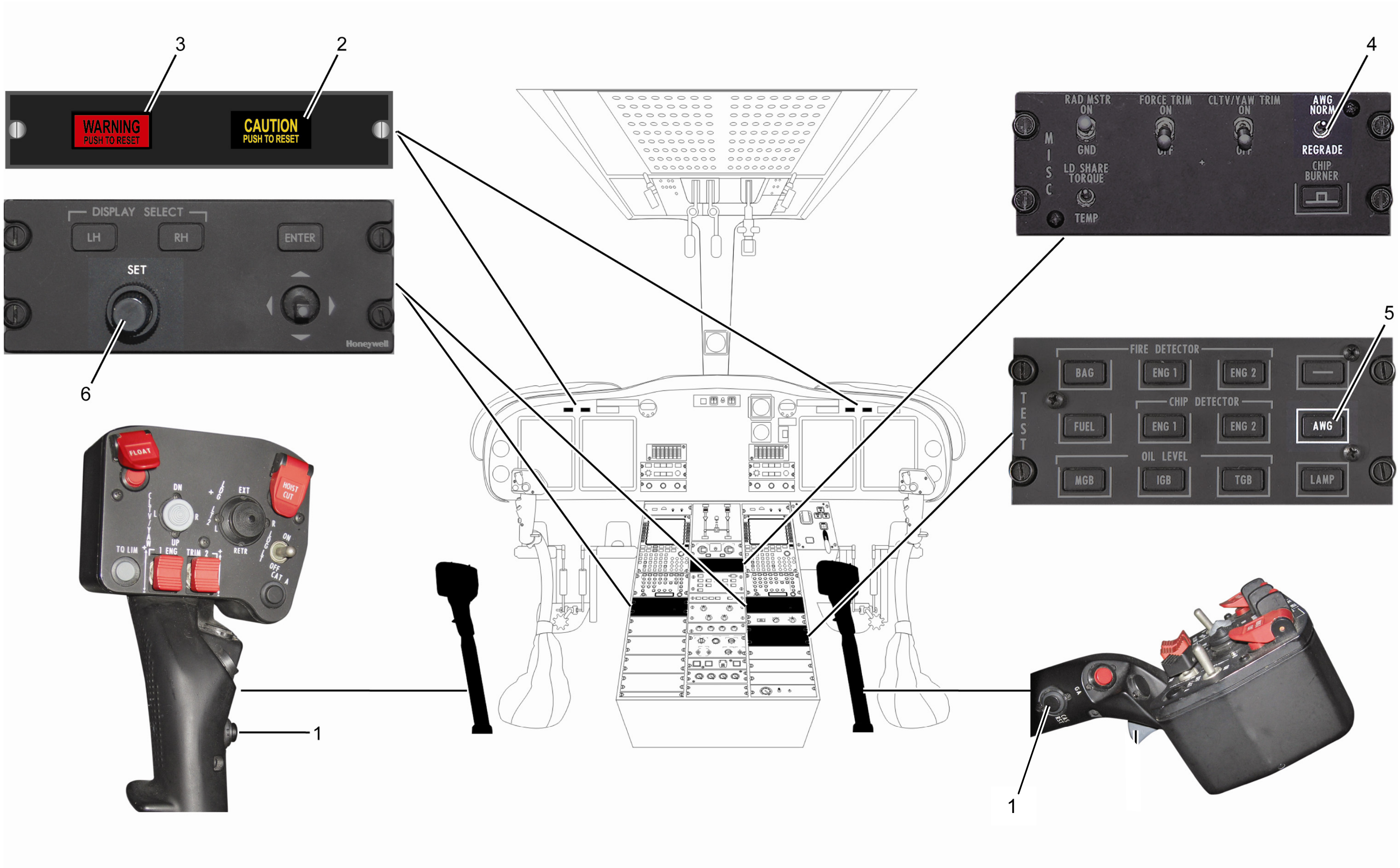
Short test ... a single short push activates the AURAL SYSTEM TEST message for one cycle

Full test …... pushing and holding the AWG TEST button for 6 seconds activates all the aural messages and tones in their priority order. Each message is generated once in TEST. This test can be done ON THE GROUND ONLY

6. SET knob

CW ………... rotated clock wise (CW) scrolls up the caution, advisory and status messages out of view

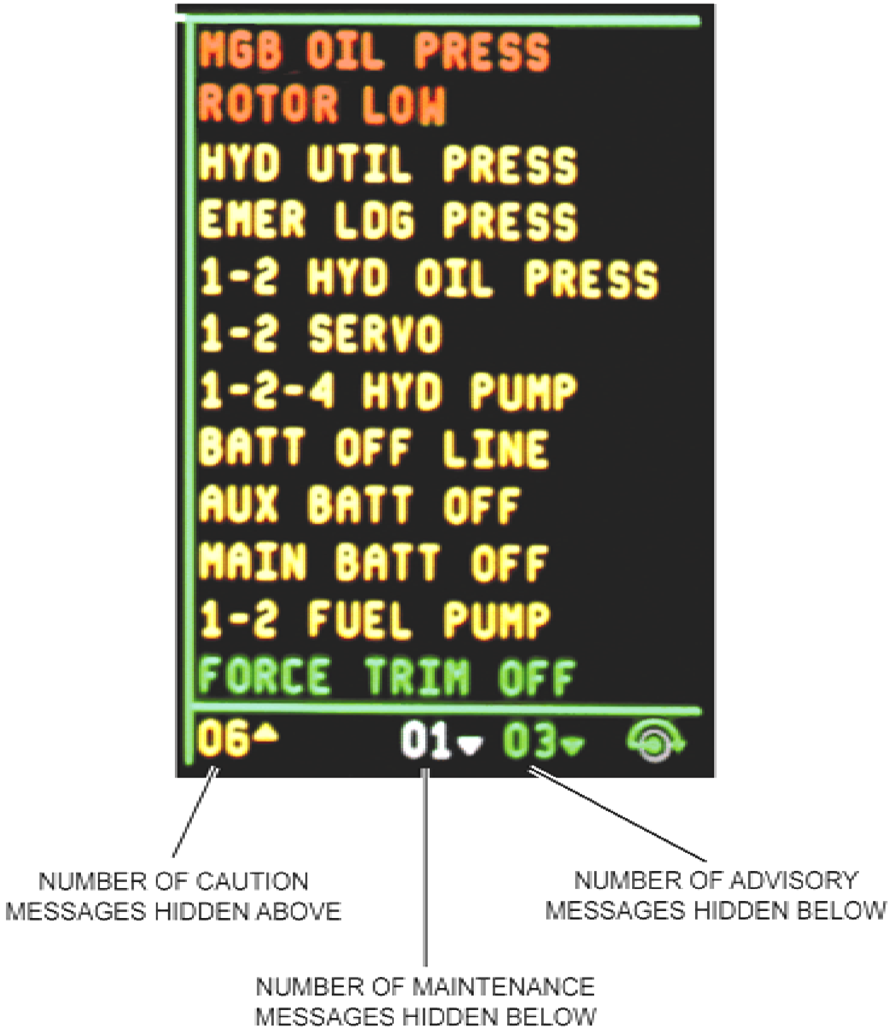
CCW ……... rotated counter clock wise (CCW) scrolls down the caution, advisory and status messages out of view



CWS – CONTROLS AND INDICATORS

PRIORITY 1	**WARNING MESSAGES**	INITIAL DISPLAY WARNING	AFTER ACKNOWLEDGEMENT WARNING
		Note 1. Warning messages are always in view Note 2. All warning messages trigger an aural warning	
PRIORITY 2	**CAUTION MESSAGES**	INITIAL DISPLAY CAUTION	AFTER ACKNOWLEDGEMENT CAUTION
PRIORITY 3	**ADVISORY MESSAGES**	INITIAL DISPLAY ADVISORY	AFTER 5 SECONDS ADVISORY
PRIORITY 4	**STATUS MESSAGES**	STATUS	Note. On ground only
		END	Note. Not considered a message

CWS – PRIORITY AND FORMAT OF MESSAGES



CURSOR CONTROL DEVICE (CCD)

CAS WINDOW AND SCROLL CONTROL

PRIORITY	AURAL MESSAGE	NR OF CYCLES	
1	**ROTOR LOW – ROTOR LOW**	Continuous	**ROTOR LOW**
2	**ENGINE 1 (2) OUT – ENGINE 1 (2) OUT**	1	**1 (2) ENGINE OUT**
3	**ENGINE 1 (2) FIRE – ENGINE 1 (2) FIRE**	Continuous	**1 (2) ENGINE FIRE**
4	**ROTOR HIGH – ROTOR HIGH**	1	**ROTOR HIGH**
5	**ENGINE 1 (2) IDLE – ENGINE 1 (2) IDLE**	Continuous	**1 (2) ENGINE IDLE**
6	**WARNING – WARNING**	1	Any other warning message
7	**AUTOPILOT – AUTOPILOT**	1	**1 (2) AP OFF** **1 (2) AP FAIL**
8	**AIRSPEED – AIRSPEED**	1	VNE Exceeded
9	**Flight Director Mode Change Chime**	1	Change of FD mode status
10	**Flight Director Reference Change Chime**	1	FD reference change for ALT or RHT modes
11	**ALTITUDE – ALTITUDE**	1	Departure from selected altitude
12	**LANDING GEAR**	1	**LANDING GEAR**
13	**150 FEET**	1	Descending below 150 ft AGL

Note: Additional aural messages are provided in case of optional installations such as TCAS or EGPWS

AURAL MESSAGES AND TONES

CAS CAUTION MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
AWG FAIL	Aural warning system failure. Loss of aural warning	AURAL WARNING SYSTEM FAILURE	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES

PFD MESSAGES

nCASMSCP on PFD display	MAU 1 & MAU 2 CAS WARNING message list has 'n' discrepancies Pilot press ENTER key on CCD while the symbol is displayed adjacent the caption to toggle between CAS LIST 1 and 2. Confirm which message(s) are giving the MISCOMPARE	CAS WARNING MESSAGE LIST DISCREPANCY	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES
nCASMSCP on PFD display	MAU 1 & MAU 2 CAS CAUTION message list has 'n' discrepancies Pilot press ENTER key on CCD while the symbol is displayed adjacent the caption to toggle between CAS LIST 1 and 2. Confirm which message(s) are giving the MISCOMPARE	CAS CAUTION MESSAGE LIST DISCREPANCY	

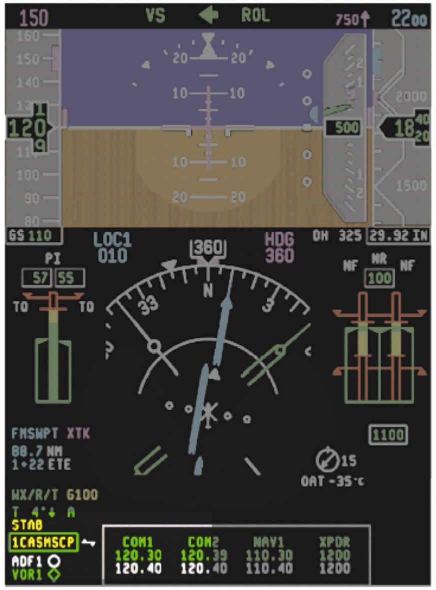
CAS MISCOMPARE

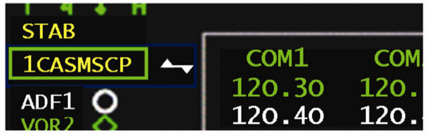
The CAS miscompare monitor verifies that both CAS Lists are the same.

The function calculates a checksum for the entire list of active CAS messages. Additionally, a second checksum is calculated for the warning messages.

A miscompare annunciation (CASMSCP) is displayed in the bottom left corner of the PFD if the active list checksum fails to compare for 7 seconds or if the warning message check-sum fails to compare for 3 seconds .

While the CASMSCP annunciation is highlighted by the cursor, the pilot can toggle the CAS message lists by pressing the "ENTER" pushbutton on the CCD. This allows the pilot to ensure that he has read all messages to resolve the discrepancy.





INDEPENDENT INSTRUMENTS

The following independent instruments are installed in the AW139 cockpit:

- Two Chronometers (Clocks)

CHRONOMETER (CLOCK)

Two independent digital chronometers are installed on the instrument panel, one in front of each pilot.

The chronometer provides the following functions:

- Local Time (LT) clock, in 12-hour format
- Universal Time Coordinate (UTC) clock, in 24-hour format
- Flight Time (FLT) counter, up to 99 hours, 59 minutes and 59 seconds
- Stop Watch (SW) counter, 99 hours, 59 minutes and 59 seconds
- Down Counter (DC), which counts down from a maximum of 99 hours, 59 minutes and 59 seconds

The chronometer is provided with three pushbutton switches and a liquid crystal display (LCD) with six numerals and annunciators to indicate the display mode.

The operating power for the pilot chronometer is fed from the ESS 2 bus via the CLOCK PLT circuit breaker.

The operating power for the copilot chronometer is fed from the MAIN 1 bus via the CLOCK CPLT circuit breaker.

An internally mounted AAA-size alkaline battery in each chronometer keeps the watch alive when the aircraft power is removed.

OPERATION

The clock is kept up-to-date also when the helicopter is not powered by the internal alkaline battery, but the display only operates when power is supplied by the relevant bus bar.

When activated, counters (FLT, SW, DC) keep counting even if a different display mode is selected.

The Flight Time counter is controlled by the WOW microswitch: it automatically starts counting up at lift off and stops counting at touchdown.

Flight Time is continuously summed up until it is manually reset by holding the ST/SP button while in FLT mode.

CHRONOMETER – CONTROLS AND INDICATORS

1. RST or SET pushbutton

Pressed in LT mode	Enters Local Time clock setting mode
Pressed in UTC mode	Enters Universal Time Coordinate clock setting mode
Pressed in FLIGHT mode	No effect
Pressed in SW mode	Resets the stopwatch
Pressed in DC mode	1st press: Resets the Down Counter 2nd press: Enters Down Counter setting mode

2. MODE pushbutton

Pressed	Selects the display mode toggling between LT, UTC, FLIGHT, SW, DC

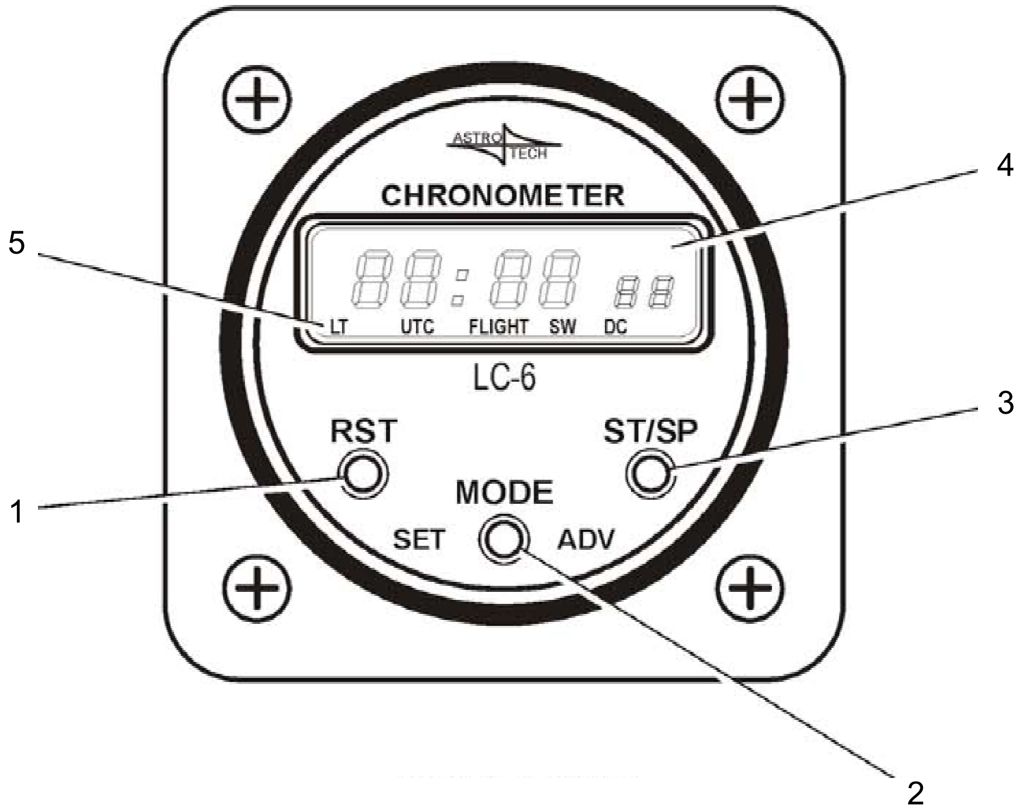
3. ST/SP or ADV pushbutton

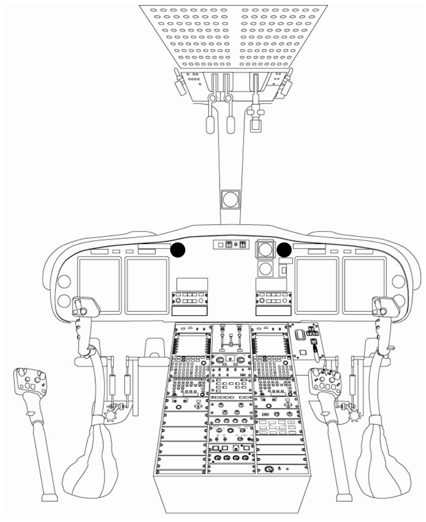
Pressed in LT or UTC clock setting mode	Advances the hour or the minute or resets the seconds
Pressed in FLIGHT mode	Resets the Flight Time counter
Pressed in SW or DC mode	Starts or stops the counter

4. LCD display

5. Annunciators

   Displayed one at a time, show the current mode of the chronometer





CHRONOMETER – CONTROLS AND INDICATORS

SETTING THE CHRONOMETER

LT OR UTC CLOCK SETTING

When in LT (or UTC) mode, press the SET pushbutton: the hours digits are displayed, only.

Press the ADV pushbutton to advance the hours. If in LT mode range is 00: to 12: then back to 00:. If in UTC mode range is 00: to 24: then back to 00:.

Press the SET pushbutton: the minutes and seconds are displayed, only.

Press the ADV: the seconds are reset to 00 and held.

Press ADV pushbutton again to advance the minutes (00: to 59: and then back to 00:). Set the actual time plus 1 minute.

Press the SET pushbutton to return to LT (or UTC) mode: clock is stopped.

Start the clock by pressing the ST/SP pushbutton.

FLIGHT TIME COUNTER SETTING

Flight time counter operation is automatic: it sums up the helicopter airborne time up to 99:59:59.

When in FLT mode, press the ST/SP pushbutton to reset the counter.

STOP WATCH SETTING

When in FLT mode, press the ST/SP start or stop the stopwatch, or press the RST pushbutton to reset.

If the RST pushbutton is pressed while the stop watch is running, it reset to zero and then continues to count.

DOWN COUNTER SETTING

When in DC mode, press the SET pushbutton: the counter shows 0:00:00.

Press SET a second time to enter the DC setting mode: the hours digits are displayed, only.

Press the ADV pushbutton to advance the hours (00: to 99: then back to 00:).

Press the SET pushbutton: the minutes are displayed, only.

Press ADV pushbutton again to advance the minutes (00: to 59: and then back to 00:).

Press the SET pushbutton: the seconds are displayed, only.

Press ADV pushbutton again to advance the seconds (00: to 59: and then back to 00:).

Press the SET pushbutton to return to DC mode: the selected time is displayed.

Start the down counter by pressing the ST/SP pushbutton.

When the down counter reaches 0:00:00, the counter begins counting up until stopped and the display flashes for 1 minute.

MULTI-PURPOSE FLIGHT DATA RECORDER (MPFDR)

The Multi-Purpose Flight Data Recorder (MPFDR) system is composed by:

- One Multi-Purpose Flight Data Recorder Unit
- One Control Unit
- One Cockpit Area Microphone
- One Accelerometer

The MPFDR system operates as a Flight Data Recorder (FDR) and Cockpit Voice Recorder (CVR); it automatically records selected aircraft parameters and 4 audio sources into a crashworthy, fireproof, deep-waterproof, solid state memory module.

The MPFDR system operating power is fed by the ESS 1 bus via the FDR circuit breaker.

The MPFDR system is also supplied by the BATT Bus, via the FDR/CVR circuit breaker located in the nose compartment, for shutdown operation.

MULTI-PURPOSE FLIGHT DATA RECORDER UNIT

The MPFDR Unit (or FDR/CVR) is a solid state airborne recorder and consists of a solid state memory module and an electronic interface. The solid state memory module is protected for crash survivability and is able to store a minimum of 25 hours of aircraft data (FDR) and 120 minutes of 4-channel audio (CVR).

When the memory is full, the FDR/CVR keeps recording replacing the oldest data with the newest ones.

The FDR/CVR Unit is installed in the tailboom forward area.

An Underwater Locator Beacon (ULB) is fitted to the Crash Survivable Memory Module as an aid to locate the module in the event of an accident over water.

The ULB is triggered by an internal water sensor emits 37.5 kHz frequency; it is provided with an autonomous battery that permits 30 days of operation minimum, from sea level down to 20,000 ft of depth.

CONTROL UNIT

The Control Unit (also defined as Cockpit Control Unit) permits the ground crew to test the system and erase the recorded audio.

The Control Unit is installed in the LH rear avionic bay and contains the Cockpit Area Microphone Pre-amplifier, a Headphone jack socket, push-buttons and indicators for the self-test and voice erase facilities of the FDR/CVR.

Aside the Control Unit, a connector permits downloading of all recorded aircraft data and audio parameters to a suitably equipped laptop computer when the helicopter is on the ground and both engines are off.

ACCELEROMETER

The Accelerometer senses accelerations along the three major helicopter axes (lateral, longitudinal and vertical) and provides them to MAU 1.

MAU 1 transfers the accelerometer data along with other aircraft data to the FDR/CVR Unit via a digital bus, for recording.

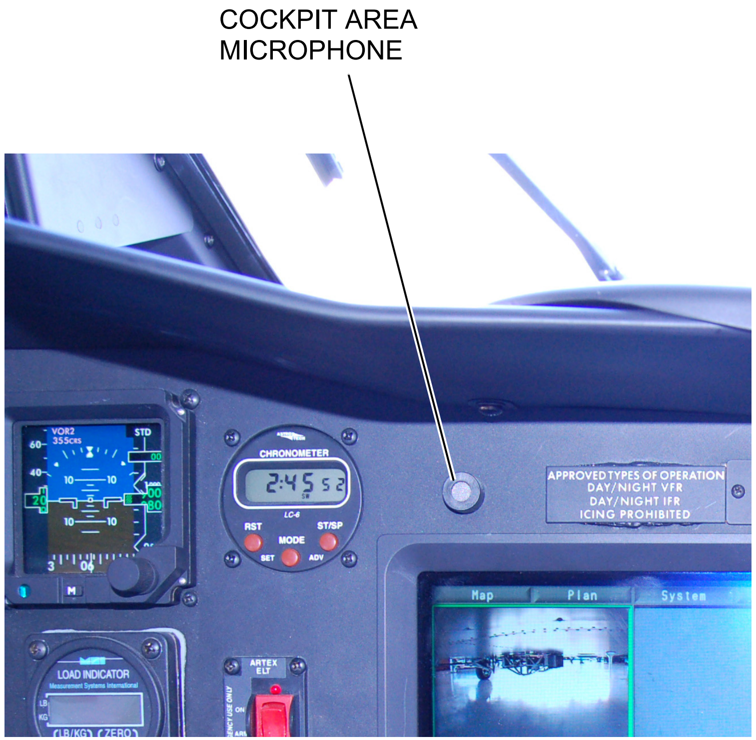
The Accelerometer is mounted inside the baggage compartment roof.

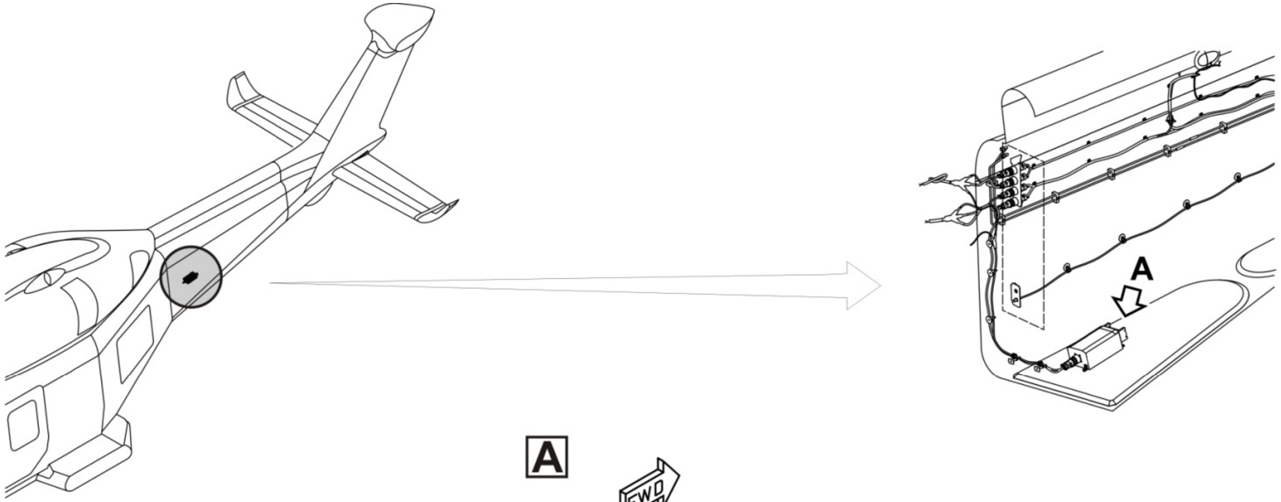
The Accelerometer sensing range is as follows:

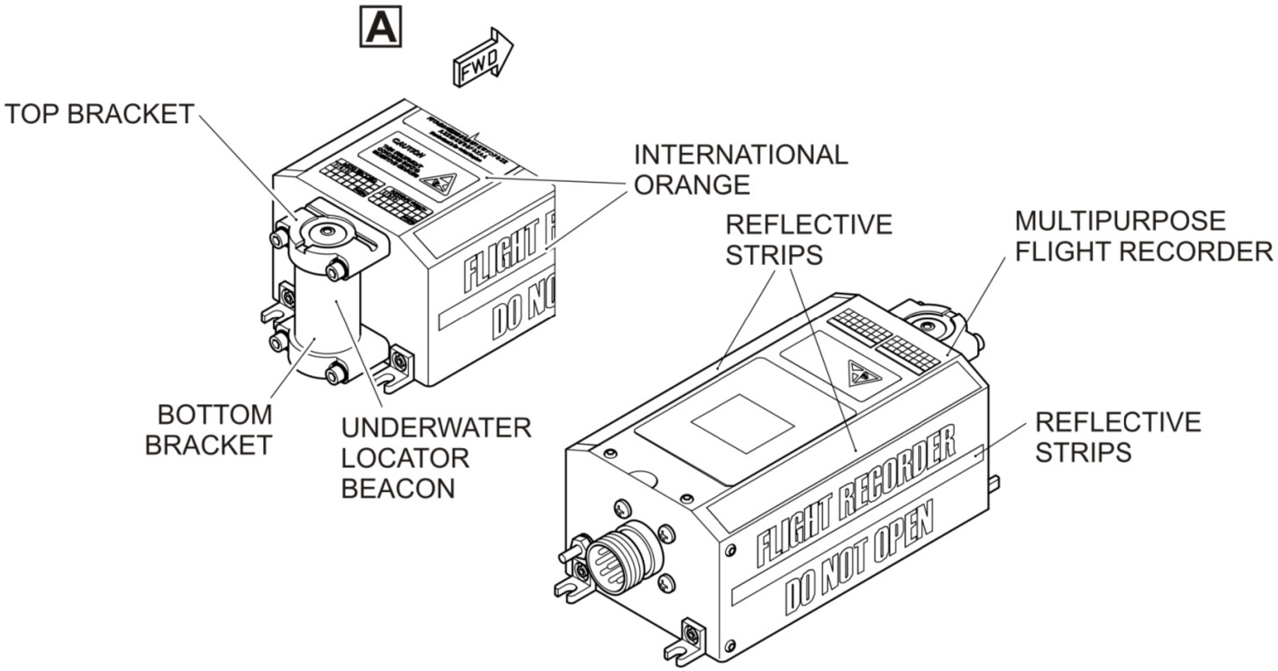
- Vertical acceleration: +6 G to –3 G
- Longitudinal acceleration: +1 G to –1 G
- Lateral acceleration: +1 G to –1 G

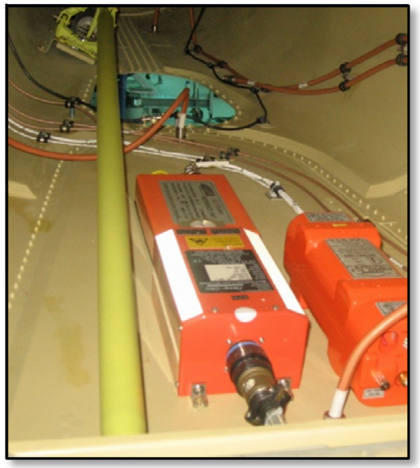
COCKPIT AREA MICROPHONE

The Cockpit Area Microphone is installed on the Instrument panel, above Pilot MFD (DU3), and provides one of the audio inputs to the FDR/CVR.

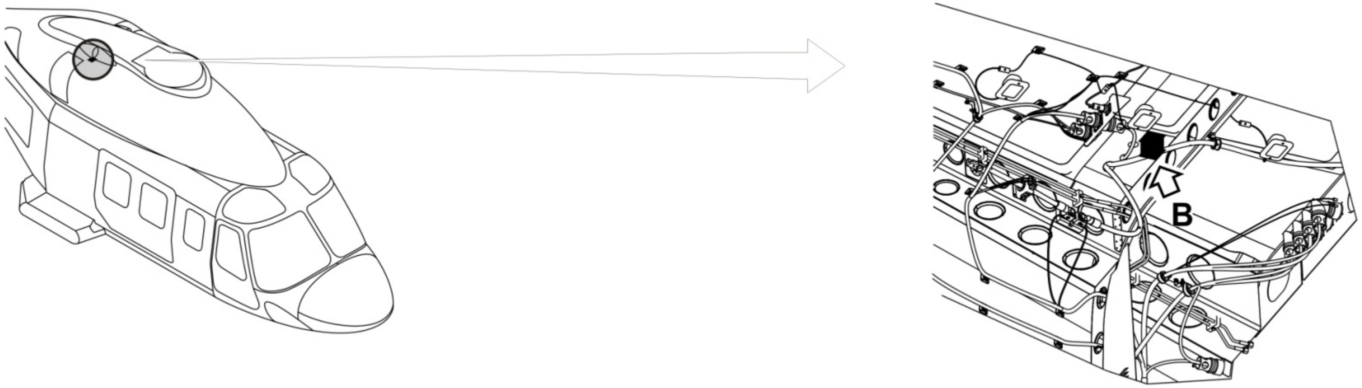


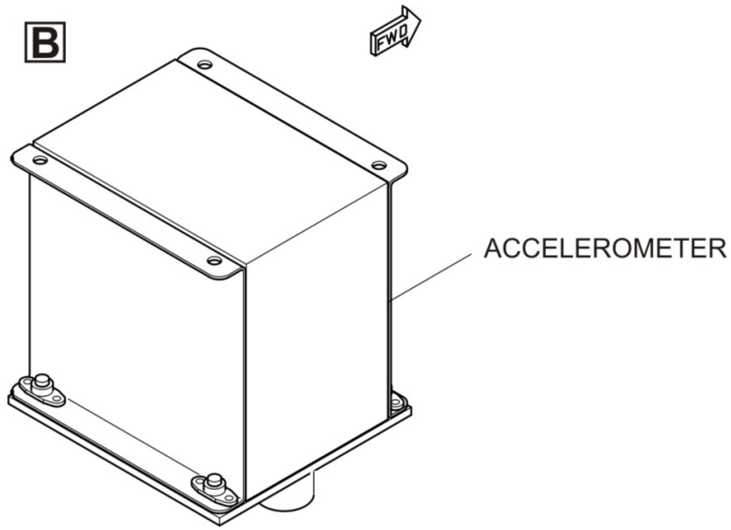




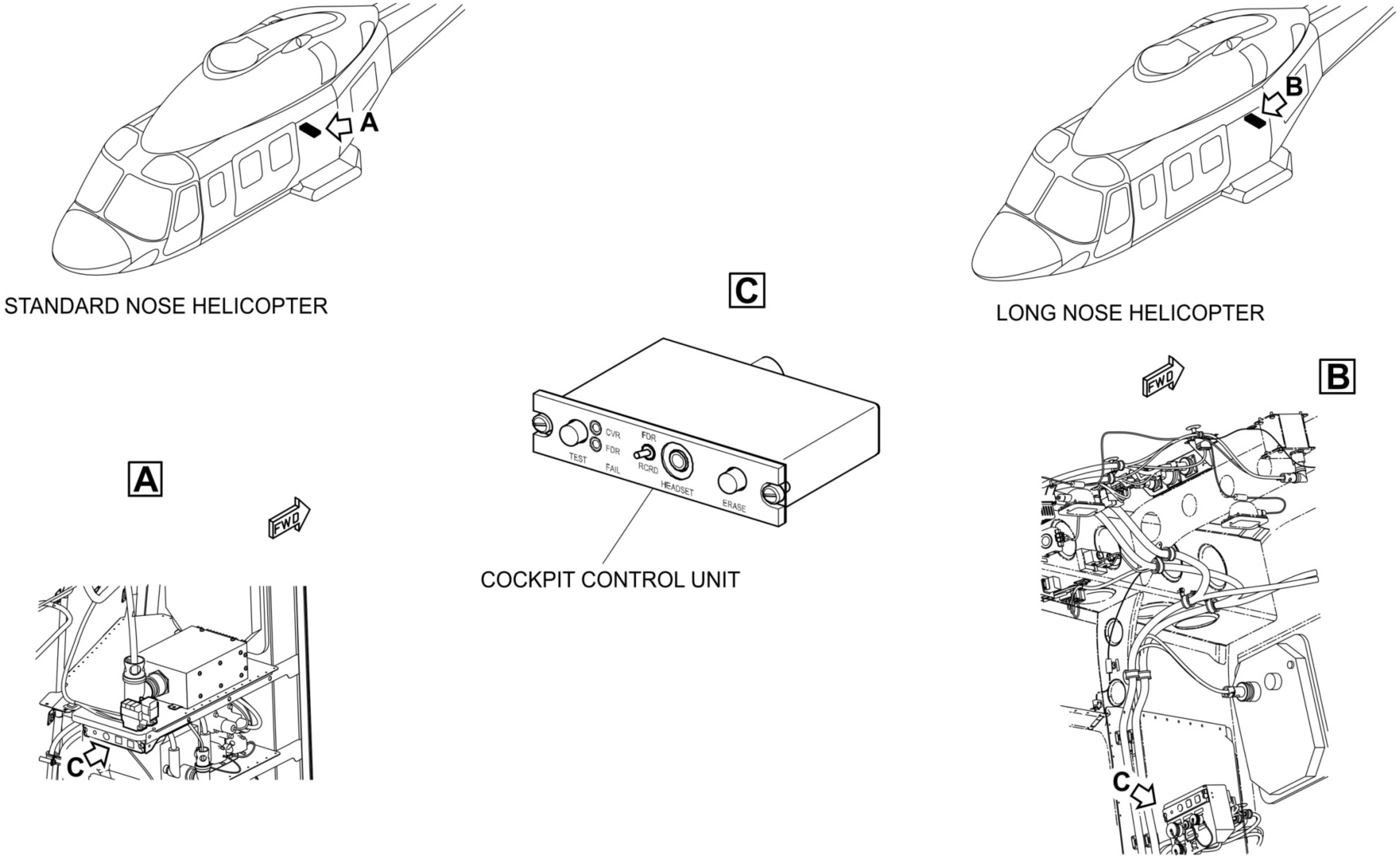


FDR/CVR – LOCATION OF MAJOR COMPONENTS





FDR/CVR – LOCATION OF MAJOR COMPONENTS



FDR/CVR – LOCATION OF MAJOR COMPONENTS

FDR/CVR OPERATION

The FDR/CVR starts and stops recording automatically.

The FDR enters the Recording Status when power is available to the FDR/CVR Unit from ESS 1 bus and either:

- At least one engine is not OFF, or
- Helicopter is airborne (weight off wheels)

The CVR starts recording upon power application to the FDR/CVR Unit from ESS 1 bus and stops within 10 minutes if the FDR non-recording condition persists.

When the CVR stops recording the CVR FAIL caution message appears in the CAS window.

In a standard flight, the CVR starts recording as soon as the BATTERY MASTER switch is moved to ON and stops recording 10 minutes after the second engine is turned off after landing (considering that pre-start checks last less than 10 minutes).

In a standard flight, the FDR starts recording as soon as the first ENG MODE switch is moved to IDLE and stops recording as soon as the second engine is off after landing.

In case of dual engine failure during flight, FDR stops recording at touchdown.

FDR DATA

The aircraft flight parameters are provided to the FDR Unit by MAU 1 via a dedicated ARINC 573 digital bus, and are listed in the following table:

NO.	RECORDED PARAMETER
1	Time
2a	Pressure Altitude (ALT)
2b	Vertical Speed (VS)
3	Indicated Airspeed (IAS)
4a	Magnetic Heading
4b	Magnetic Variation
5a	Normal Acceleration
5b	Tri-axial Accelerometer Valid
5c	Normal Acceleration (AHRS)
6	Pitch Attitude
7	Roll Attitude
8a	Radio PTT key (pilot)
8b	Radio PTT key (copilot)
9a	NF1 & NF2
9b	Torque 1 & Torque 2
9c	ITT1 & ITT2
9d	NG1 & NG2
9e	EEC1, EEC2 Mode
9f	Power Index (displayed parameter)

NO.	RECORDED PARAMETER
9g	Power Index (displayed equivalent value)
9h	Engine 1&2 Oil Temp
10a	Rotor RPM (NR)
10b	Rotor Brake
11a	Collective Pitch
11b	Longitudinal Cyclic
11c	Lateral Cyclic
11d	Tail Rotor Pedal
11e	Hydraulic 1 Selection
11f	Hydraulic 2 Selection
12a	Hydraulic 1 Low Pressure
12b	Hydraulic 2 Low Pressure
12c	Hydraulic 1/2 Oil Temp
13	T1, OAT
14	AFCS Mode and Engagement
15a	SAS 1 On
15b	SAS 2 On
16	Main Gearbox Oil Pressure
17	Main Gearbox Oil Temperature
17a	IGB Oil Temperature
17b	TGB Oil Temperature
18	Yaw Rate
20a	Longitudinal Acceleration

NO.	RECORDED PARAMETER
20b	Longitudinal Acceleration (AHRS)
21a	Lateral Acceleration
21b	Lateral Acceleration (AHRS)
22a	Radio Altitude (pilot PFD)
22b	Radio Altitude (copilot PFD)
22c	RALT Validity
23	Glide Slope 1 & 2 Deviation
24	Localizer 1 & 2 Deviation
25	Marker Beacon 1 & 2
26a	MWL On
26b	MCL On
26c	Main Gearbox Low Pressure
26d	SAS1 & SAS2 Failure
26e	Each "Red" Warning
26f	Generator 1 & 2 Failure
26g	Inverter 1 & 2 Failure (if installed)
26h	EEC 1 & EEC 2 Failure
26i	Engine DSCWD1, DSCWD2, NCFUR1, NCFUR2, CFUR1
26j	ADC 1 & ADC 2 Valid
26k	AHRS 1 & AHRS 2 Valid
26l	NAV 1 & NAV 2 Valid
26m	DME Valid
26n	FMS Valid

NO.	RECORDED PARAMETER
26o	All "Amber" Cautions
26p	MAU1/2 failure on PFDs
26q	1/2 CASMSCP on PFDs
27	VOR/ILS 1 & 2 Frequency
28a	DME on Pilot PFD Distance
28b	DME on Copilot PFD Distance
29a	Latitude/Longitude
29b	Drift Angle
29c	Wind Speed
29d	Wind Direction
30a	WOW1
30b	WOW2
30c	Synthetic Ground
33a	Fuel Contents 1/2
33b	Fuel Flow 1/2
33c	Fuel Press 1/2
34	Altitude Rate
35	Ice Detection (if installed)
36	HUMS Data (if installed)
38a	Baro Set (Pilot)
38b	Baro Set (Copilot)
39a	Selected Altitude (Pilot)
39b	Selected Altitude (Copilot)

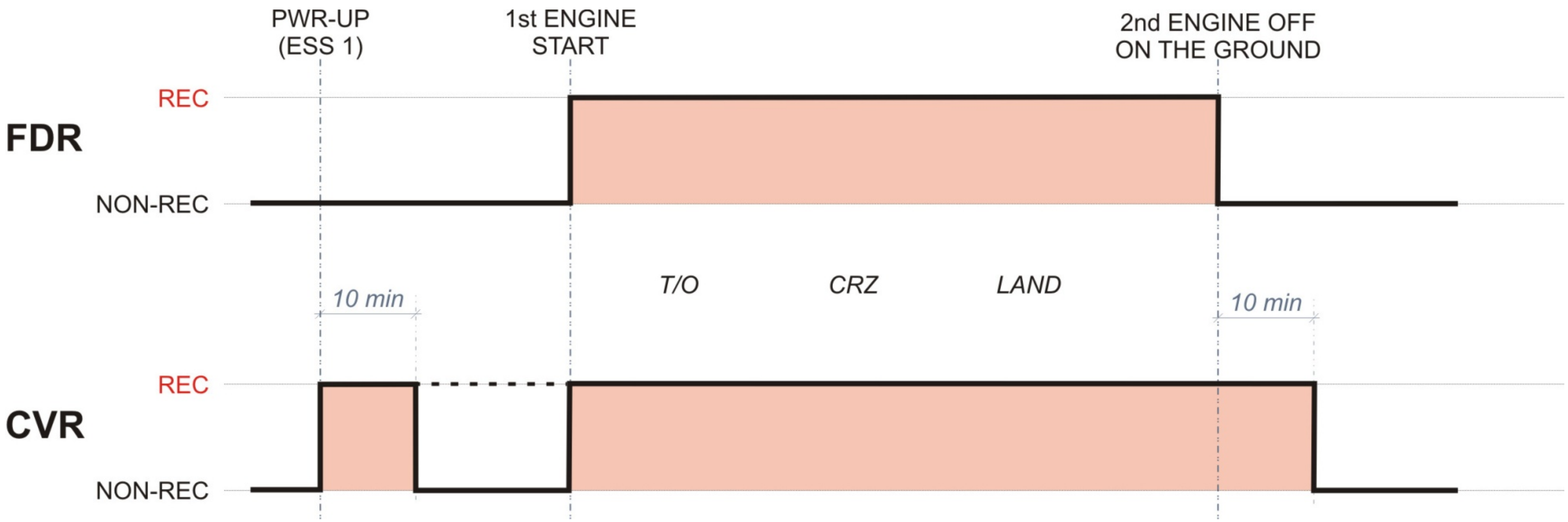
NO.	RECORDED PARAMETER
40a	Selected Speed (Pilot)
40b	Selected Speed (Copilot)
42a	Selected Vertical Speed (Pilot)
42b	Selected Vertical Speed (Copilot)
43	Selected Heading
44a	Selected Course (Pilot)
44b	Selected Course (Copilot)
44c	Selected flight path (All pilot selectable course of operation: VOR1/2, LOC1/2, FMS1/2)
44d	Selected flight path (All copilot selectable course of operation: VOR1/2, LOC1/2, FMS1/2)
45a	Selected Decision Height (Pilot)
45b	Selected Decision Height (Copilot)
46a	PFD Format (Pilot)
46b	PFD Format (Copilot)
47a	MFD Format (Pilot)
47b	MFD Format (Copilot)
47c	MFD Pilot Config (MAP, PWR PLANT, etc.)
47d	MFD Copilot Config (MAP, PWR PLANT, etc.)
48a	Loadmeter Main Battery
48b	Loadmeter Aux Battery
48c	Loadmeter Generator 1
48d	Loadmeter Generator 2
49	TCAS Alarms (if installed)

NO.	RECORDED PARAMETER
50	AWG Regrade
51	NVG mode (if installed)
52	Ground Speed
53	Day
54	Month
55	Year
56	AC Code
57	EGPWS Alarms (if installed)

CVR DATA

The four audio channels recorded by the FDR/CVR are:

- CH 1: Cabin ICS
- CH 2: Copilot Headset
- CH 3: Pilot Headset
- CH 4: Cockpit Microphone



FDR/CVR – OPERATION

FDR/CVR – CONTROLS AND INDICATORS

No control is provided to the crew for the FDR/CVR.

CAS CAUTION MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
FDR FAIL	Flight Data Recorder failed	FLIGHT DATA RECORDER FAILURE	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES
CVR FAIL	Cockpit Voice Recorder failed	COCKPIT VOICE RECORDER FAILURE	

CHAPTER
32
LANDING GEAR

SECTION 00 – GENERAL

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LANDING GEAR – GENERAL

The landing gear is a "fore and aft" retractable tricycle type. The landing gear:

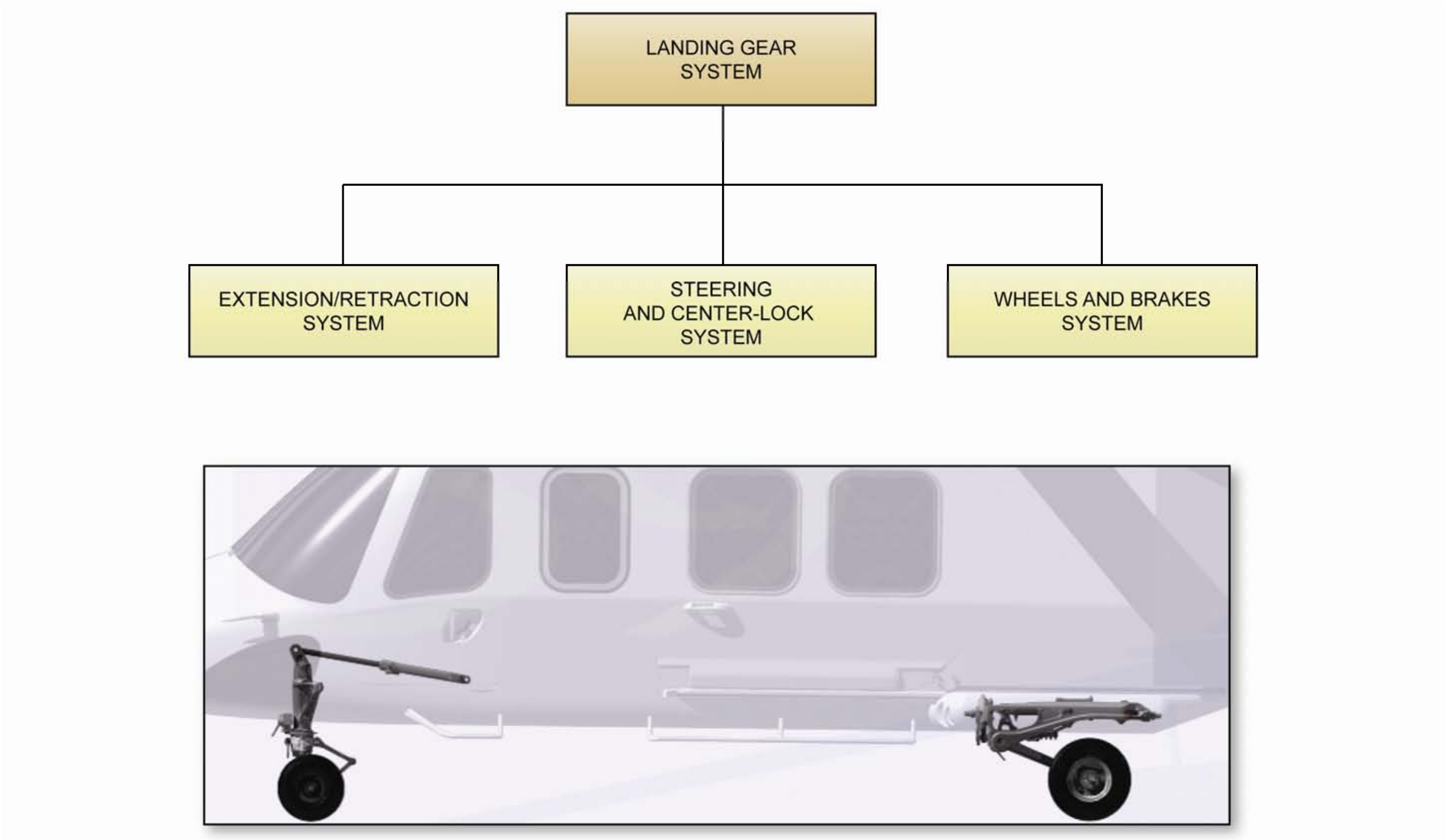
- supports the helicopter on the ground giving the correct ground clearance
- gives shock absorption and rebound control during landing, taxiing and manoeuvres on the ground
- decelerates and stops the helicopter after landing and during taxiing
- allows to park the helicopter

The landing gear includes:

- the Main Landing Gear (MLG) located on the left and on the right side of the fuselage under the sponsons
- the Nose Landing Gear (NLG) located under the cockpit
- the extension / retraction system of the MLG and NLG
- the braking system
- the automatic centering and lock system (or steering system) located on the NLG

The landing gear is operated by the pilot to perform:

- normal retraction and extension by means of the Landing Gear Control Lever (LGCL)
- emergency extension by means of the EMER DOWN pushbutton



LANDING GEAR – GENERAL LAYOUT

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MAIN LANDING GEAR – GENERAL

The Main Landing Gear (MLG) comprises the left and right landing gears located symmetrically on either side of the fuselage under the respective sponsons. Each landing gear is constituted by

- a retractable lever suspension mechanism with an oleo-pneumatic damping system
- and a single tubeless wheel

The wheel hub incorporates a safety relief plug to permit release of overpressure and two fusible plugs which release tyre pressure in case of overheating.

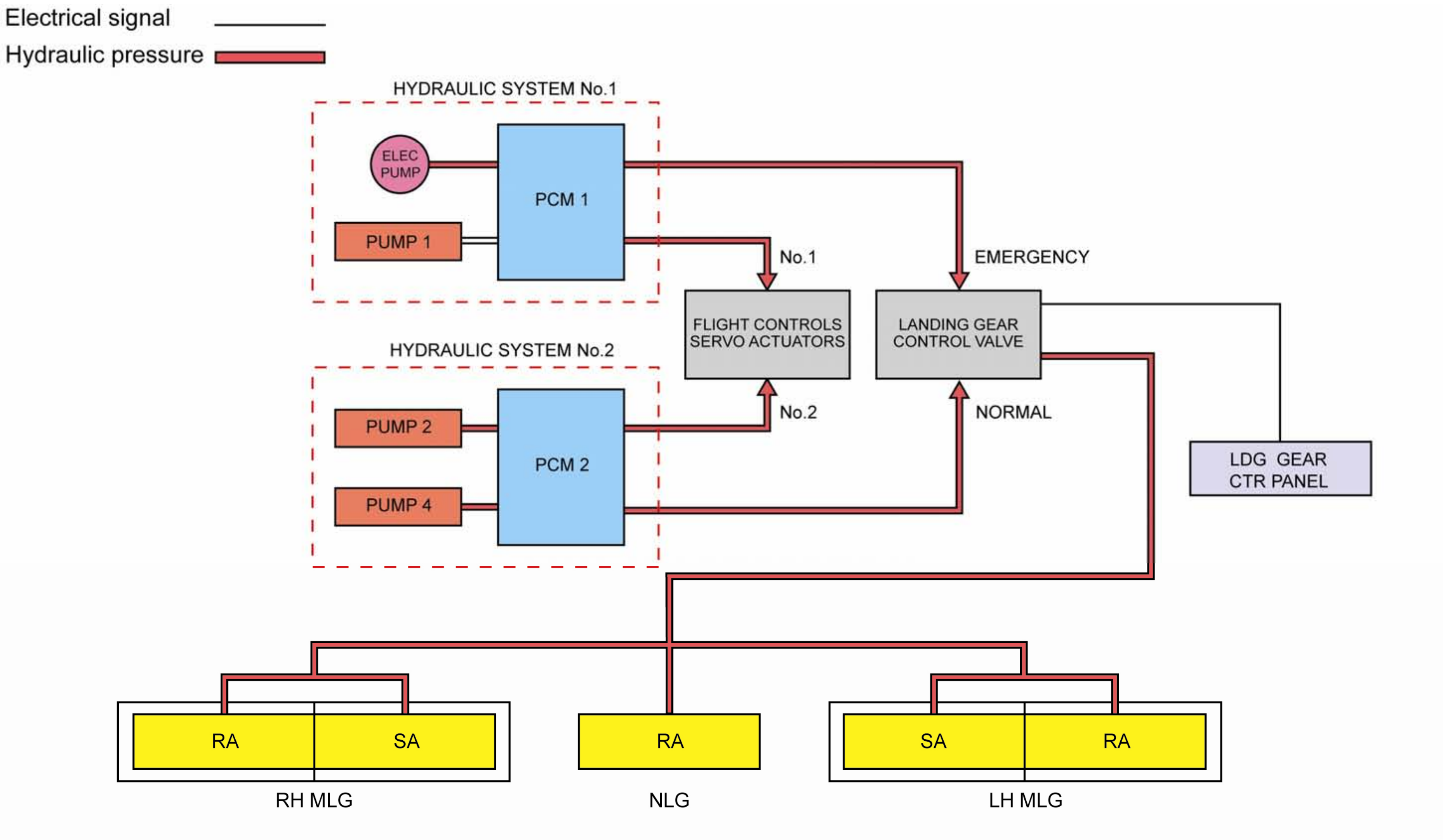
The main landing gear is maintained in UP position by hydraulic power and in DOWN position by mechanical locks inside the actuators.

Each main landing gear comprises:

- the extension/retraction (RA) actuator
- the shortening actuator (SA)
- the shock absorber
- the Weight-On-Wheel (WOW) microswitch

The extension/retraction (RA) actuator and the shortening actuator (SA) are part of the extension/retraction system described later.

The MLG can operate in three different ways: retraction, normal extension and emergency extension.



LANDING GEAR EXTENSION AND RETRACTION – BLOCK DIAGRAM

MAIN LANDING GEAR – MAIN COMPONENTS

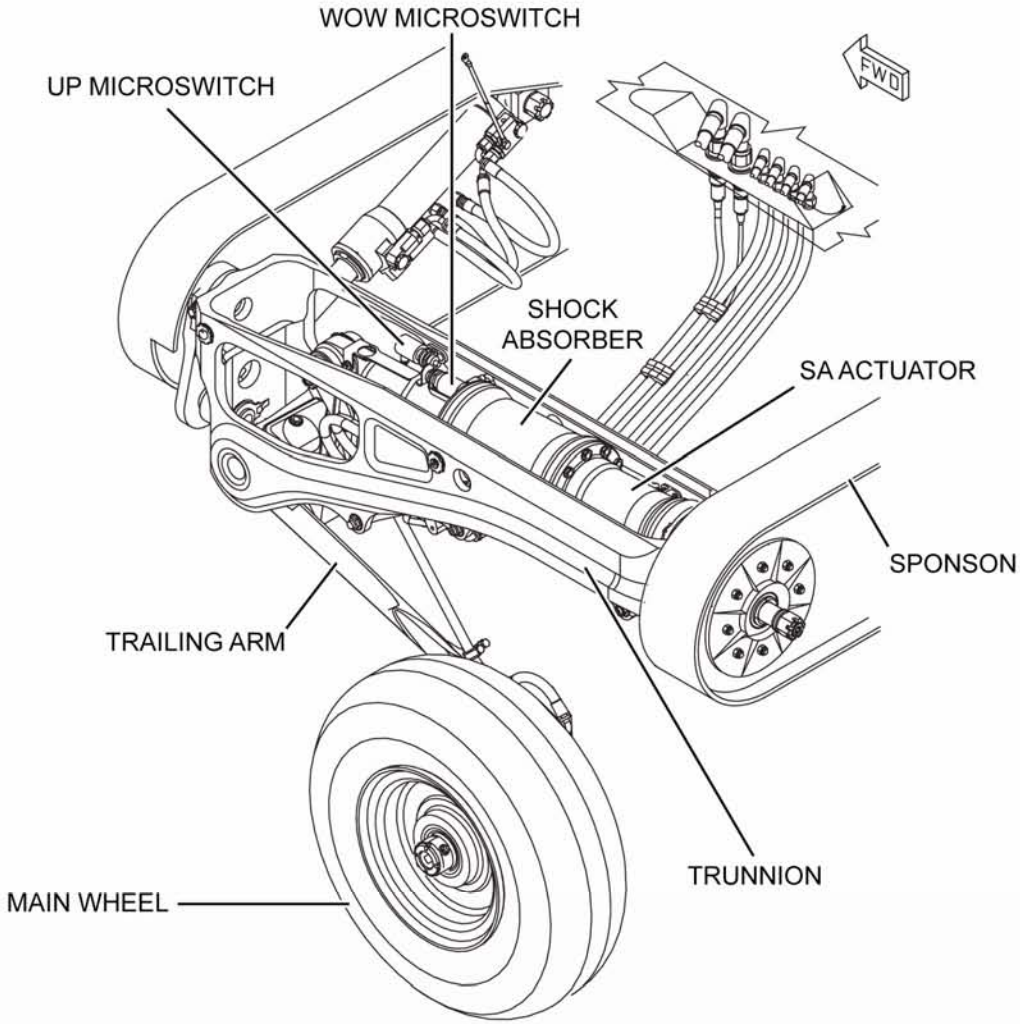
SHOCK ABSORBER

The shock absorber is a gas/oil damper unit composed by a piston rod sliding inside a cylinder. One end of the shock absorber is connected to the shortening actuator, the other end is connected to the trailing arm which is a structural element that maintains the MLG aligned longitudinally with the helicopter axis.

The piston rod includes a separator piston that prevents mixing of the gas with the oil and a variable orifice valve unit that controls the landing gear damping and rebound action.
The variable orifice valve unit is composed by a spring loaded piston which, under the oil pressure generated during landing, opens calibrated holes through which the oil moves at a controlled flow rate.

The shortening actuator extends for MLG retraction and retracts for MLG extension.

The shock absorber and the shortening actuator (SA) are joined together to build up the shock absorber assembly.



LH MLG

RH MLG

NOTE. LH and RH MLG are arranged simmetrically

MAIN LANDING GEAR COMPONENTS

MAIN LANDING GEAR SHOCK ABSORBER – PRINCIPLE OF OPERATION

In flight no load are applied to MLG (suppose the MLG retracted). The pressure of the nitrogen balances the pressure of the hydraulic fluid in the cylinder and maintains the separator piston in a neutral position.

The shock absorber starts to operate as soon as the helicopter touches the ground following two subsequent phases

- first stage damping
- second stage damping

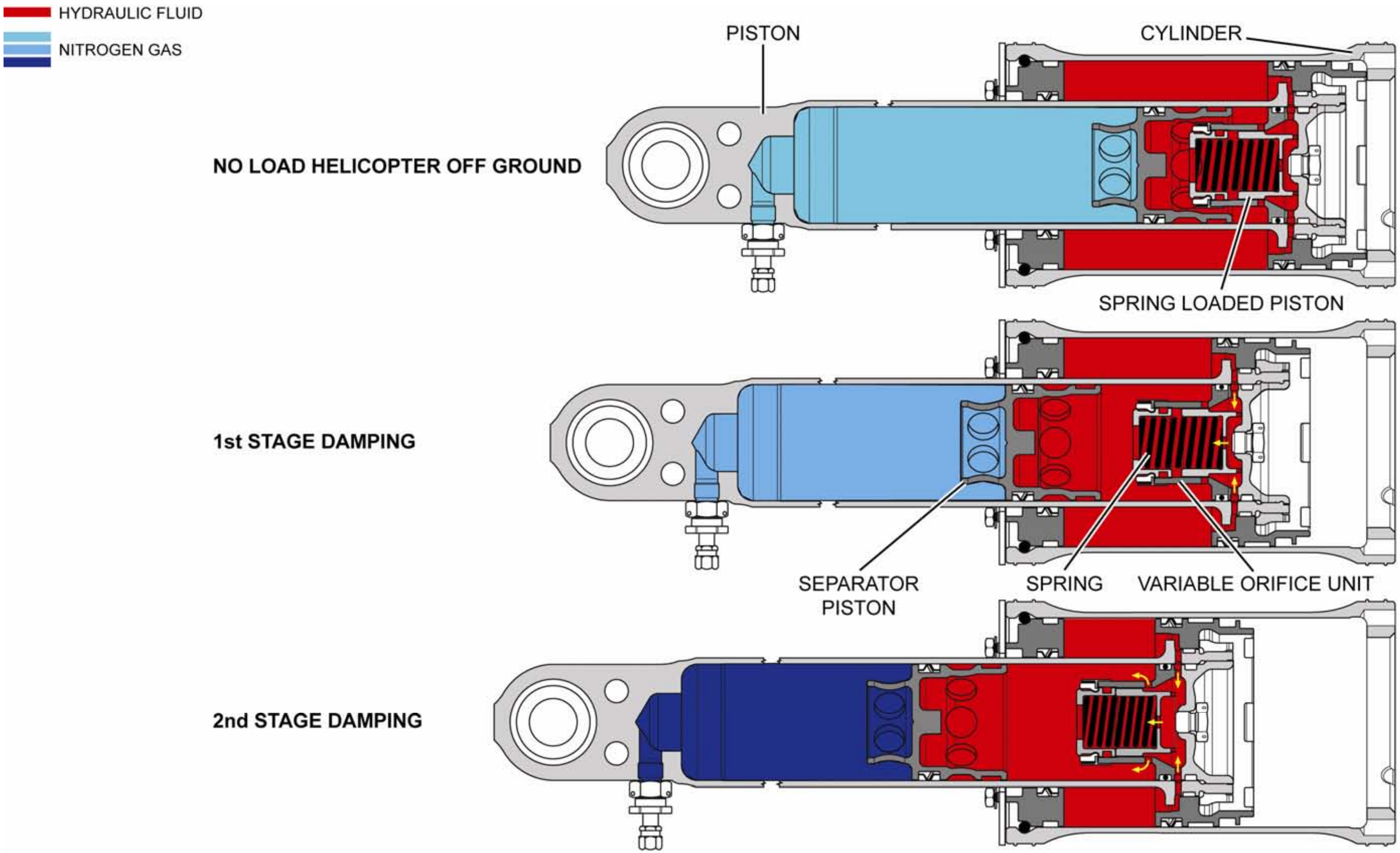
FIRST STAGE DAMPING

The first stage damping starts as the helicopter touches the ground: the trailing arm, as a result of the impact, pulls the piston rod and moves the separator piston.
The pressure of the oil in the chamber increases and consequently the oil is forced to flow through the variable device orifice.

SECOND STAGE DAMPING

During the second stage damping, the displacement forces compress the spring loaded piston which, as a consequence, opens the holes allowing the oil to flow to the separator piston. The separator piston is pushed against the nitrogen which gets pressurized and absorbs landing loads.

When the nitrogen has completely absorbed the landing loads, the pressure is at its maximum and the gas starts to extend pushing the separator back to its position forcing the oil to get back to the barrel through the holes. The shock absorber starts retracting and the oil flows back to the barrel forcing the spring loaded piston to return to its initial position closing the holes.



MAIN LANDING GEAR SHOCK ABSORBER

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NOSE LANDING GEAR – GENERAL

The nose landing gear (NLG) is a telescopic retractable type with an oleo-pneumatic damping system and twin tube-type wheels.

The main components of the nose landing gear are

- the extension/retraction actuator
- the shock absorber assembly
- the torque links

The extension/retraction actuator is part of the extension/retraction system described later.

The nose-wheel is free to swivel for taxiing or towing and automatically centers the locks at lift-off.

A center-lock device also permits keeping the nose wheel centered on the ground.

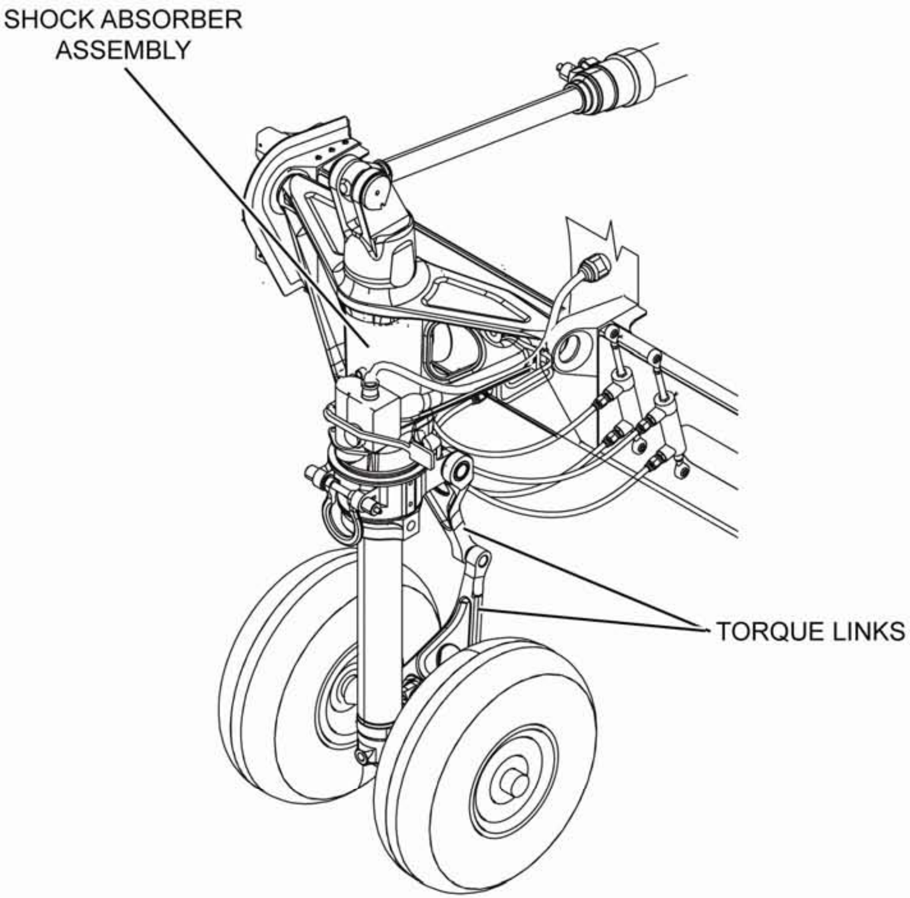
NOSE LANDING GEAR – MAIN COMPONENTS

SHOCK ABSORBER ASSEMBLY

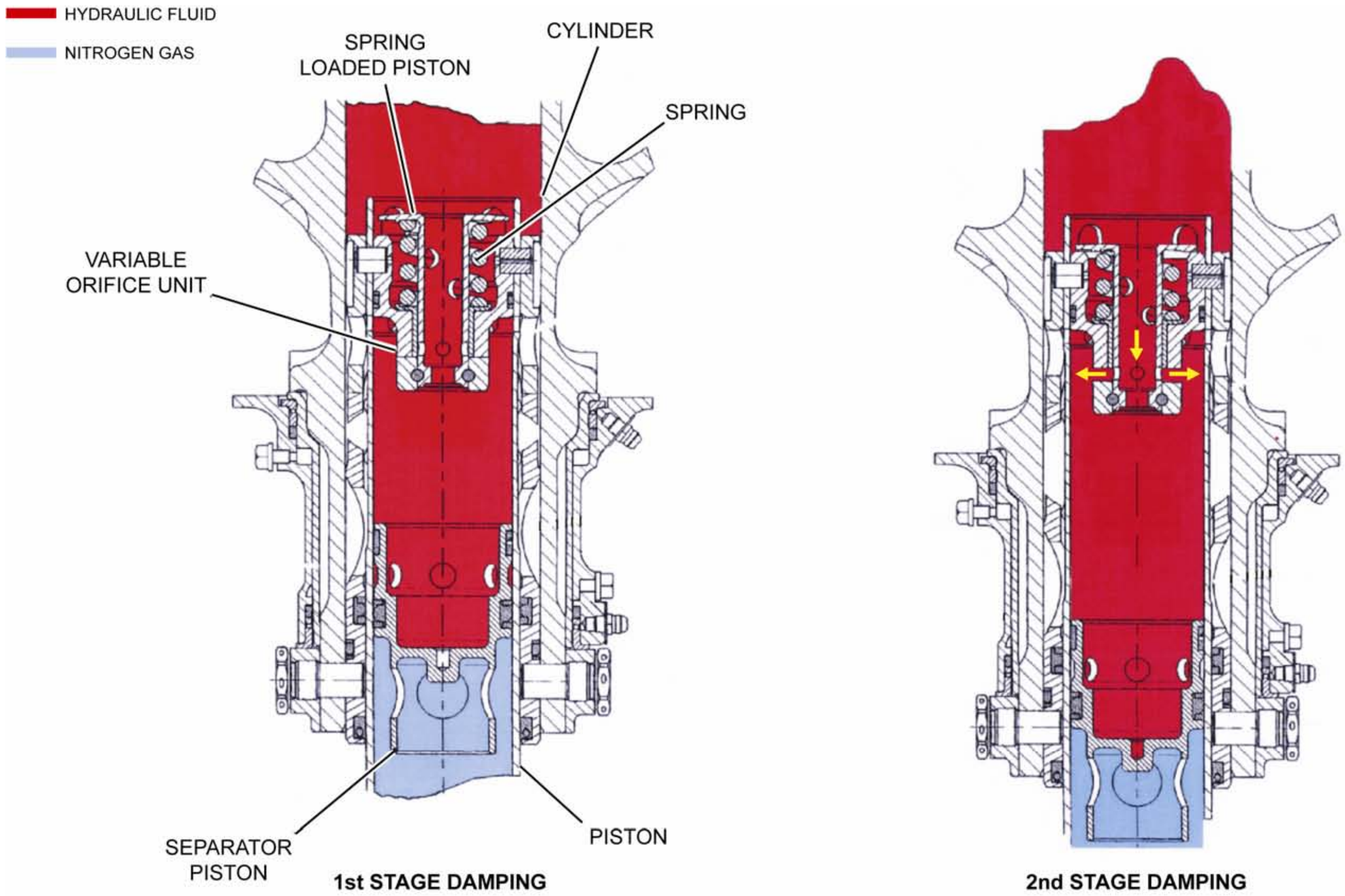
The shock absorber assembly is a nitrogen/hydraulic fluid damper made by a sliding piston rod inside a cylinder. The sliding piston rod includes a separator piston that prevents mixing of the gas with the hydraulic fluid and a valve that controls the landing gear damping and rebound actions.

TORQUE LINK

The two torque links are mechanical components allowing the nose landing gear strut to move vertically



NOSE LANDING GEAR – MAIN COMPONENTS



NOSE LANDING GEAR SHOCK ABSORBER

MAIN AND NOSE LANDING GEAR - CONTROLS AND INDICATORS

1. Landing Gear Control Lever (LGCL)

   UP ……………………… landing gear (main and nose) retracted

   DOWN………………… landing gear (main and nose) extended

   NOTE 1. LGCL must be pulled to operate.

   NOTE 2. When the helicopter is on the ground the lever is locked in the DOWN position.

2. Landing Gear Control Panel (LGCP) lights

   all extinguished .... landing gear retracted

   3 green triangles .... landing gear down and locked

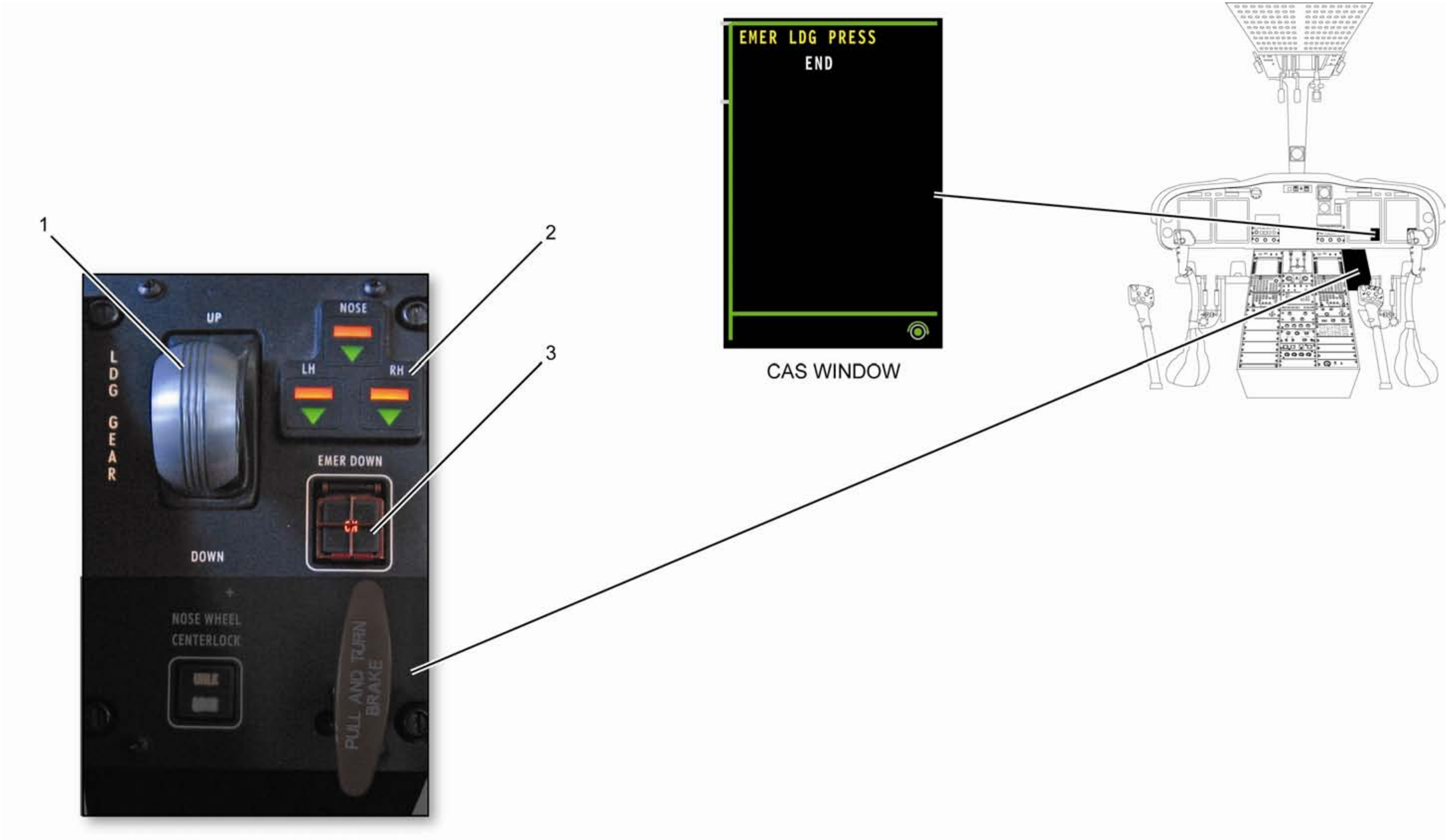
   3 amber bars ………... landing gear unlocked and/or in transition between UP and DOWN positions or viceversa (landing gear unsafe)

   NOTE. NOSE = NLG, LH = Left MLG, RH = Right MLG

3. EMER DOWN guarded push-button switch

   pressed

   ON ……………………. the landing gear extends in emergency conditions



MAIN AND NOSE LANDING GEAR – CONTROLS AND INDICATORS

EXTENSION/RETRACTION SYSTEM - GENERAL

The extension/retraction system allows the extension and retraction of the main and nose landing gears under inputs controlled by the pilot.

The extension/retraction system comprises

- one Landing Gear Control Valve (LGCV)
- two main and one nose landing gear extension/retraction actuators (RA)
- two main landing gear shortening actuators (SA)
- the Landing Gear Control Panel (LGCP)
- the Landing Gear Control Lever (LGCL)
- the microswitches

The extension/retraction system is hydraulically powered by

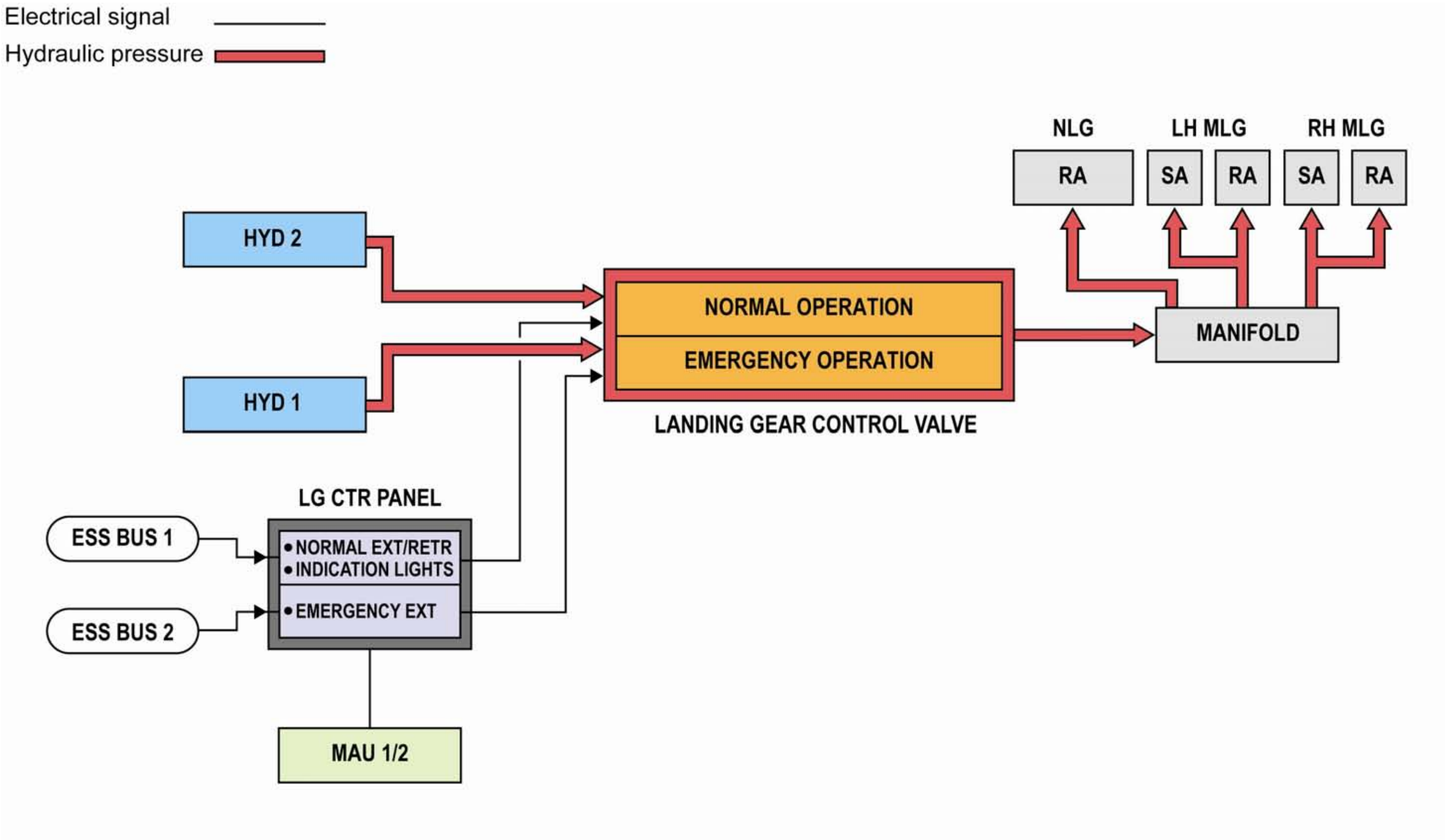
- hydraulic system no.2 for normal extension/retraction
- hydraulic system no.1 for emergency extension

The LGCV acts on a manifold, which interface and operates the hydraulic actuators of the nose and main landing gears.

The extension/retraction system is electrically operated through the Landing Gear Control Panel (LGCP) located in the cockpit and through a logical sequence controlled by electrical microswitches.

The landing gear is mechanically locked when fully extended and hydraulic pressure is received from actuators.

The landing gear is held in the retracted position by hydraulic pressure; in case of loss of hydraulic power the landing gear legs come out of their bays by effect of their own weight.



EXTENSION/RETRACTION SYSTEM – BLOCK DIAGRAM

EXTENSION/RETRACTION SYSTEM–MAIN COMPONENTS

LANDING GEAR CONTROL PANEL (LGCP) AND LANDING GEAR CONTROL LEVER (LGCL)

The Landing Gear Control Panel is located on the forward RH side of the central console. The LGCP includes the controls and indications for:

- LG normal extension/retraction (LGCL and LG status annunciators)
- LG emergency extension (EMER DOWN guarded push-button switch)
- Nose wheel centerlock (pushbutton switch)
- Park brake handle

The LGCL has two positions: UP and DOWN to control the landing gear control valve, LG normal retraction and extension. The control lever must be pulled before operating it. The LG status annunciators show the LG position with the following logic:

- 3 green lights for DOWN and LOCKED position with LGCL at DOWN
- all lights extinguished for LG in UP position with LGCL at UP
- 3 amber lights for any other condition (LG unsafe)
- Green/amber light to display the NLG center lock status

The LGCL locked in the DOWN position when the helicopter is on the ground (WOW microswitch).

EMERGENCY DOWN (EMER DOWN) PUSHBUTTON SWITCH

An EMER DOWN guarded switch control the emergency section of the Landing Gear Control Valve. The switch is able to move down the landing gears in case of failure in the normal hydraulic circuit of the control lever. In this case the emergency switch will be used to electrically actuate the emergency spool of the Landing Gear Control Valve.

LANDING GEAR CONTROL VALVE (LGCV)

The Landing Gear Control Valve (LGCV) controls the hydraulic fluid flow towards to the landing gear actuators. The LGCV comprises two solenoid valves identified as UP and DOWN solenoid valves.
When the pilot moves the Landing Gear Control Lever to the UP or DOWN positions, electrical power is supplied to the relevant solenoid valve that, in turn, supply hydraulic power to the UP or DOWN circuit respectively.
The landing-gear control valve is connected to the hydraulic operated components of the NLG and MLG via a manifold.

MLG EXTENSION / RETRACTION ACTUATOR

The extension/retraction actuator is an hydraulic actuator which, together with the shortening actuator, extends and retracts the MLG according to the pilot selection. The

extension/retraction actuator is attached to a structural element of the MLG (the trunnion).

MLG SHORTENING ACTUATOR (SA)

The main landing gear Shortening Actuator (SA) increases its length for the retraction of the landing gear and decreases its length for the extension.

NLG EXTENSION / RETRACTION ACTUATOR (RA)

The NLG extension/retraction actuator is an hydraulic actuator which extends and retracts the NLG according to pilot selections on the Landing Gear Control Lever (LGCL).

POSITION MICROSWITCHES

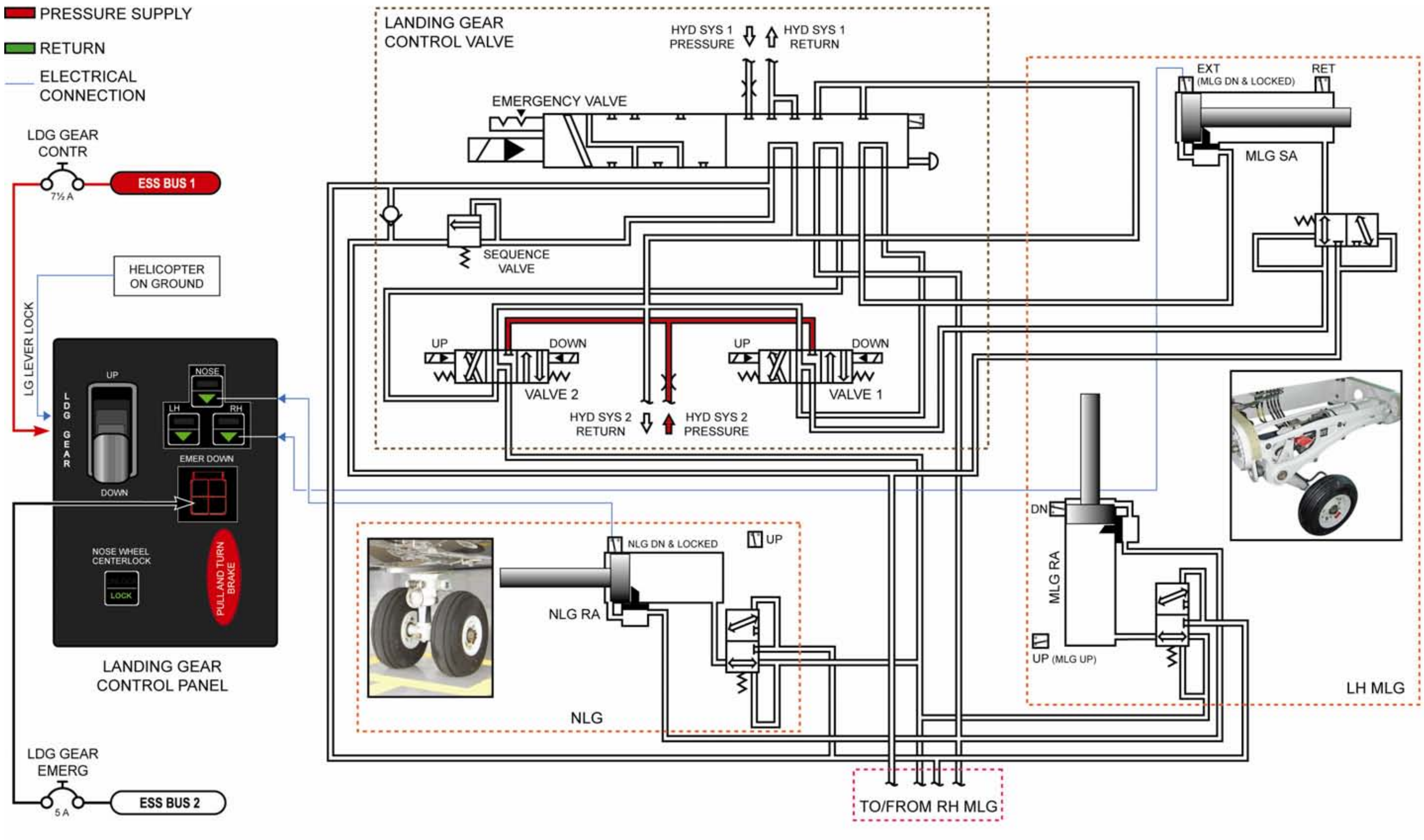
Position microswitches monitor the extended, retracted and locked conditions of each actuator of the MLG and NLG to control the normal extension/retraction sequences and provide LG status indications on the LGCP.

- the MLG and NLG retracted microswitches. These microswitches are operated when the landing gears are fully stowed in the aircraft bays (following UP selection)
- the MLG and NLG RA extended and locked microswitches. These microswitches are operated when the internal mechanical lock is engaged (following DOWN selection)
- the MLG SA actuator extended microswitches. These microswitches are operated by the shock absorber when the SA actuator is fully extended (following UP selection)
- the MLG SA actuator retracted microswitches. These microswitches are operated when the SA actuator is fully retracted (following DOWN selection)
- the MLG SA actuator locked microswitches. These microswitches are located inside the actuator and are operated by an internal device when the SA actuator is fully retracted (following DOWN selection)
- the Weight On Wheels (WOW) microswitches. These microswitches are located on the shock absorber of each leg of the MLG and provide information of "aircraft on the ground" or "aircraft in flight" status to various helicopter systems.

LANDING GEAR – PRINCIPLE OF OPERATION

ON GROUND

When the helicopter is on the ground the Landing Gear Control Lever (LGCL) is locked DOWN by a pin controlled by the Weight-On-Wheel (WOW) microswitches.



ON GROUND – LANDING GEAR DOWN AND LOCKED

NORMAL OPERATIONS

The pilot operates the normal retraction and extension of the landing gear acting on the LGCL which controls the Landing Gear Control Valve (LGCV).

The following figures show the extension and retraction operation sequenced in phases.

Note: only the LH MLG is shown in the diagram; RH MLG operates in the same way.

Actuator position microswitches control all the extension/retraction phases.

NORMAL RETRACTION

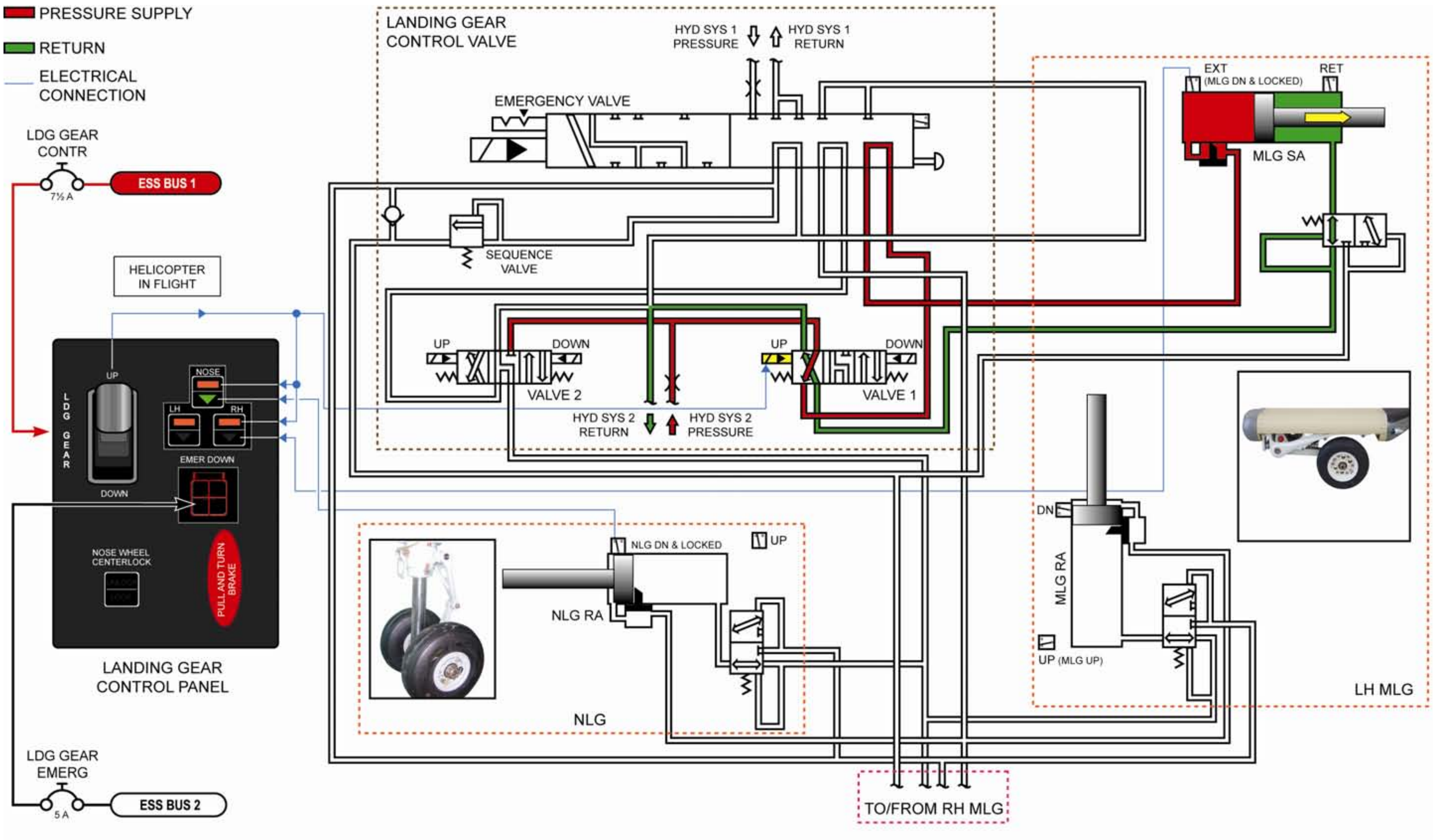
Normal retraction occurs in two phases:

PHASE 1 – MLG SHORTENING

When the helicopter lifts-off, the WOW release the LGCL locking pin and the pilot can move the LGCL to UP position. The NOSE, LH and RH amber lights on the LGCP are illuminated because the LGCL position is UP while the LG is not.

The retraction procedure starts only if the NLG is locked in the center position (LOCK illuminated green in the NOSE WHEEL CENTERLOCK pushbutton).

The VALVE 1 UP solenoid is energized and pressure from the hydraulic system no.2 is delivered to the MLG Shortening Actuator (MLG SA) to unlock from DN position and to shorten the MLG. As the SA piston unlocks, the position microswitch extinguishes the relevant green triangular annunciator on the LGCP.

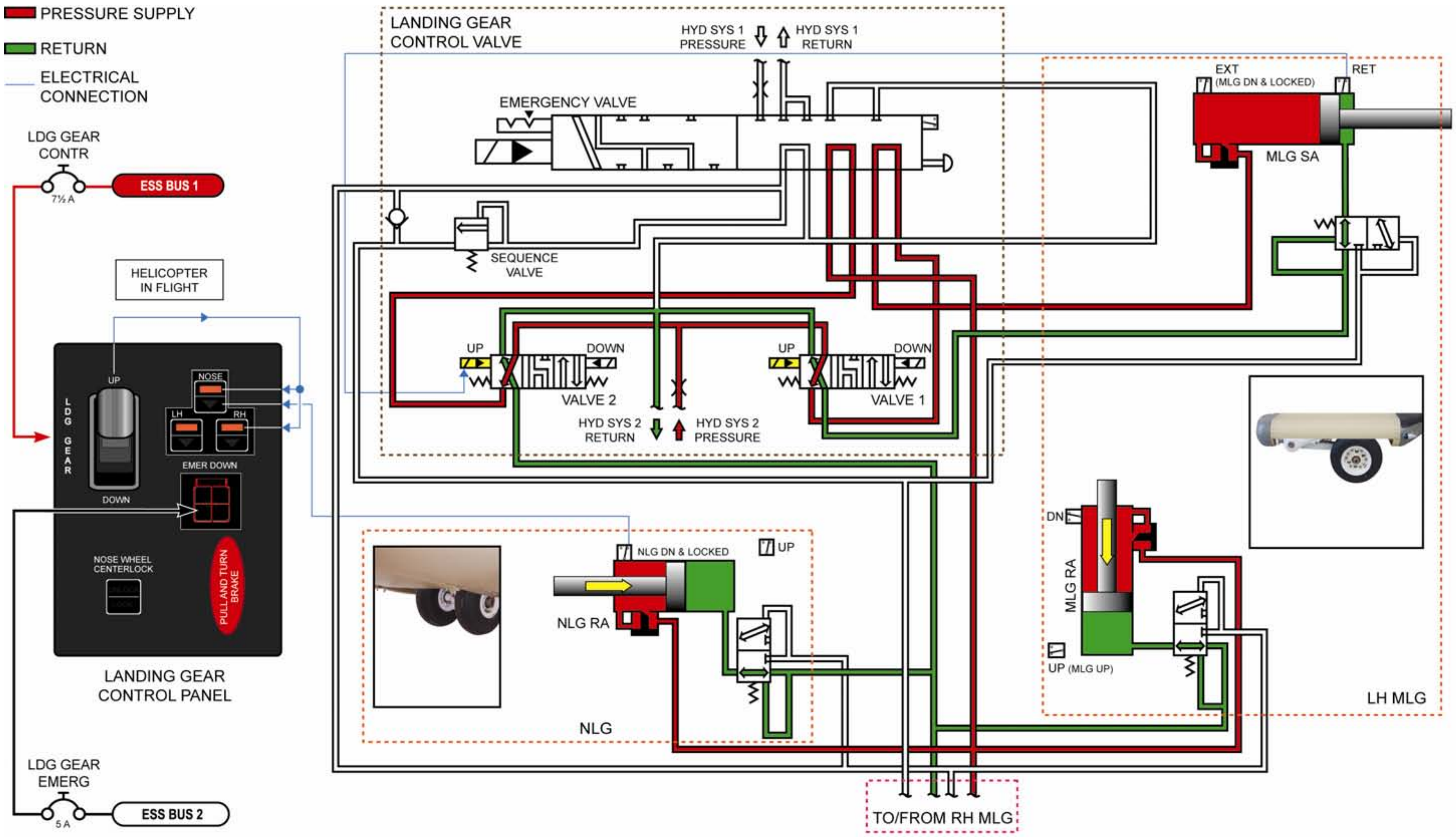


NORMAL RETRACTION PHASE 1 – MLG SHORTENING

PHASE 2 – NLG and MLG RETRACTION

When both SA are fully retracted, the VALVE 2 UP solenoid is also energized and pressure is delivered to NLG and MLG Retraction Actuators (RA).

As the NLG RA piston unlocks, the position microswitch extinguishes the NOSE green triangular annunciator on LGCP.

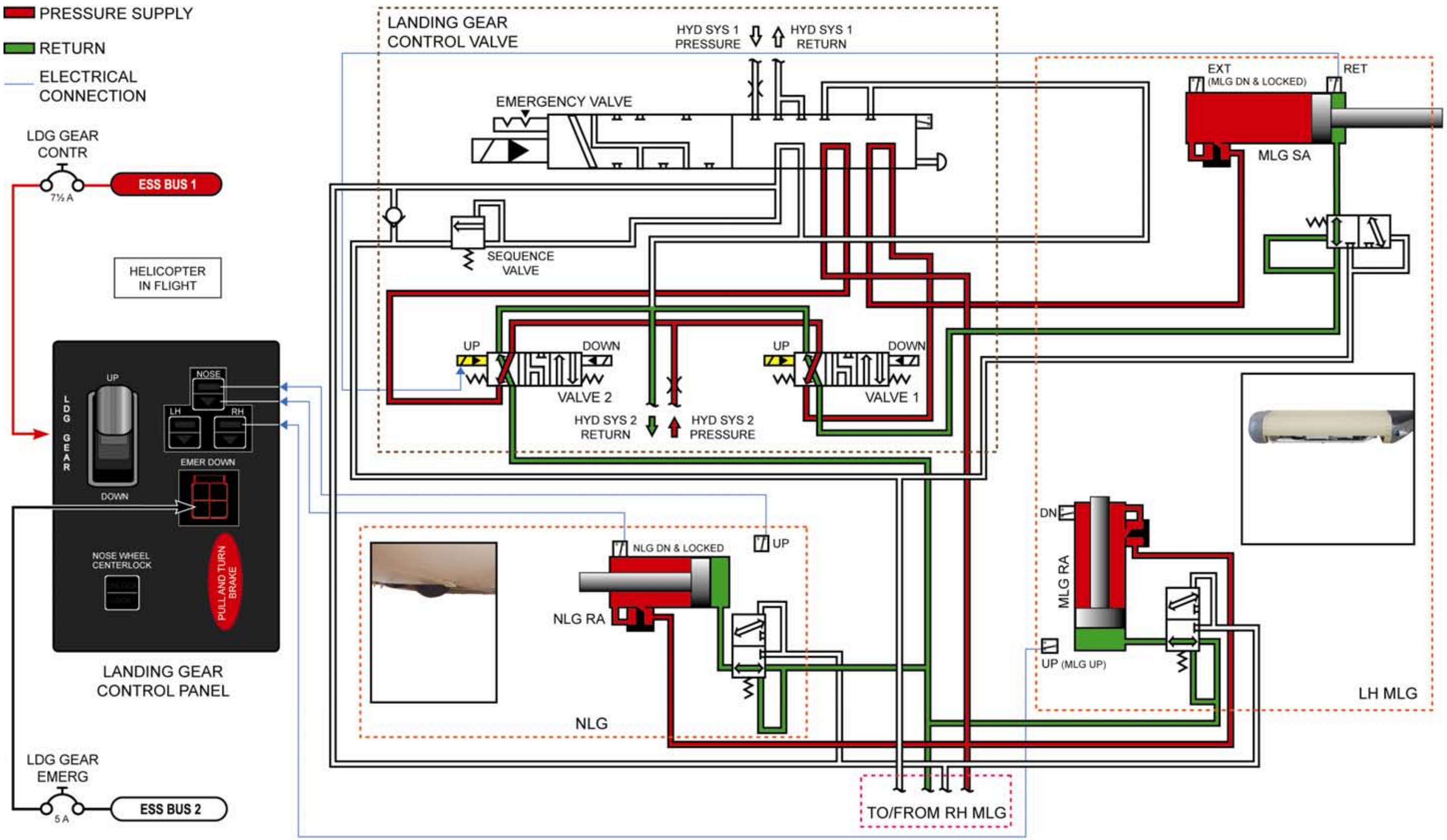


NORMAL RETRACTION PHASE 2 – NLG AND MLG RETRACTION

LANDING GEAR RETRACTED

When the NLG RA and MLG RA are fully retracted and each landing gear contacts the relevant UP microswitch on the airframe, the corresponding NOSE, LH or RH amber light on the LGCP extinguishes.

Both VALVE 1 and VALVE 2 UP solenoids remain energized to keep the pressure in all actuators as long as the LG is retracted.



LANDING GEAR RETRACTED

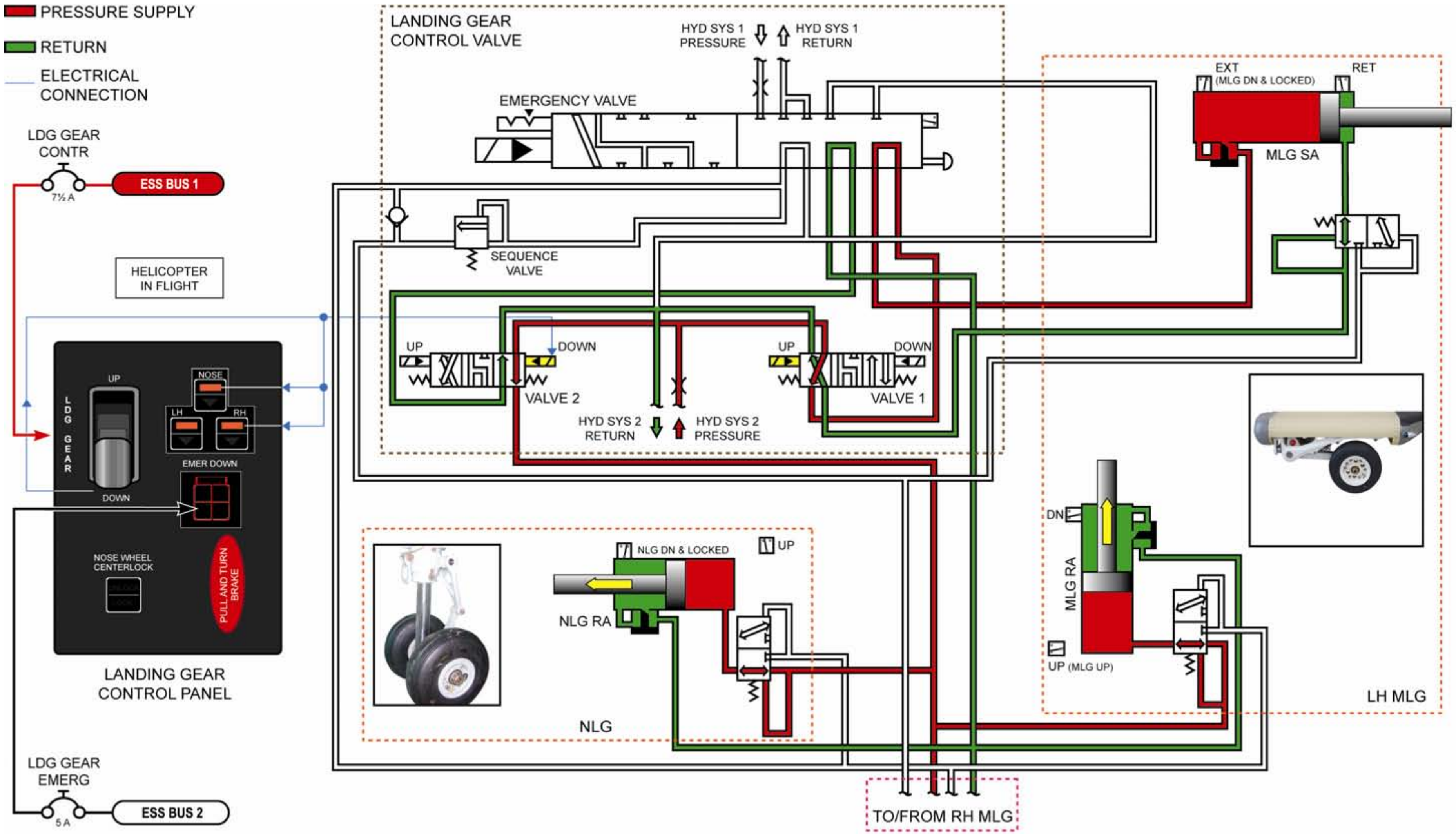
NORMAL EXTENSION

Normal extension occurs in two phases:

PHASE 1 – NLG and MLG EXTENDING

When the pilot moves the LGCL to DOWN position, the NOSE, LH and RH amber lights on the LGCP illuminate because the LGCL is DOWN while the LG is not DOWN and LOCKED.

With LGCL at DOWN, the VALVE 2 DOWN solenoid energizes and pressure from the hydraulic system no.2 is delivered to the NLG and MLG retraction actuators to retract.

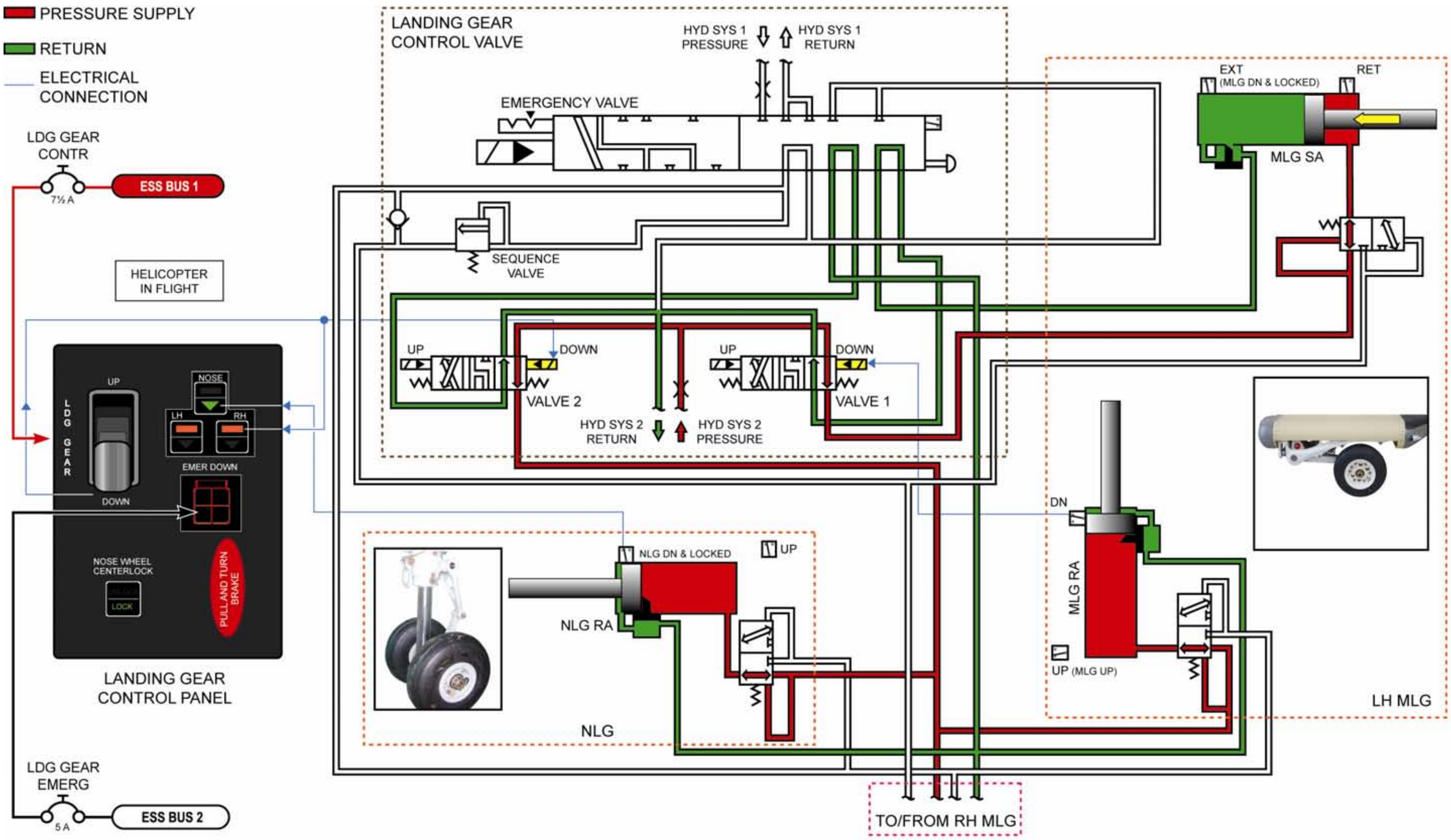


NORMAL EXTENSION PHASE 1 – NLG AND MLG EXTENDING

PHASE 2 - NLG DN and LOCKED, MLG ELONGATING

When the NLG RA is fully extended and locked, the position microswitch illuminates the NOSE green triangular annunciator (NLG DN & LOCKED) and extinguishes the corresponding amber light.

When both MLG RA are fully extended (DN) the position microswitches energize, the VALVE 1 DOWN solenoid and pressure is delivered to the MLG SA to elongate (extend) the MLG.

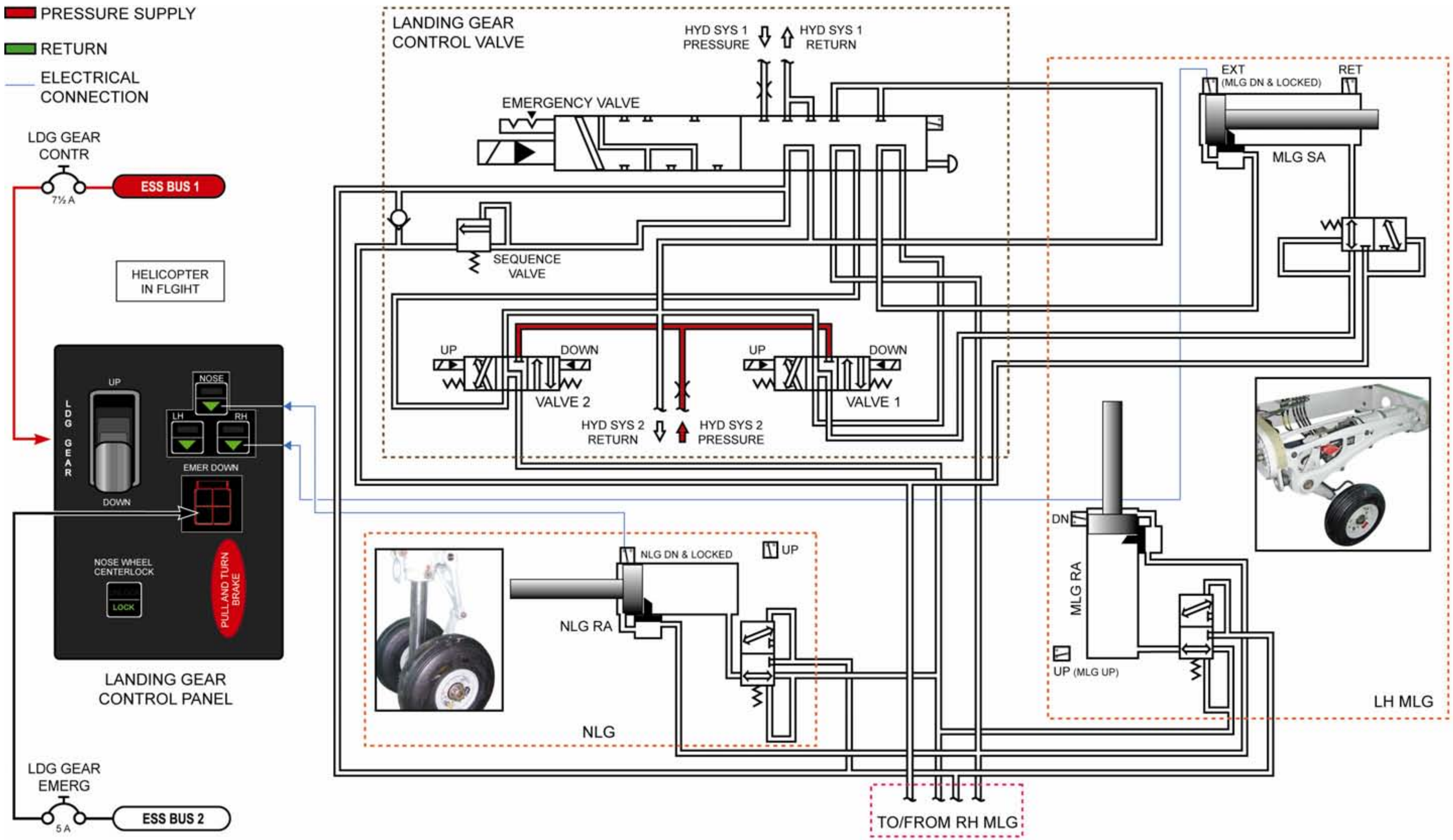


NORMAL EXTENSION PHASE 2 – NLG DN AND LOCKED, MLG ELONGATING

LANDING GEAR DOWN AND LOCKED

When both SA are fully extended and locked, the VALVE 1 DOWN and VALVE 2 DOWN solenoids are de-energized so that the system is no more pressurized.

The three green lights NOSE, LH and RH are all illuminated and the three amber lights are all extinguished.



LANDING GEAR DOWN AND LOCKED

EMERGENCY EXTENSION

Emergency extension permits lowering of the LG in case the normal operation is ineffective.

Emergency extension of the landing gear is controlled using the EMER DOWN pushbutton which acts on the emergency valve of the LGCV and utilities:

- hydraulic power from system no.1
- electrical power from ESS BUS 2

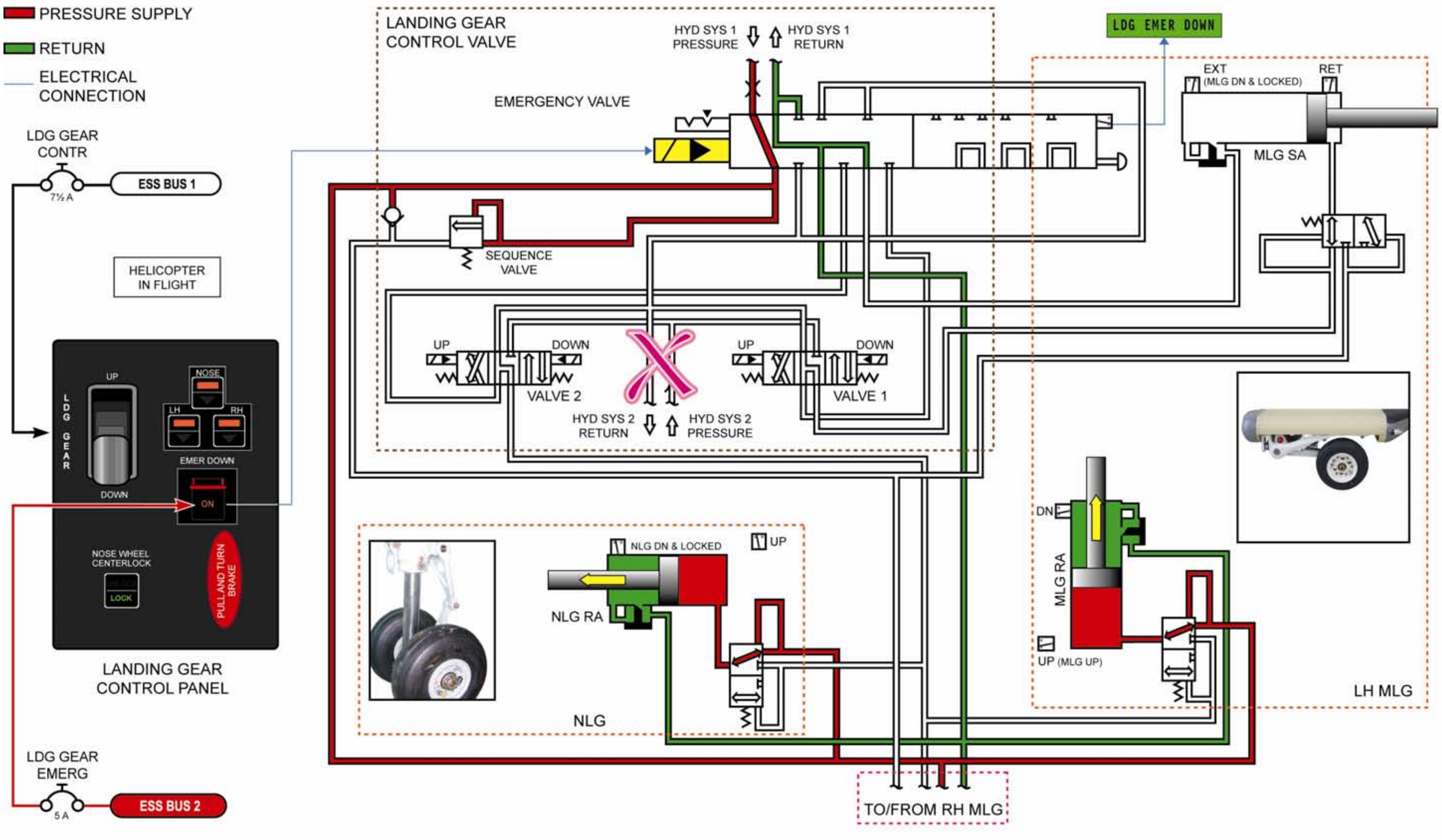
The position of the Landing Gear Control Lever (UP or DOWN) is not relevant for emergency extension.

Emergency extension occurs in two phases.

PHASE 1

If, after selecting the LGCL to DOWN, any LG position indicators remain blank or amber and normal landing gear extension is confirmed to be ineffective, the emergency extension is to be carried out by lifting the guard and pressing the EMER DOWN pushbutton switch which illuminates (amber ON legend).
The emergency valve solenoid is energized and pressure from hydraulic system no.1 is delivered to the MLG SA to elongate (extend) the MLG and the NLG RA.
The NOSE, LH and RH amber lights illuminate.

The emergency valve position microswitch provides the advisory LDG EMER DOWN on the CAS window of the MFD.

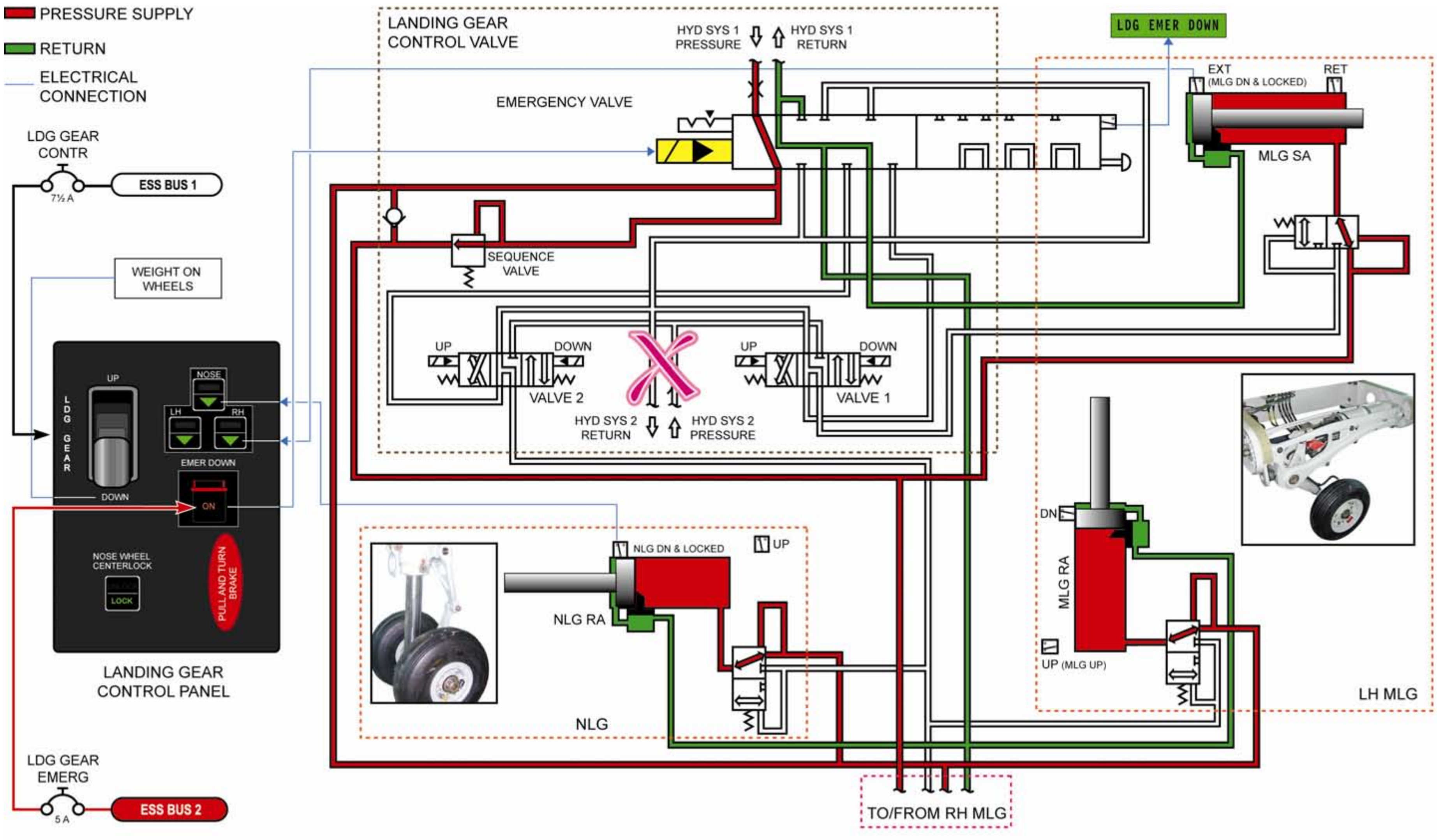


EMERGENCY EXTENSION – PHASE 1 (TRANSITIONING)

PHASE 2 (TRANSITIONING)

When MLG RA and NLG RA are fully down, pressure restores in the lines and the sequence valve opens.

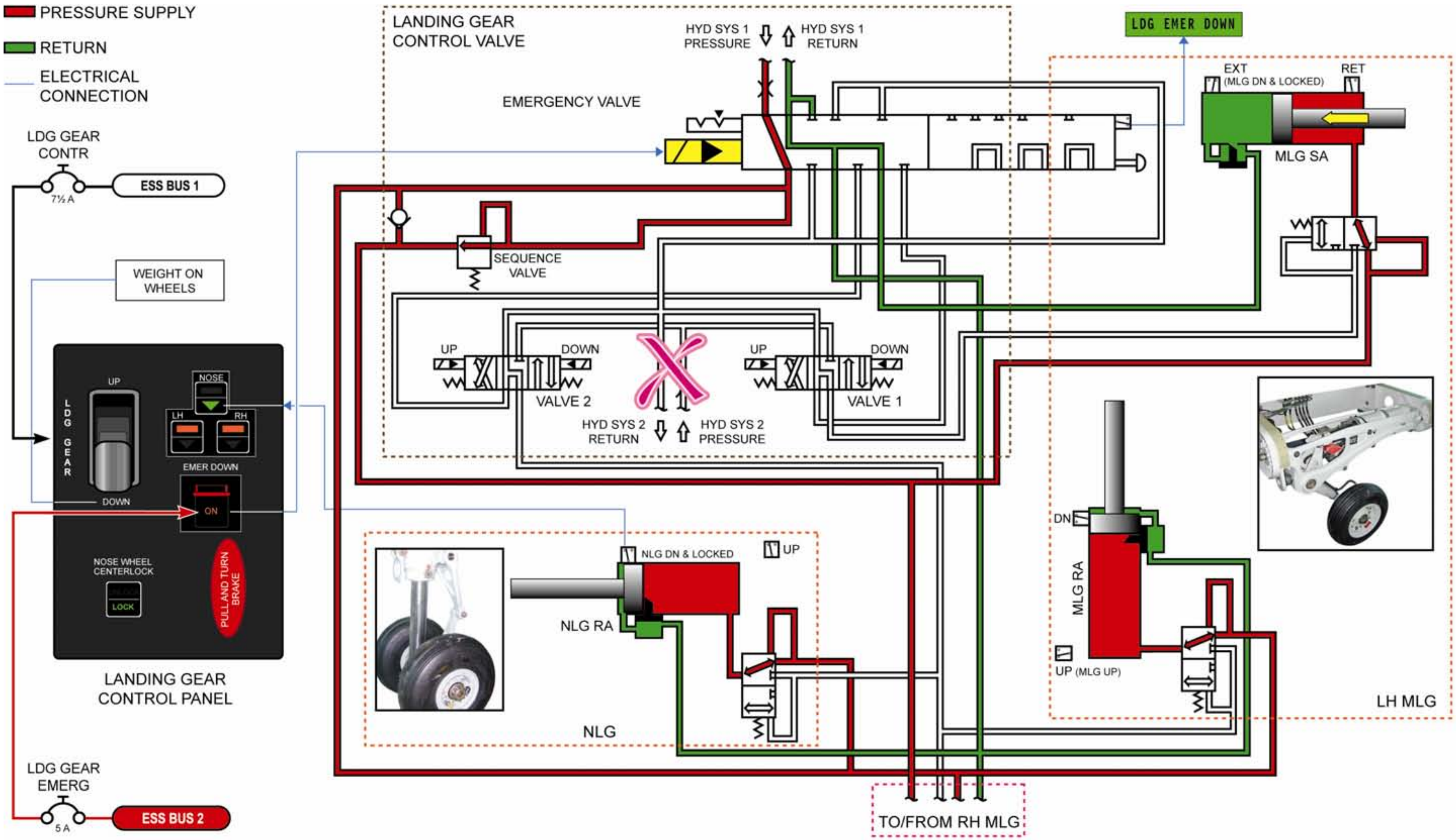
Pressure is delivered to MLG SA to elongate (extend) the MLG.



EMERGENCY EXTENSION – PHASE 2 (TRANSITIONING)

PHASE 2 (END)

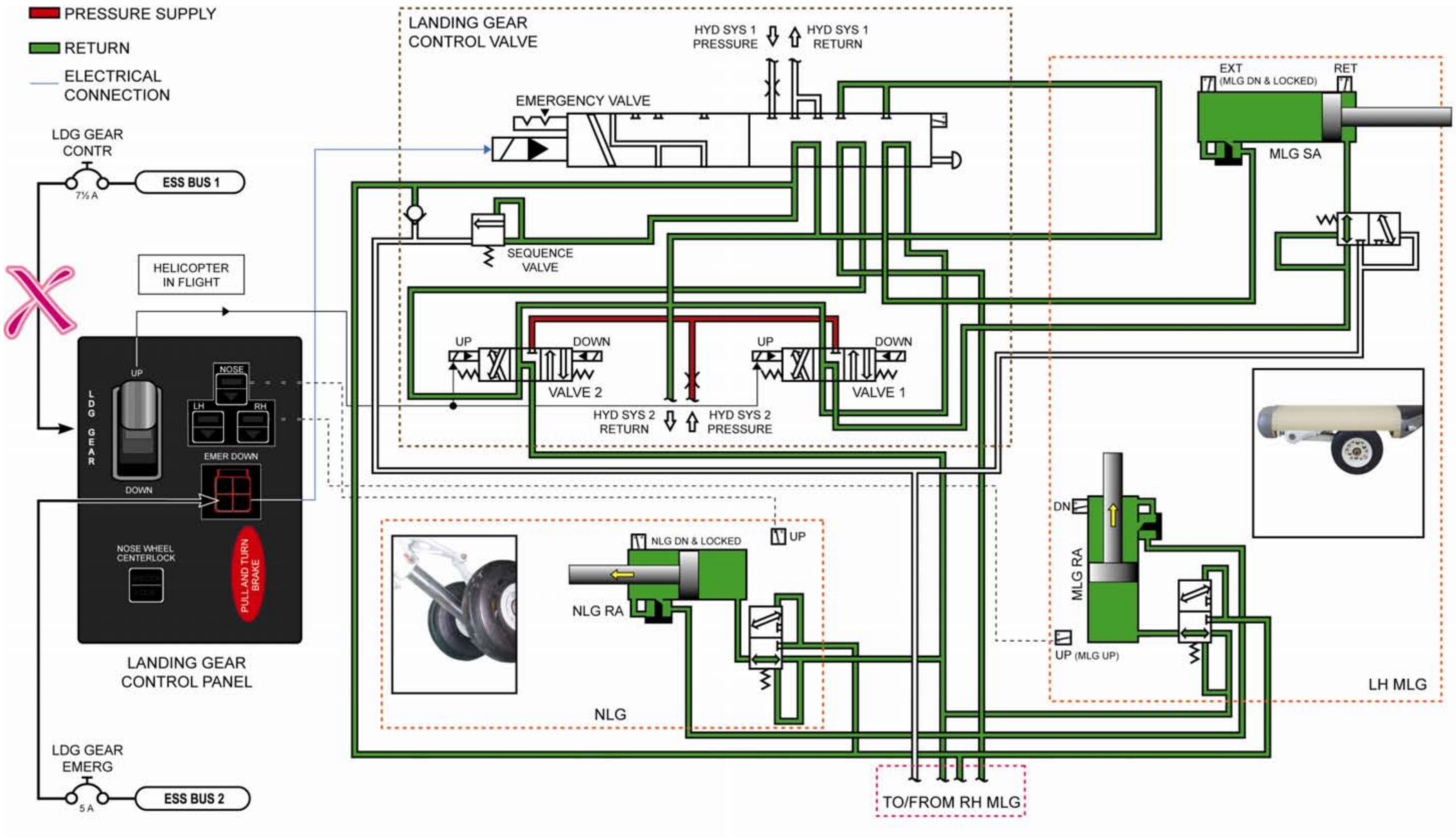
When all legs are down and locked, the three triangular green lights illuminate.



EMERGENCY EXTENSION – PHASE 2 (END)

LOSS OF ELECTRICAL POWER WITH LG UP

In case of loss of electrical power from ESS BUS 1 or LDG GEAR CONTR circuit breaker tripped, the solenoids inside the LGCV de-energize and pressure is relieved from LG actuators and landing gear fall because of gravity. The LGCL is not effective and the lights in the LGCP cannot illuminate. Emergency extension is to be carried out.



LOSS OF ELECTRICAL POWER WITH LG UP

NOSE WHEEL STEERING AND CENTER-LOCK SYSTEM – GENERAL

The nose wheel steering system gives the pilot control of the aircraft during ground movement. After take-off, centering mechanisms will automatically align the wheels in a fore and aft direction for retraction in the NLG bay.
For high speed rolling, the nose wheels need to be held directionally aligned.

The nose wheel center-lock is controlled by a switch on the Landing Gear Control Panel (LGCP) or mechanically by the rotation of a lever connected to the center-lock assembly.

Pressing the switch once engages the center-lock, locking the nose wheels in a directionally aligned fore and aft position. Pressing the switch a second time will disengage the center-lock, leaving the nose wheels free for ground manoeuvring.

NOSE WHEEL STEERING AND CENTER-LOCK SYSTEM – MAIN COMPONENTS

CENTER LOCK ASSEMBLY

The center lock assembly is composed by an actuator incorporating a gear assembly that drives a locking pin.
The locking pin can be engaged/disengaged:

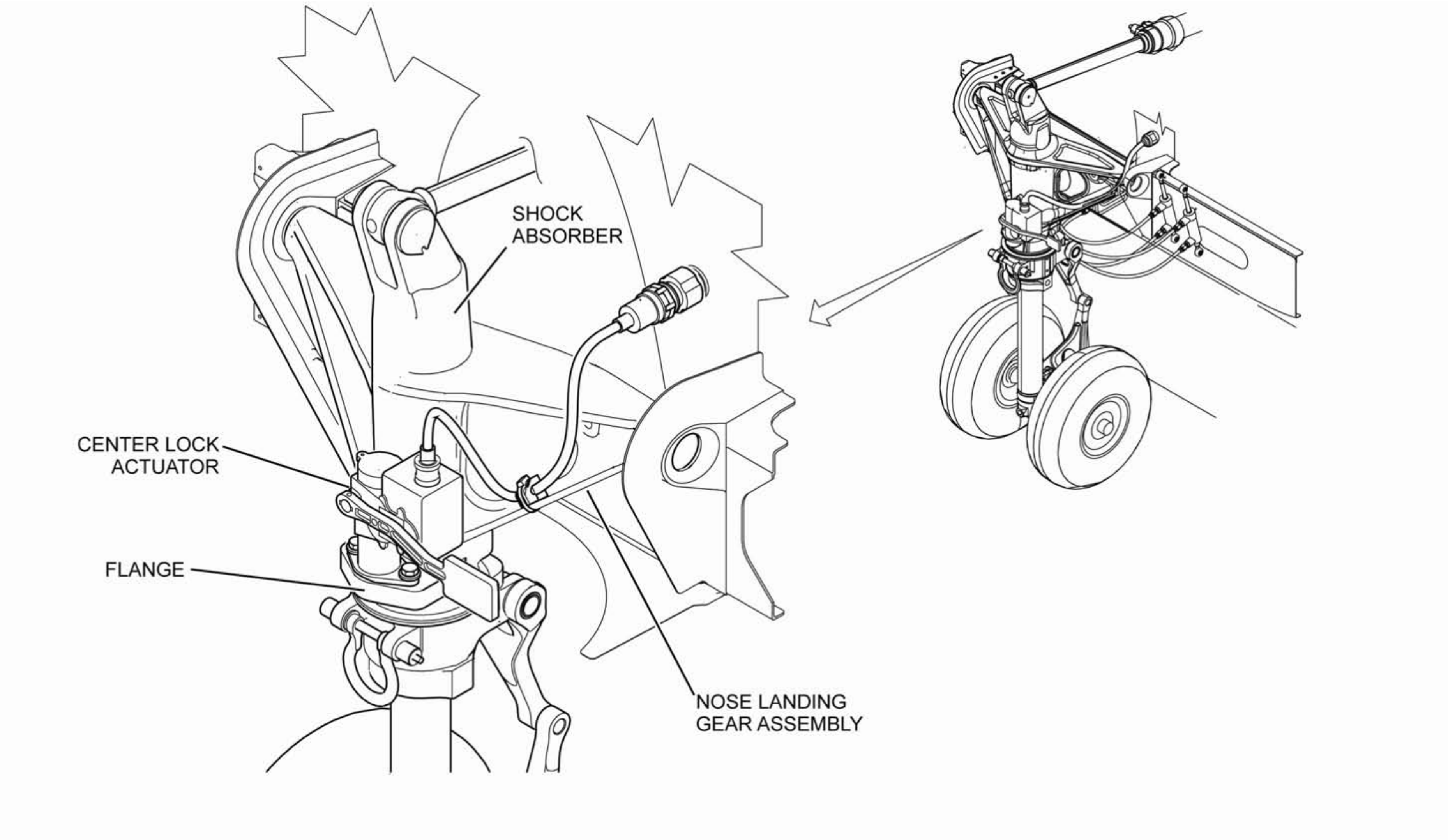
- electrically by means of the NOSE WHEEL CENTERLOCK push-button switch on the LGCP;
- mechanically by the rotation of a lever connected to the assembly.

The locking pin fits into a hole provided on a flange on the bottom of the shock absorber cylinder.

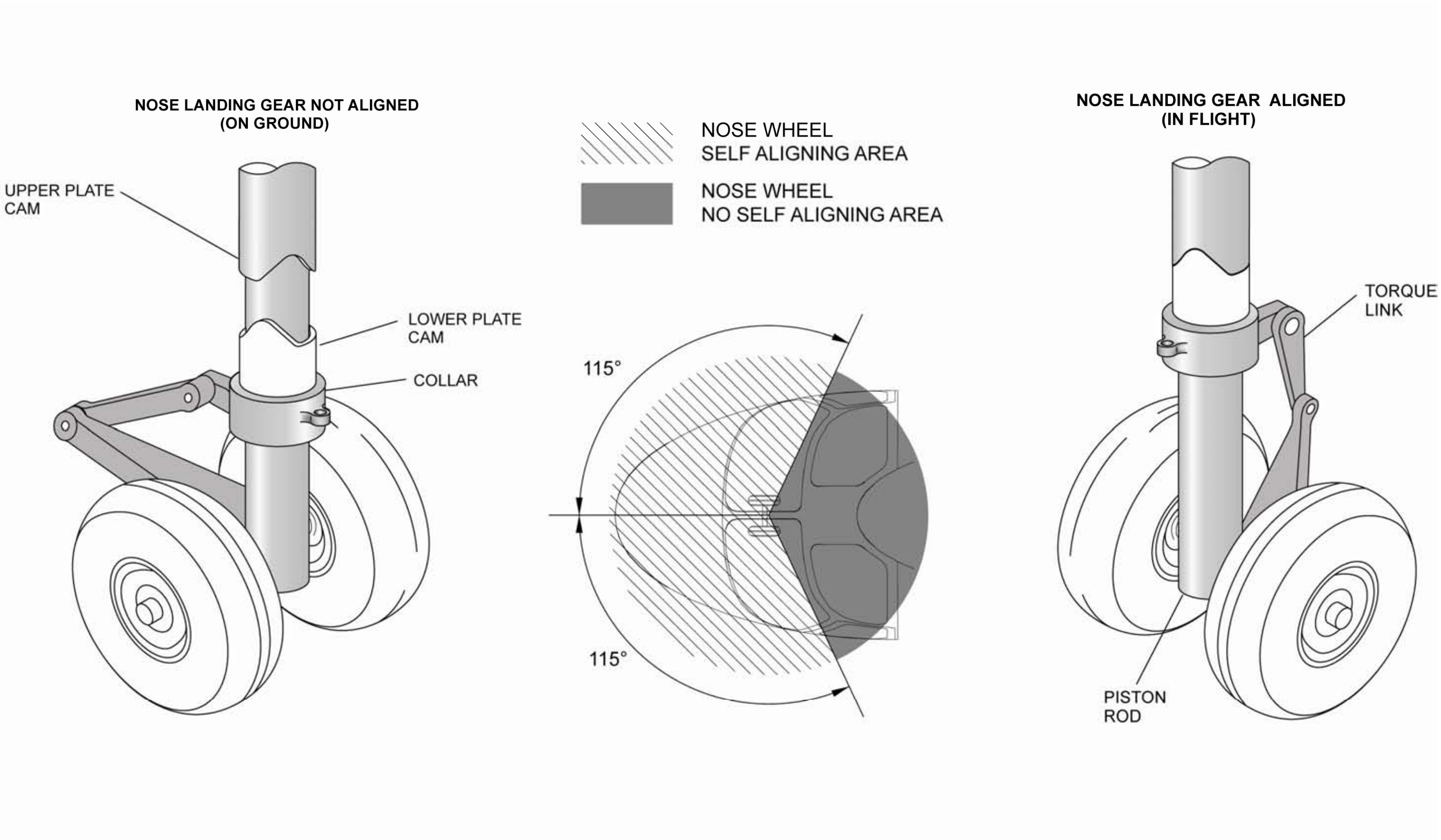
CENTERING ASSEMBLY

The centering assembly comprises two cam shaped plates one at the top and the other at the bottom inside the shock absorber cylinder.
When the nose landing gear is centered, the two cam plates match together providing the fore and aft alignment of the nose landing gear.

When the pilot is taking-off with the nose landing gear out of the fore and aft position, the sliding piston rod moves downwards to its fully extended position by effect of its weight. The upper cam slips on the lower cam forcing the sliding piston rod to rotate until the two profiles continuously match together.



CENTER LOCK ASSEMBLY



CENTERING ASSEMBLY

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NOSE WHEEL STEERING AND CENTER LOCK SYSTEM – CONTROLS AND INDICATORS

1. NOSE WHEEL CENTERLOCK pushbutton

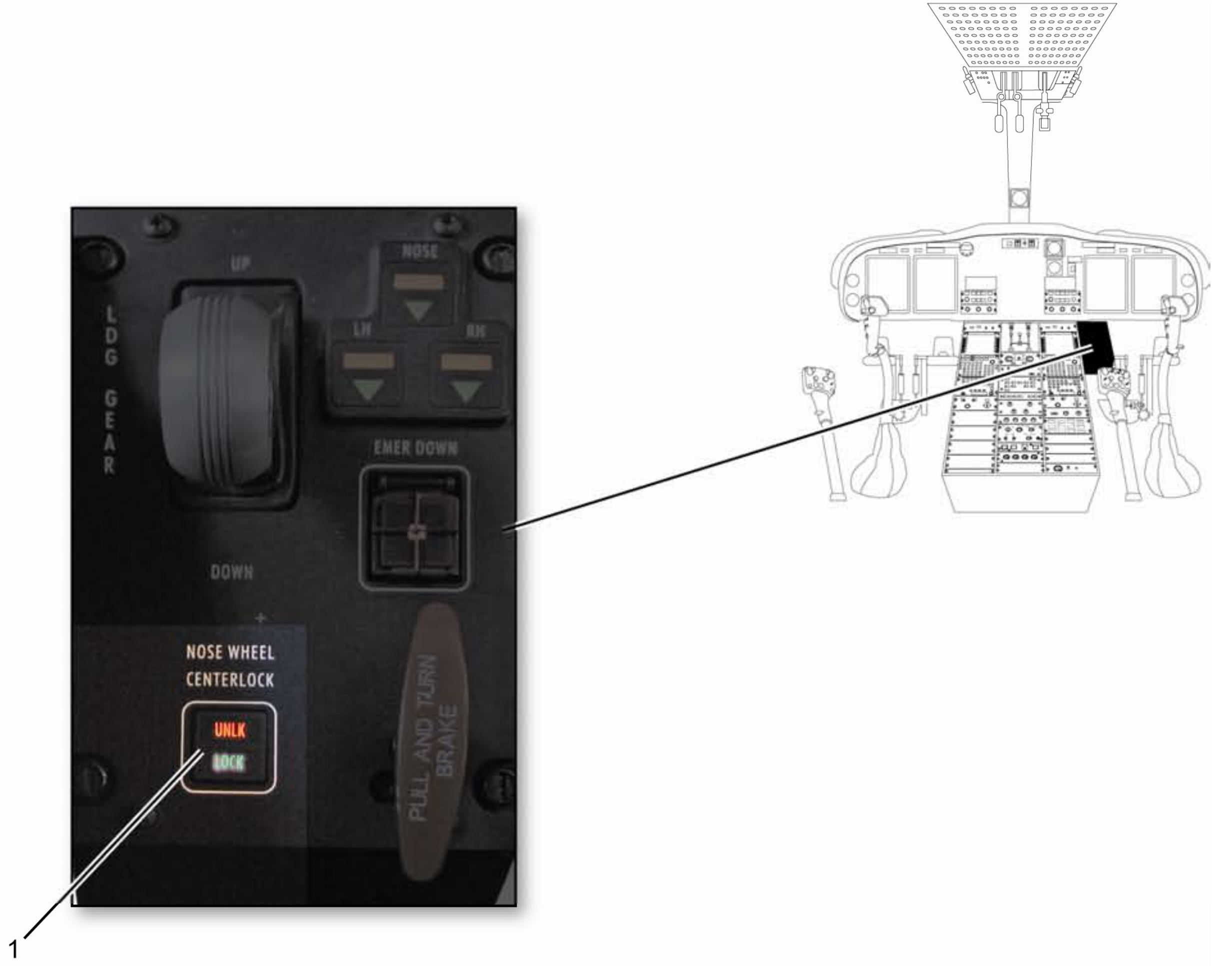
   LOCK lighted green …….. removes the lock of the nose wheel in the center position

   UNLK lighted amber …….. provides the lock of the nose wheel in the center position

   NOTE. When the green LOCK legend is illuminated the lock is engaged. If the green legend is lighted-off and the amber UNLK legend is blinking than the lock has been armed but not engaged.

   The locking logic is controlled by the WOW (Weight On Wheel) microswitch according to this rule:

   - during take-off, if the NLG is unlocked, the amber light UNLK will blink during the locking phase. When the NLG is centered and locked the green light LOCK will illuminate. At this stage it is possible to retract the landing gear.
   - during landing it is possible to unlock the NLG only when the helicopter will be on ground (WOW signal ON).



NOSE WHEEL STEERING AND CENTER LOCK SYSTEM – CONTROLS AND INDICATORS

NOSE WHEEL STEERING AND CENTERLOCK SYSTEM – OPERATIONS

COCKPIT OPERATIONS

- Nose landing gear wheels locked in center position. Pressing the NOSE WHEEL push-button on the LGCP, the electrical motor is powered and the locking pin is removed from the hole. The green caption LOCK lights off and the amber caption UNLK lights on (the caption blinks during locking pin travel). The caution NOSE WHL UNLK is provided to the MFD. The NLG is free to swivel for taxiing or towing. Static steering can be achieved by using yaw control. Dynamic steering can be achieved by using differential braking.
- Before take-off, pressing the NOSE WHEEL push-button the actuator is powered to insert the locking pin inside the hole. The amber caption UNLK lights off and the green caption LOCK lights on. If the hole is not aligned with the locking pin, the amber caption UNLK keeps blinking until they are aligned by steering the helicopter using yaw pedals (alternatively the helicopter can be lifted off such that the centering assembly will align the nose landing gear). The caution NOSE WHL UNLK on the MFD disappears.
- If the nose wheels are not centered and the helicopter lifts-off, the WOW microswitches automatically provide the electrical power to the center lock actuator to engage the locking pin. When the landing gears are retracted the green caption LOCK lights off.
- When the helicopter is in flight and the NOSE WHEEL remains unlocked, the caution NOSE WHL UNLK is illuminated and the retraction of the landing gear system is inhibited.

NOTE

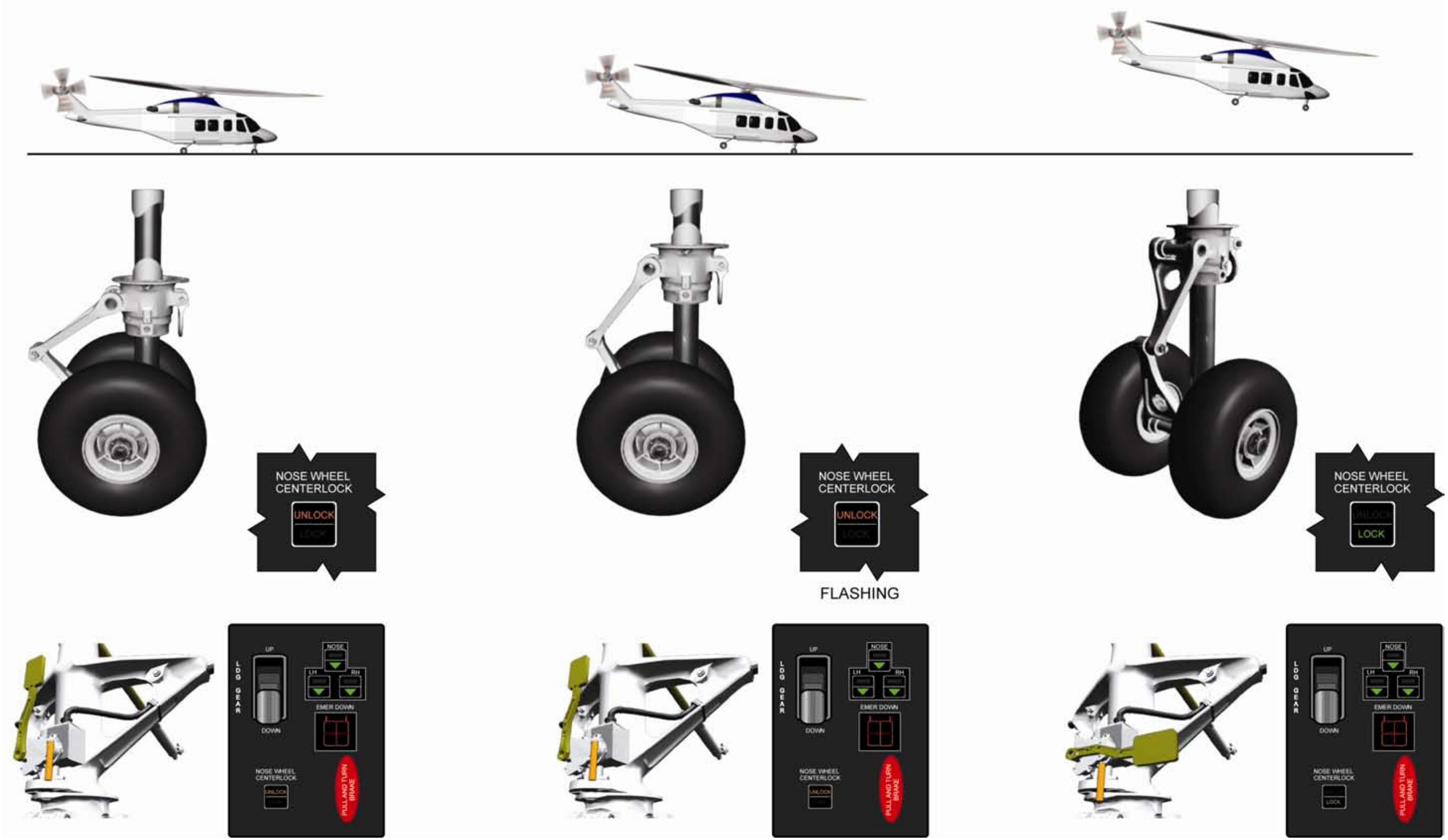
- Pilot can take-off with NLG unlocked.
- A mechanical system (CAM) provides for NLG centering.
- When airborne (Weight Off Wheels) the centerlock actuator automatically moves the locking pin into the flange hole.

MANUAL OPERATIONS

It is possible to manually engage/disengage the actuator locking pin through a yellow lever located in front of the assembly

- when the lever is horizontal the pin is engaged
- when the lever is vertical the pin is disengaged

A ground locking pin can be inserted between the lever and the body of the assembly for safe towing. The locking pin can be aligned with the hole by hand rotating the nose landing gear when the towing bar is attached.



NOSE WHEEL STEERING – COCKPIT OPERATIONS

UNLOCKED POSITION

LOCKED POSITION

NOSE WHEEL STEERING – MANUAL OPERATIONS

MANUAL OPERATIONS - CENTERING PIN POSITIONS

WHEEL BRAKE SYSTEM – GENERAL

The wheel brake system is composed by a brake assembly installed on each main landing gear wheel. The system allows two different type of braking: dynamic braking and static braking.

WHEEL BRAKE SYSTEM – MAIN COMPONENTS

PARK BRAKE CONTROL LEVER

The park brake control lever is a red lever positioned vertically. The lever has to be pulled and turned clockwise to insert the park brake. On top of the lever there is the white mark PULL AND TURN BRAKE.

RESERVOIR

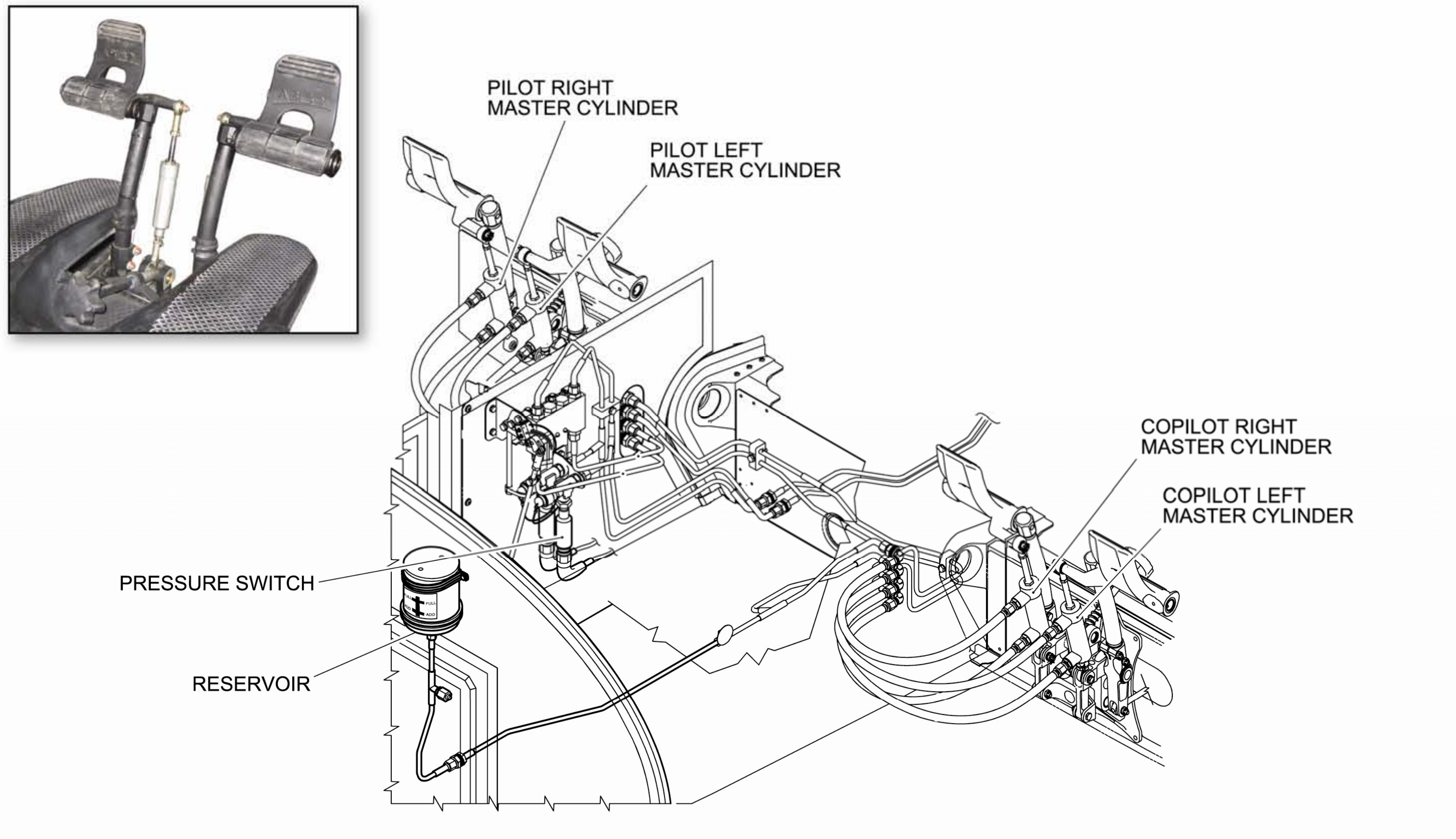
The reservoir contains the hydraulic fluid of the circuit and is directly connected to the brake master cylinders.

MASTER CYLINDERS (PILOT AND COPILOT)

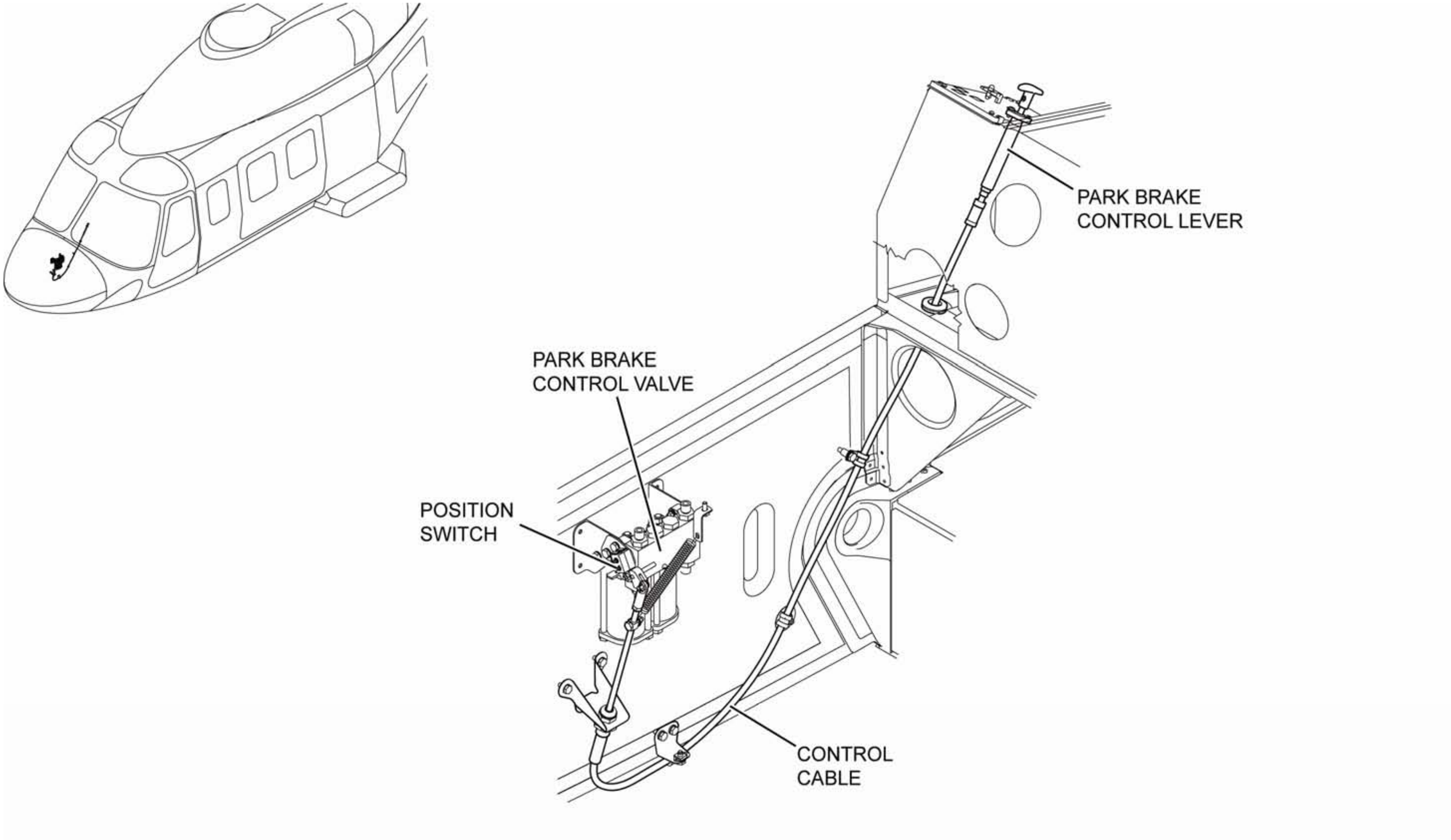
The pilot left (right) master cylinders supplies the pressure necessary to operate the left (right) brake of the main landing gear. When the pilot pushes the left (right) brake pedal, the piston of the master cylinder extends and removes the pressure from the brake. The (left and right) master cylinders of the pilot are installed in series with those of the copilot. When the pilot operates the brakes, the necessary hydraulic flow is supplied through the related copilot master cylinder.

PARK BRAKE CONTROL VALVE

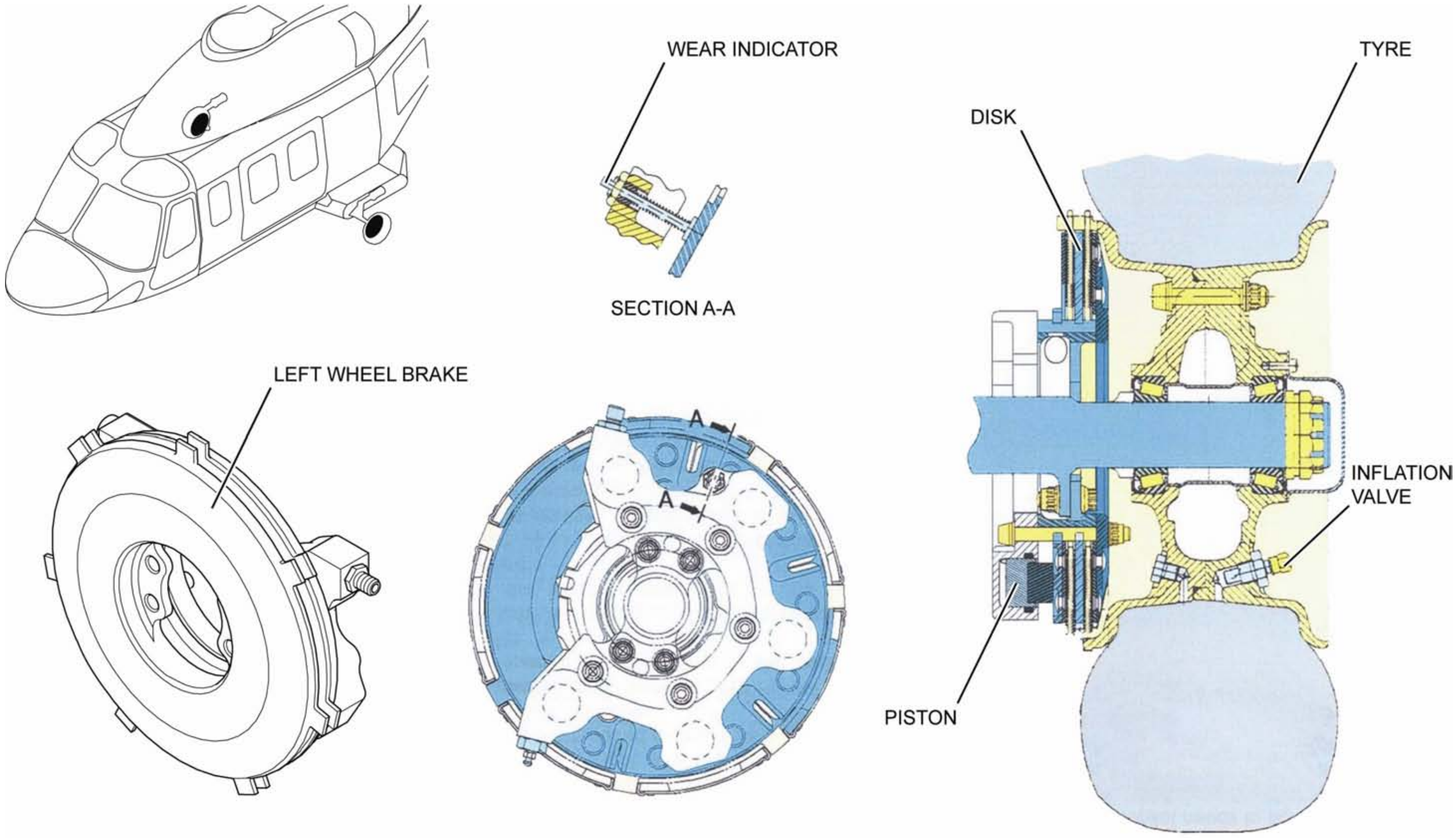
The park brake control valve is a direct-acting, mechanically operated valve. The valve is designed to trap and maintain hydraulic brake pressure when the helicopter is parked. The unit also provides automatic compensation for change in hydraulic oil volume due to thermal variations or minor linkage in the brake system. This is accomplished by means of spring loaded accumulators.



WHEEL BRAKE SYSTEM – MAIN COMPONENTS (1 OF 3)



WHEEL BRAKE SYSTEM – MAIN COMPONENTS (2 OF 3)



WHEEL BRAKE SYSTEM – MAIN COMPONENTS (3 OF 3)

WHEEL BRAKE SYSTEM – CONTROLS AND INDICATORS

The wheel brake system controls include the pilot and copilot brake pedals and the park brake handle located on the Landing Gear Control Panel.

1. COPILOT BRAKE PEDALS

   press the pedals to brake wheels

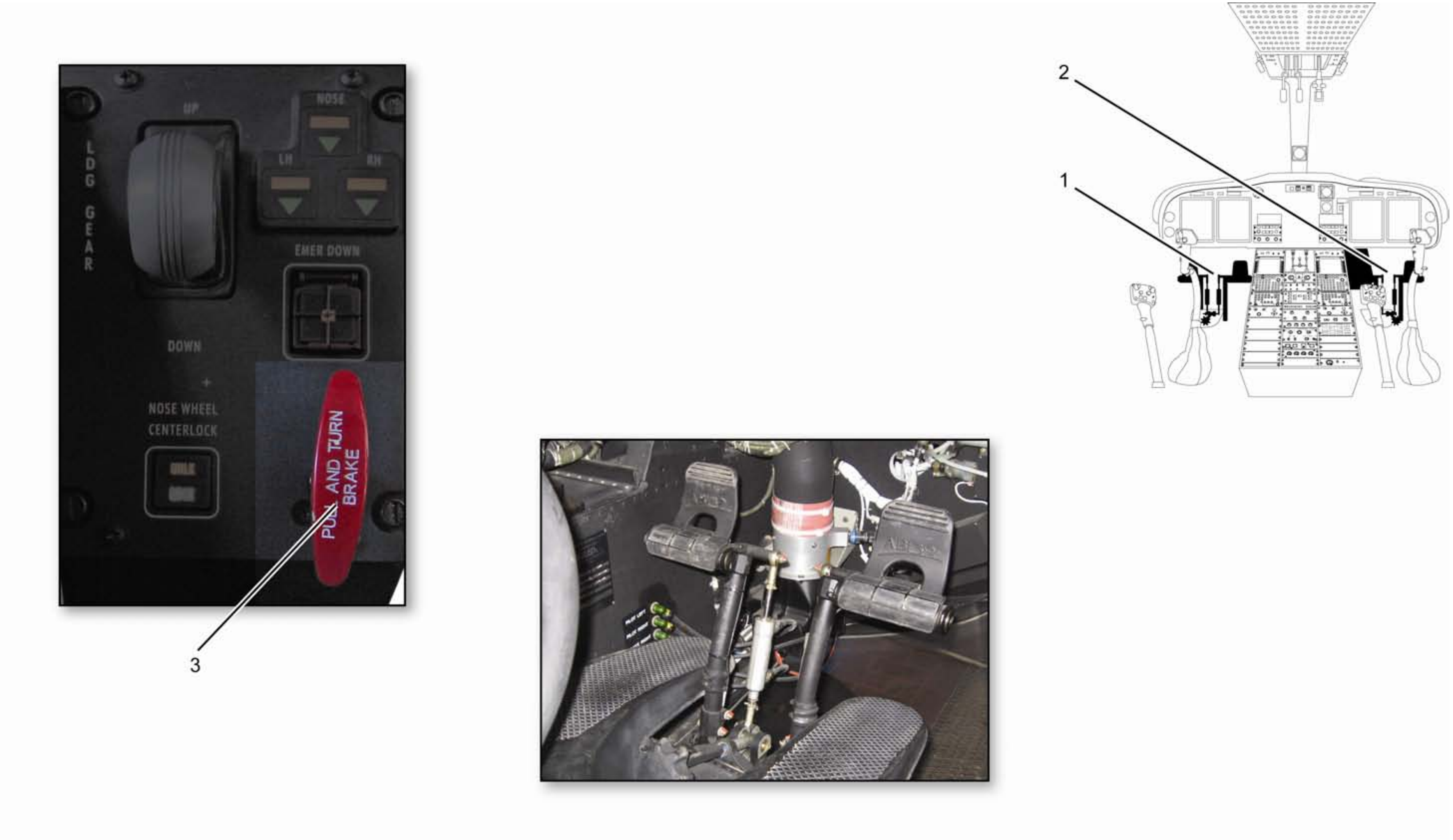
2. PILOT BRAKE PEDALS

   press the pedals to brake wheels

3. PULL AND TURN BRAKE lever

   Pull and rotate 90° clockwise the handle to insert the park brake

   It is possible to reset the handle only if the left brake pedal is preventively pushed by the pilot (or copilot) so that the pressure into the brake wheel line is drained and the lock removed.



WHEEL BRAKE SYSTEM – CONTROLS AND INDICATORS

WHEEL BRAKE SYSTEM – PRINCIPLE OF OPERATION

Dynamic braking is achieved through pedal levers installed on the yaw pedals. Each lever operate a hydraulic master cylinder which generates the required pressure by means of a hydraulic fluid contained in a reservoir. The pilot hydraulic master cylinders are supplied through the copilot ones.

There is no priority between pilot and copilot to generate pressure in the brake system.

Differential braking can be achieved during taxiing on ground by operating only left or right pedal to obtain the dynamic steering of the helicopter (the lock of the nose wheel in center position must be disengaged).

Static braking is achieved when the helicopter is parked by pulling and rotating clockwise the park brake handle and pushing on the brake pedals. To remove the locking action the pilot has to push the left brake pedal so the hydraulic pressure overcome a retaining mechanism and unblock the movement.

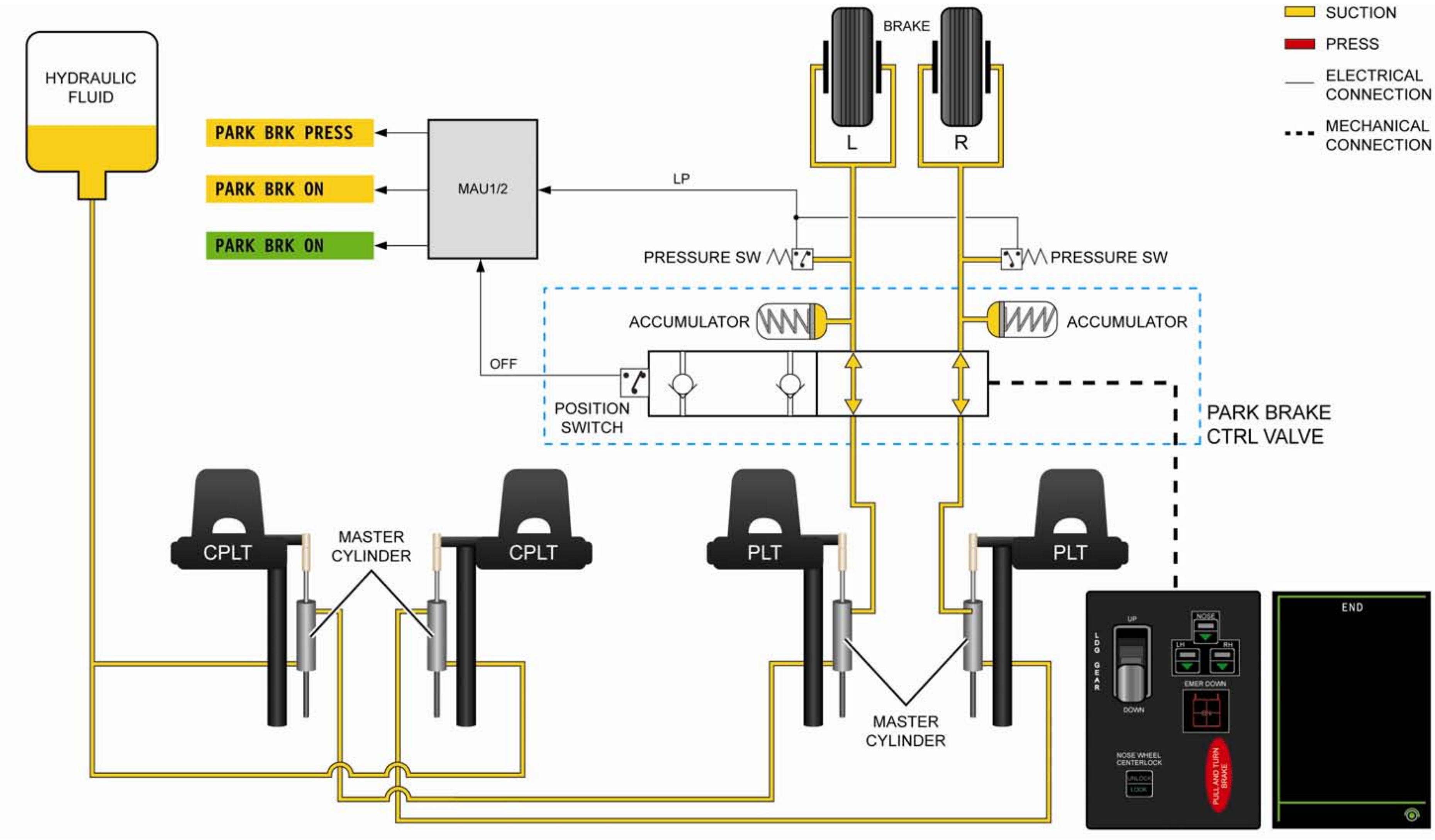
If the pressure detected by the pressure switch reaches an acceptable value, then the advisory PARK BRK ON is provided on the MFD.

If the pressure does not reach an acceptable value, then the caution PARK BRK PRESS is provided to the MFD. In this case the pilot (or the copilot) has to push again and again on the brake pedals until the caution disappears.

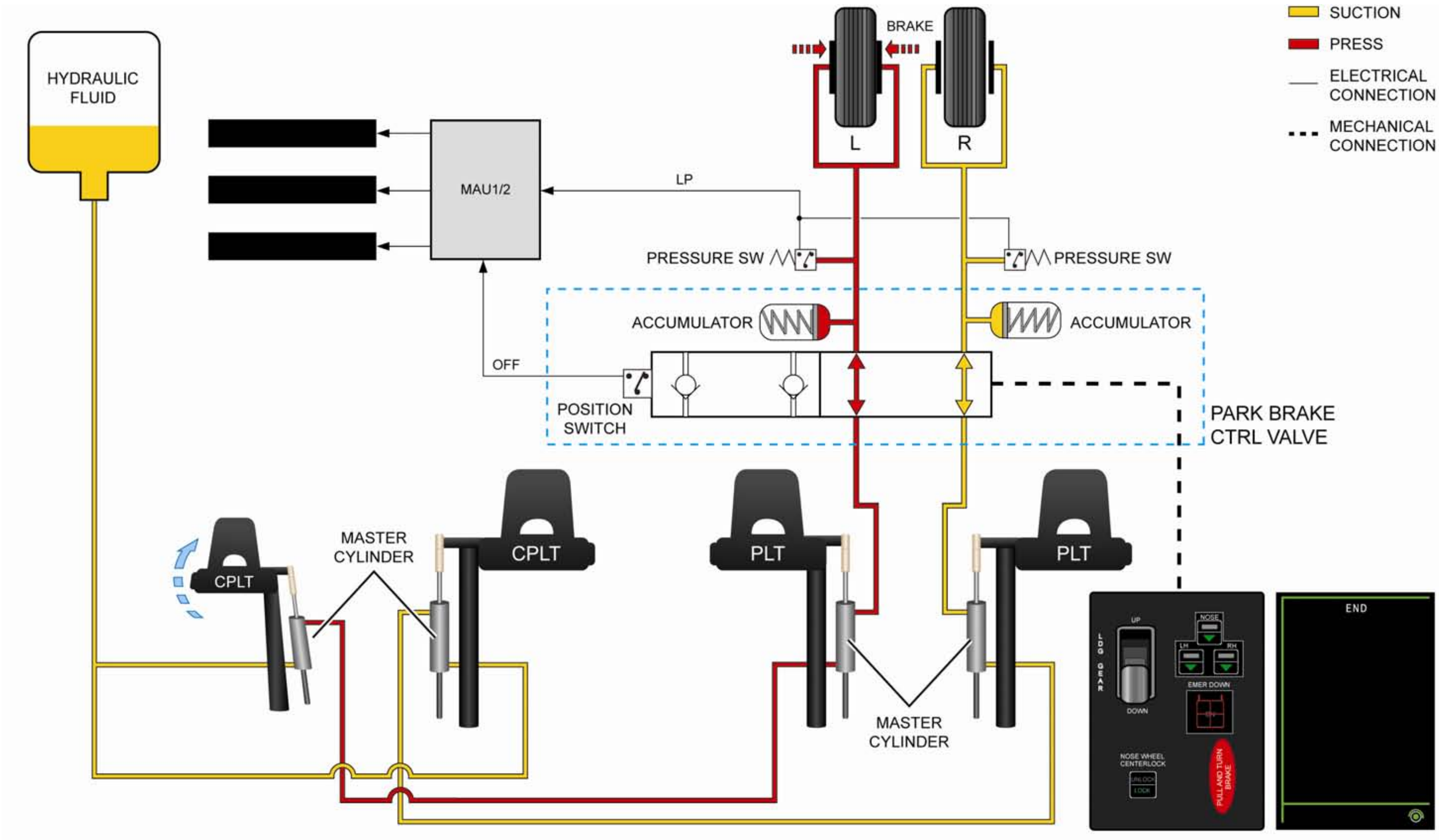
The caution and advisory messages are operative only if the park brake handle is pulled (as detected by the park brake valve position switch).

LIMITATIONS

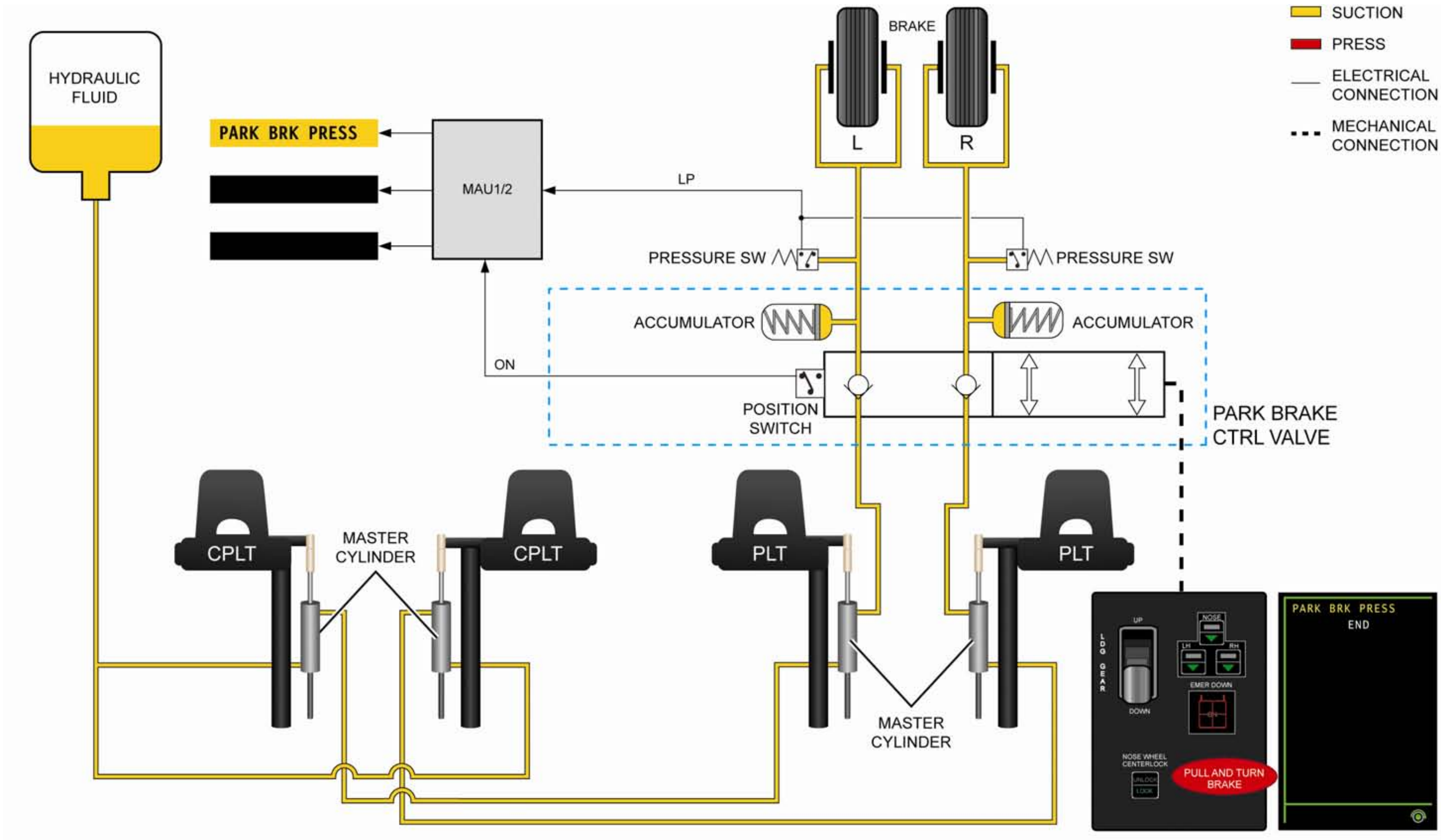
Refer to AW139-RFM-4D Section 1.



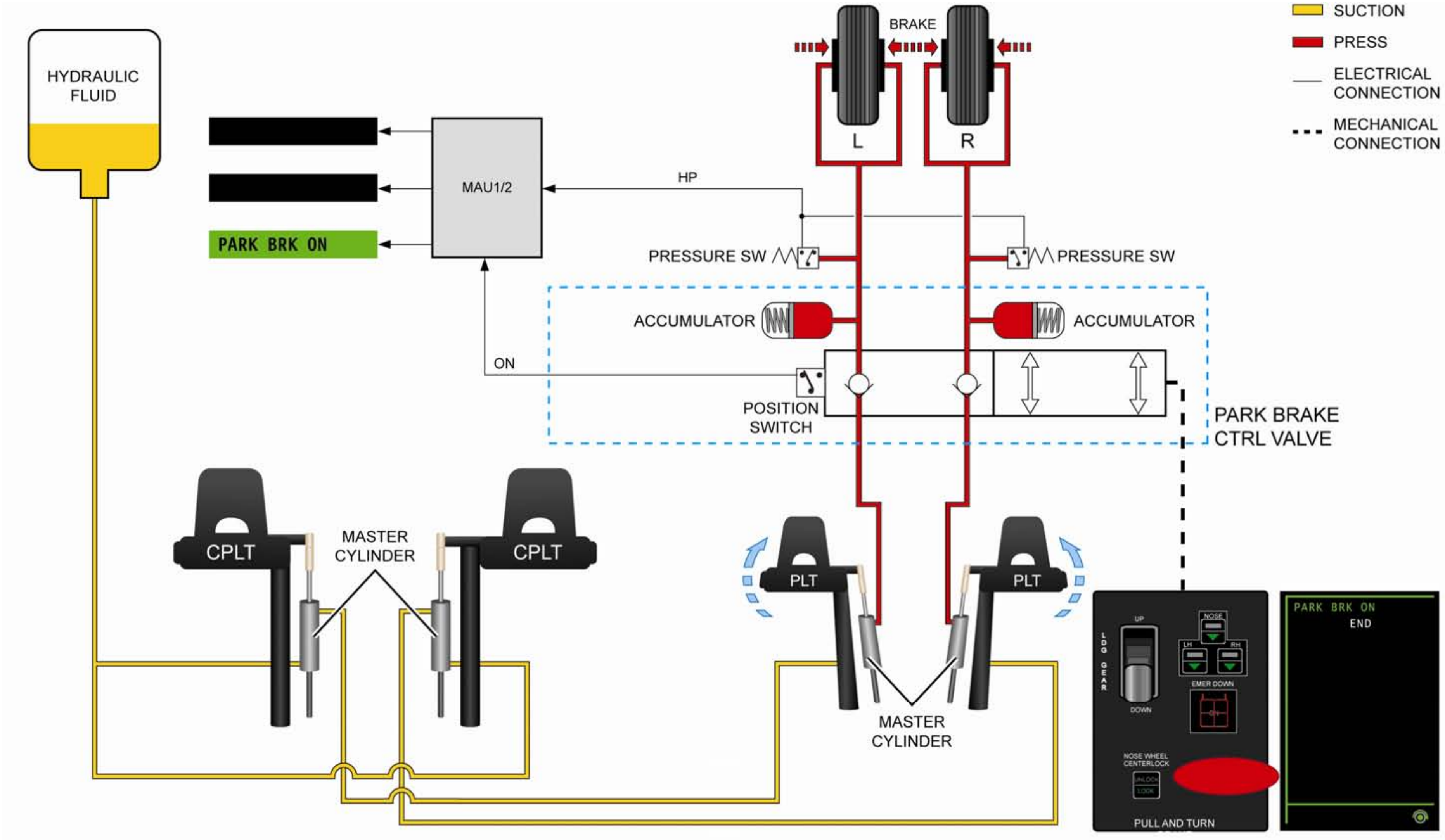
WHEEL BRAKE SYSTEM – SCHEMATIC



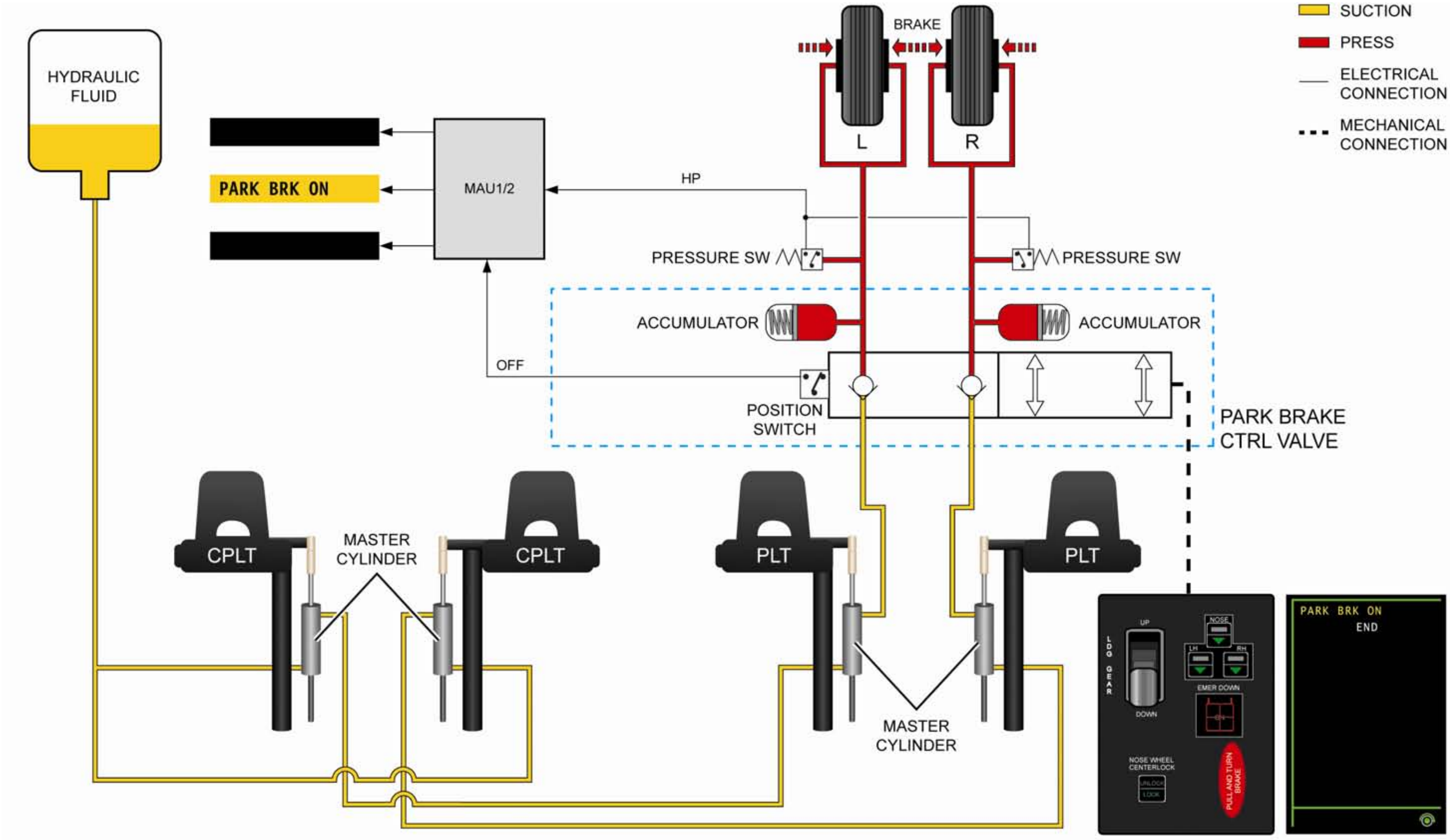
EXAMPLE OF NORMAL BRAKING (CPLT)



PARKING BRAKE (1 OF 2)



PARKING BRAKE (2 OF 2)



PARK BRAKE FAILURE

LANDING GEAR CAS CAUTION MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
LANDING GEAR	Landing gear retracted when aircraft height is less than 150 ft AGL + voice (only) message "150 FEET"	LANDING GEAR RETRACTED	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES MISCELLANEOUS

LANDING GEAR CAS ADVISORY MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
LDG EMER DOWN	Landing gear lowered using emergency down system		Section 2 NORMAL PROCEDURES

NOSE WHEEL STEERING AND CENTER LOCK SYSTEM CAS CAUTION MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
NOSE WHL UNLK	Nose wheel not locked in fore and aft direction NOTE. The landing gear retraction is inhibited when the NOSE WHL UNLK caution illuminated.	NOSE WHEEL UNLOCKED (IN FLIGHT)	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES HYDRAULICS

WHEEL BRAKE SYSTEM CAS CAUTION MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
PARK BRK PRESS	No pressure in park brake system with PARK BRAKE lever in ON position	PARK BRAKE MALFUNCTION	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES
PARK BRK ON	Park brake system pressurised	PARK BRAKE ON	MISCELLANEOUS

WHEEL BRAKE SYSTEM CAS ADVISORY MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
PARK BRK ON	Park brake handle is pulled and trapped pressure is above required value		Section 2 NORMAL PROCEDURES

CHAPTER
33
LIGHTING SYSTEM

SECTION 00 – GENERAL

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LIGHTS – GENERAL

The lighting system includes the following sub-systems:

- Interior lights
- Exterior lights
- Emergency lights

INTERIOR LIGHTS

Interior lights supply illumination for:

- cockpit instruments & control panels
- cockpit ambient and map reading
- anti storm, utility and dome lights
- cabin ambient and passenger reading lights
- baggage compartment

All interior lights can be NVG-compatible (optional kit); in that case a Mode selector switch is installed on the LT control panel.

DISPLAYS, INSTRUMENTS AND CONTROL PANELS

Display Units brightness is adjusted by four individual knobs on the DIM control panel.

Other cockpit instruments and control panels are provided with integral illumination controlled by three knobs on the LT control panel, according to their location area: Overhead Console, Instrument Panel or Central Console.
The LT control panel knobs are also provided with a switch and permit:

- turning on and off the integral lights
- adjustment of integral lights brightness
- selection of day/night mode for the annunciators

ANTI-STORM LIGHTS

Anti storm lights are halogen lights installed on the overhead panel. They are used to prevent pilot dazzling caused by storm lightning.

COCKPIT UTILITY LIGHTS

The cockpit utility lights consist of two lamps secured to supports located on the overhead console side walls, one on the right side (Pilot cockpit light) and one on the left side (Copilot cockpit light).
The supports are provided with a swiveling joint so that each cockpit utility light beam can be adjusted in direction.
Each light has a flexible cable, so that it can be removed from the support and handled by pilot or copilot.
Each cockpit utility light incorporates the controls for:

- turning the light on and off
- brightness adjustment

- color selection
- beamwidth adjustment

COCKPIT DOME LIGHT

The cockpit dome light provides flooding ambience light from the overhead console.
The cockpit dome light may consist of a standard white-light bulb or a solid-state white LED array (or combined white-and-green LED array in case of NVG-compatibility).
The cockpit dome light is controlled by a single knob on the LT control panel.

MAP READING LIGHTS (OPTIONAL)

In case optional Map Holders are installed on the instrument panel, dedicated lights and associated controls are also incorporated.

CABIN LIGHTS

In the standard helicopter (hard-liner ceiling), cabin lights consist of six (or eight) fluorescent tubes or solid-state LED arrays; in helicopters equipped with soft-liner cabin ceiling, cabin lights consist of four lamps or four solid-state LED arrays (or combined white-and-green LED arrays in case of NVG-compatibility).
Cabin lights are controlled by the pilot via a single knob on the LT control panel.

PASSENGER READING LIGHTS

In the standard helicopter (hard-liner ceiling), twelve passenger reading lights are installed, two per PSU (Passenger Service Unit).
Each passenger can turn his own light on and off via a dedicated push-button on the PSU.
Pilots have no control on the passenger reading lights.

BAGGAGE COMPARTMENT LIGHTS

Three dome lights provide illumination of the baggage compartment.
Each baggage compartment light may consist of a standard white-light bulb or a solid-state white LED array (or combined white-and-green LED array in case of NVG-compatibility).
The baggage compartment lights are automatically activated when any of the baggage doors is open and power is available on MAIN 2 BUS.

INTERIOR LIGHTS

Provide illumination for:

- **Cockpit Instruments & Control Panels**
- **Cockpit ambient & Map reading**
- **Anti storm**
- **Cabin ambient and Passenger reading**
- **Baggage Compartment**

EXTERIOR LIGHTS

Provide illumination for:

- **Anti Collision**
- **Navigation**
- **Landing**

EMERGENCY LIGHTS

Provide illumination for:

- **Emergency Egress**

INTERIOR LIGHTS – CONTROLS

The DIM control panel allows individual brightness adjustment of the Display Units.

1. CPLT PFD knob
   Rotated ………… adjusts the brightness of the copilot PFD from minimum (MIN) to maximum (MAX)

2. CPLT MFD knob
   Rotated ………… adjusts the brightness of the copilot MFD from minimum (MIN) to maximum (MAX)

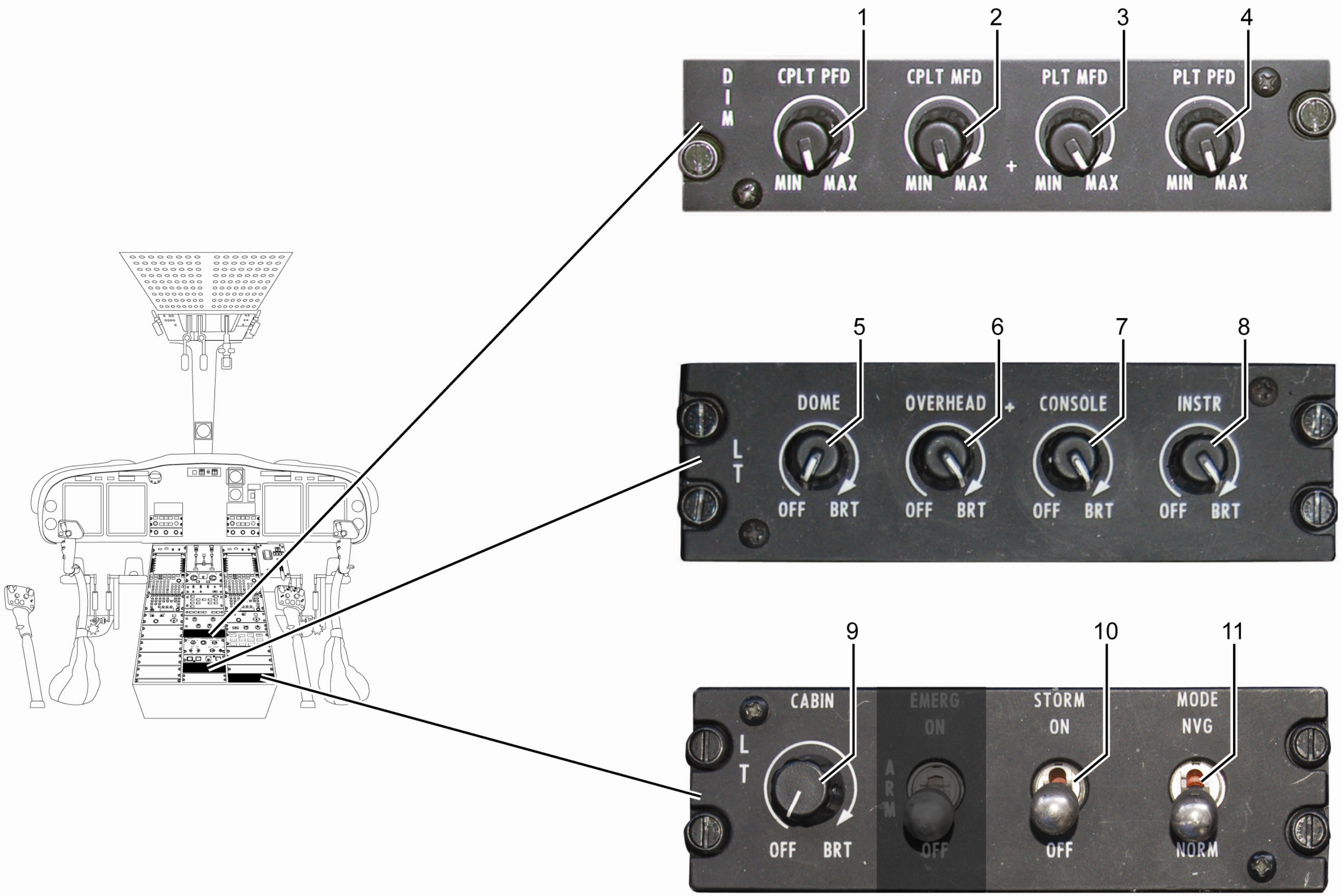
3. CPLT MFD knob
   Rotated ………… adjusts the brightness of the pilot MFD from minimum (MIN) to maximum (MAX)

4. PLT PFD knob
   Rotated ………… adjusts the brightness of the pilot PFD from minimum (MIN) to maximum (MAX)

The LT control panel allows control and brightness adjustment of different cockpit lights.

5. DOME switch/knob
   OFF ..…………… (switch actuated) turns off the cockpit dome light
   Rotated ………… adjusts the dome light intensity from min to max

6. OVERHEAD knob
   OFF ..…………… (switch actuated) turns off the overhead console lights and sets the overhead panel annunciators at full brightness (day mode)
   Rotated ………… adjusts the light intensity of the overhead console lights and annunciators from min to max



INTERIOR LIGHTS CONTROLS

7. CONSOLE knob

OFF …..…………	(switch actuated) turns off the central console lights and sets the central console annunciators at full brightness (day mode)
Rotated …………	adjusts the light intensity of the central console lights and annunciators from min to max

8. INSTR knob

OFF …..…………	(switch actuated) turns off the instrument panel lights and sets the instrument panel annunciators at full brightness (day mode)
Rotated …………	adjusts the light intensity of the instrument panel lights and annunciators from min to max

The LT control panel allows control and brightness adjustment of the cabin lights, control of the emergency and the storm lights.

9. CABIN knob

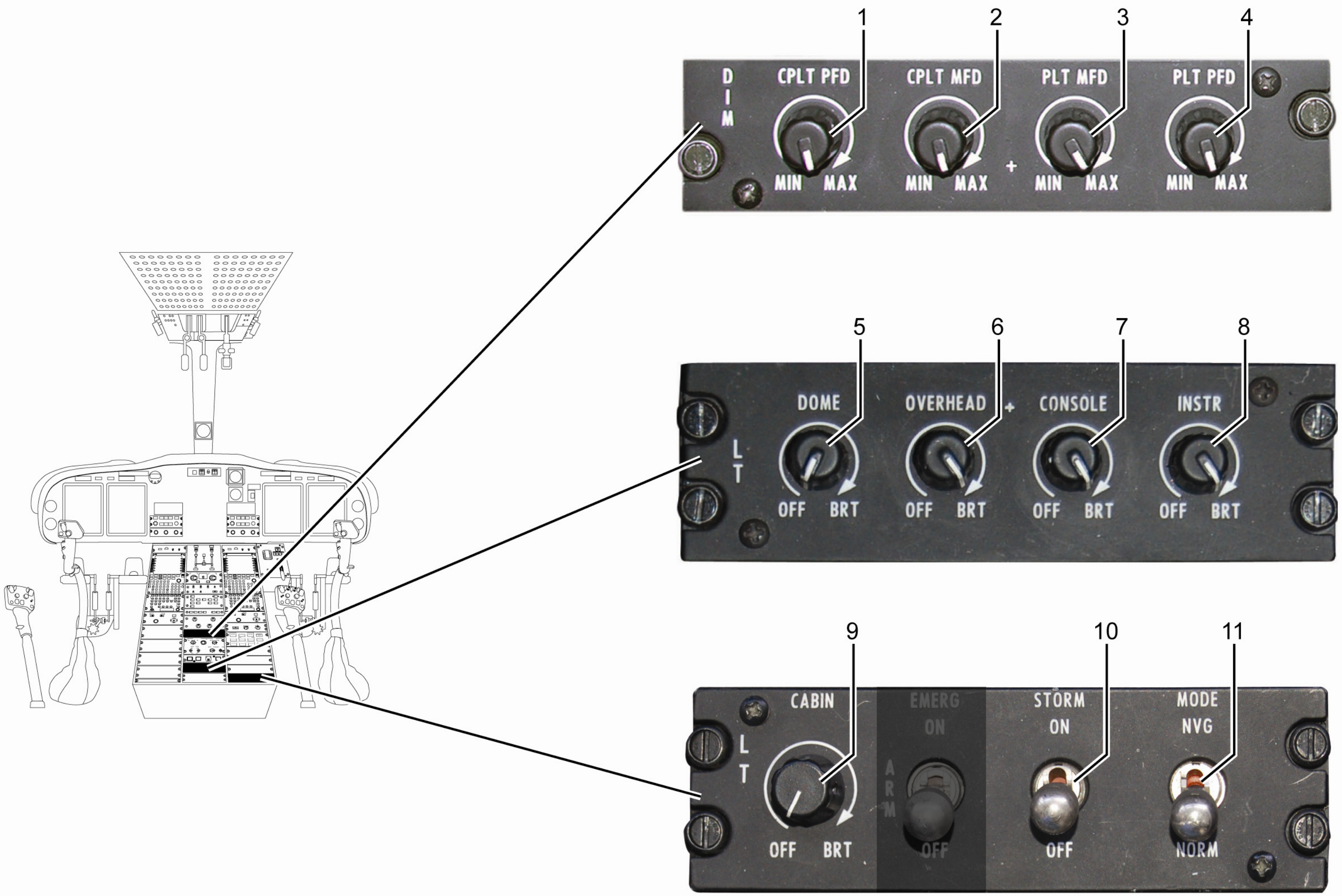
OFF …..…………	(switch actuated) turns off the cabin lights
Rotated …….……	adjusts the light intensity of the cabin lights from min to max

10. STORM switch

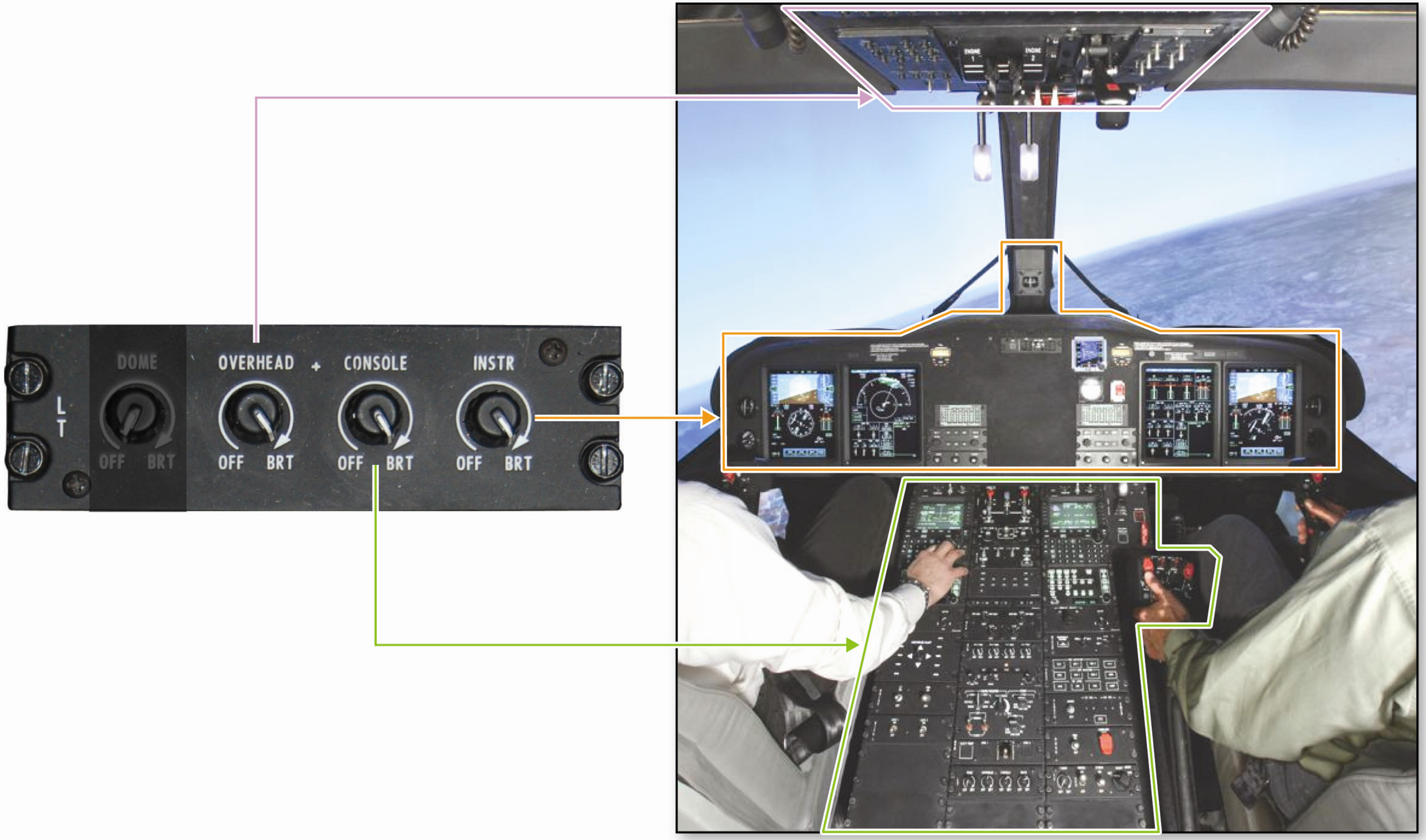
OFF ………..…….	storm lights are off
ON ……….………	storm lights are illuminated

11. MODE switch (optional)

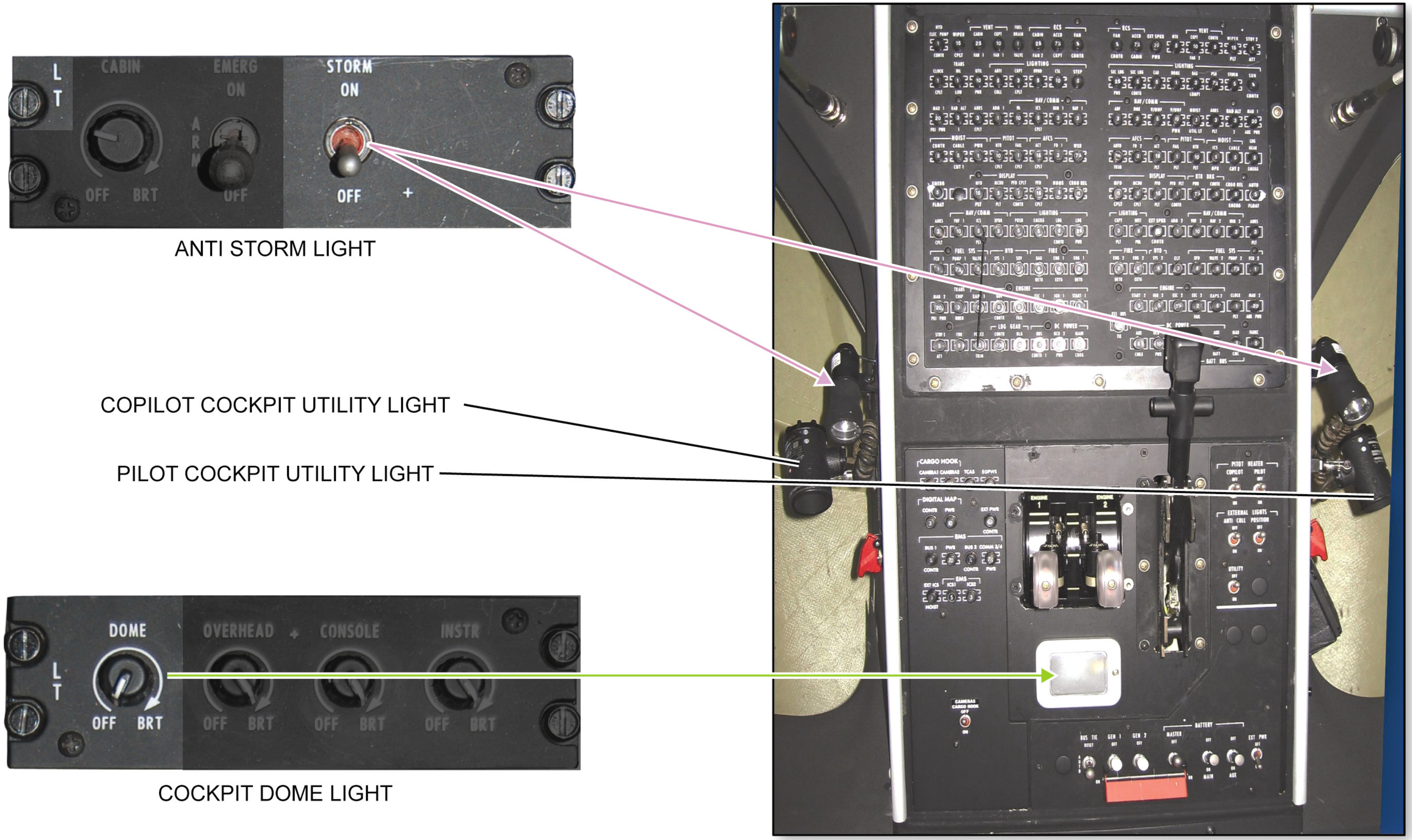
NORM …..………	Interior lights and displays illuminate with standard visible light
NVG …..…………	Interior lights and displays illuminate with NVG-compatible light



INTERIOR LIGHTS CONTROLS



COCKPIT LIGHTS (1 OF 2)



COCKPIT LIGHTS (2 OF 2)

COCKPIT UTILITY LIGHTS controls

12. Push-button

Pressed ………… Momentarily illuminates utility light at full brightness for as long as it is held pressed

13. OFF/DIM/BRIGHT knob

OFF …..………… (switch actuated) turns off the utility light

Rotated ………… adjusts the light intensity of the utility light from minimum (DIM) to maximum (BRIGHT)

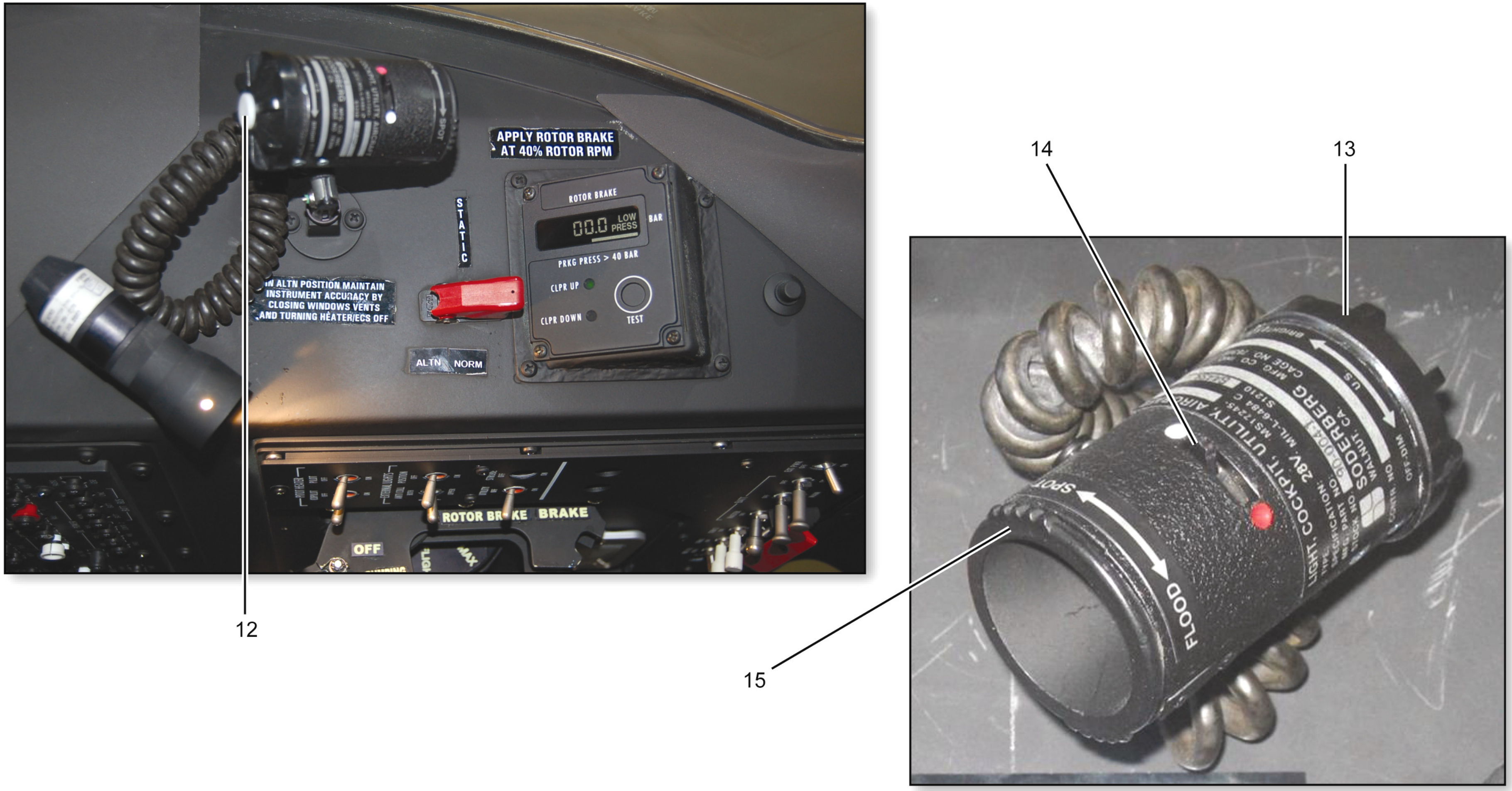
14. Color selector

White dot ………. Utility light color is white (no filtering)

Red dot …………. Utility light color is red (red filter mechanically set in front of the bulb)

15. FLOOD/SPOT ring

Rotated ………… adjusts the utility light beamwidth from minimum (SPOT) to maximum (FLOOD)



COCKPIT UTILITY LIGHT (TYPICAL)

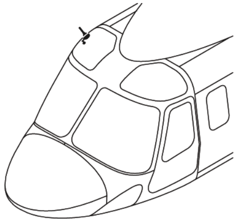
OAT INDICATOR LIGHT controls.

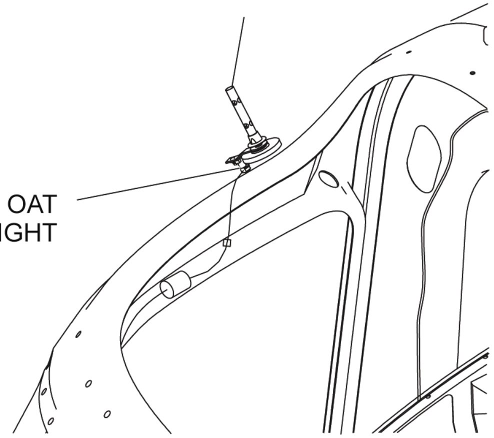
16. OAT LIGHT push-button

Pressed .... If the OVERHEAD knob (see 6.) is not set to OFF, momentarily illuminates the Standby OAT light for as long as it is held pressed

NOTE: The OAT indicator light illuminates only if the OVERHEAD rotating knob on the LT panel is set in a position different from OFF.

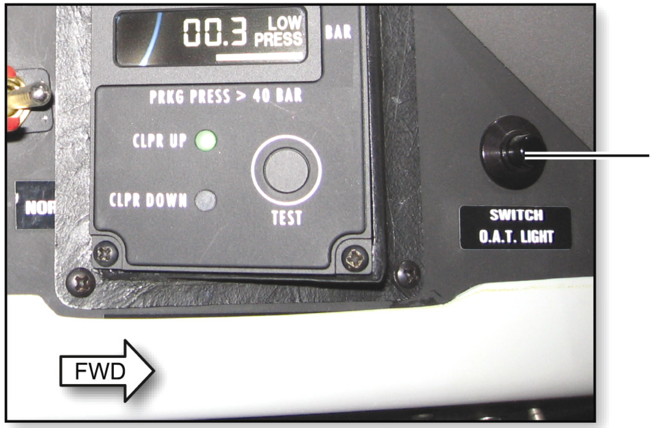
OAT INDICATOR LIGHT





External view of RH Pilot top window

NOTE. OAT INDICATOR LIGHT ILLUMINATES ONLY
IF THE OVERHEAD KNOB IS NOT OFF



View of forward area of RH Overhead Console side wall

OAT INDICATOR LIGHT

MAP HOLDERS AND MAP READING LIGHTS (OPTIONAL)

CABIN LIGHTS

PASSENGER SERVICE UNIT (PSU) controls and indicators

17. Passenger Reading Light pushbuttons

Pressed ……………………… the relevant reading light on

18. Passenger Warning Lights

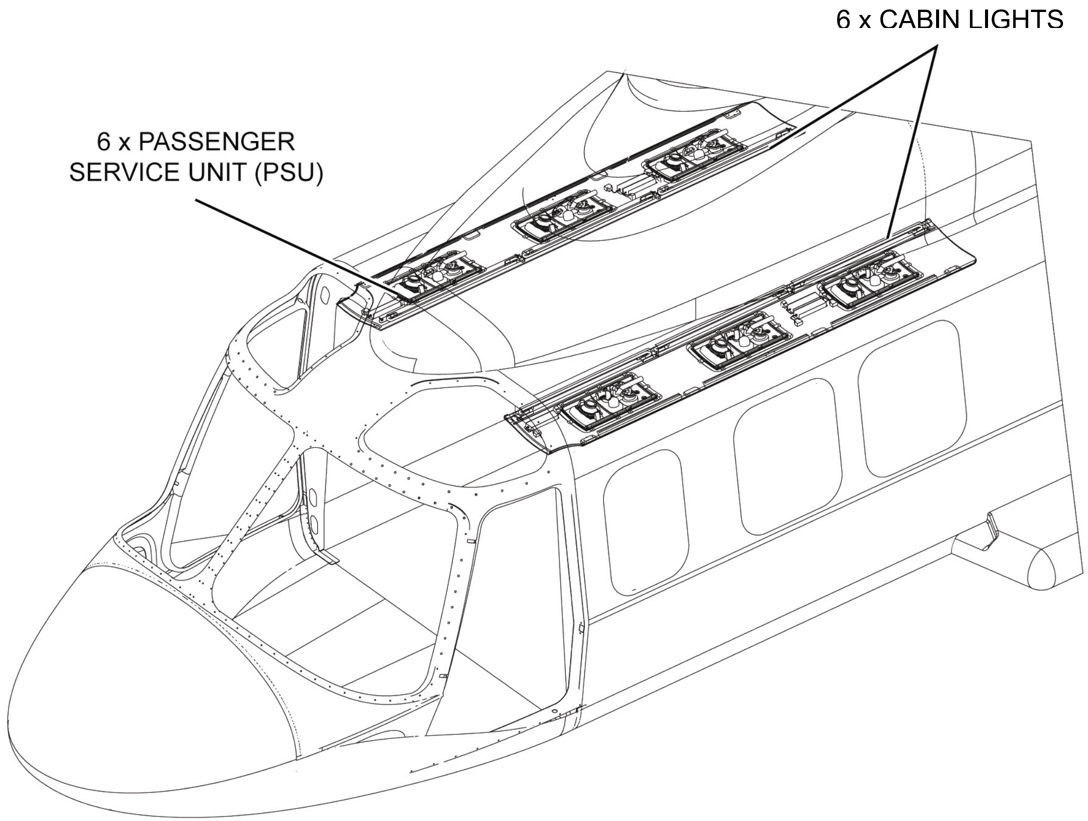
No Smoking icon ……………. Illuminates when the pilot (or the copilot) activates the CHM1 or NO SMK pushbutton on the audio panel

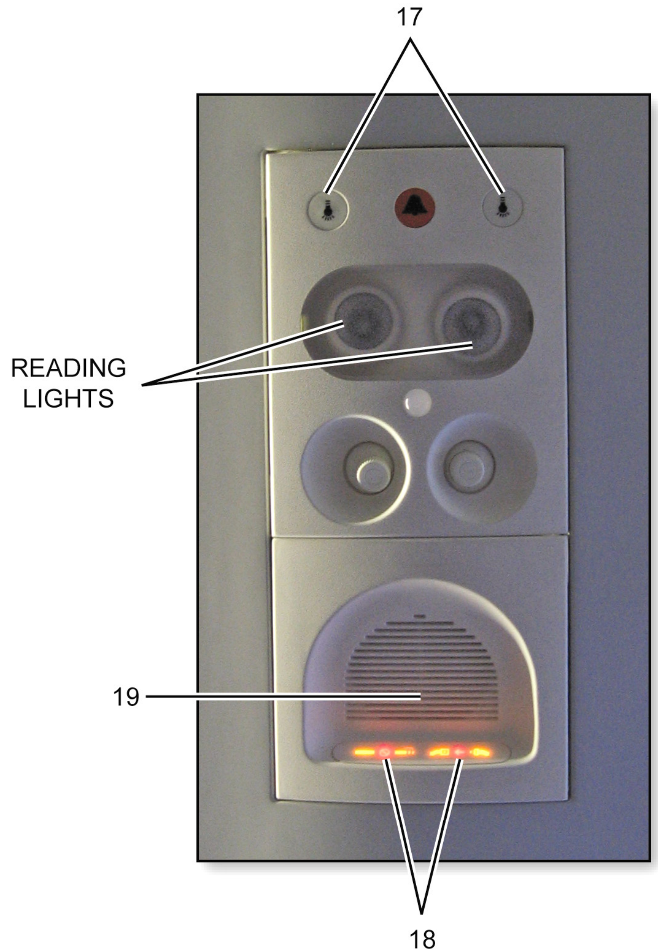
Fasten Seat Belt icon ………. Illuminates when the pilot (or the copilot) activates the CHM2 or SEAT BLT pushbutton on the audio panel

NOTE: Passenger Warning lights are controller via the Audio control panel (see chapter 23-00)

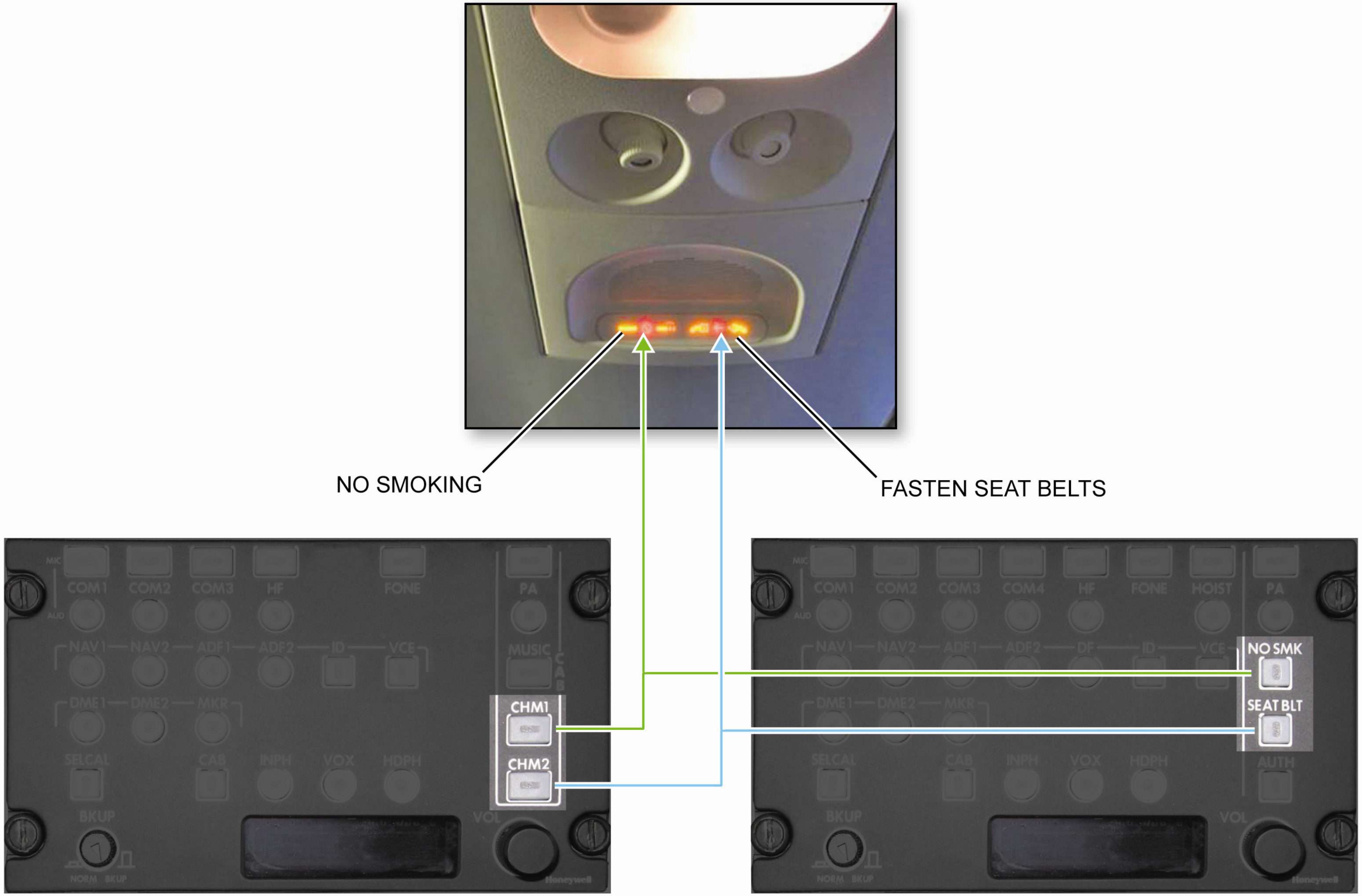
19.

Loudspeaker …………………. allows a chime to be heard when any Passenger Warning light is turned on or off.





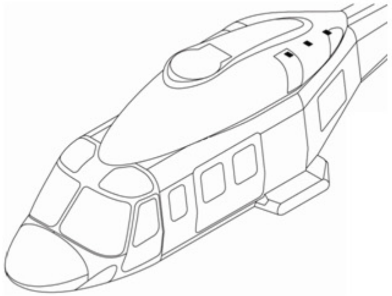
PASSENGER SERVICE UNIT (PSU) LIGHTS



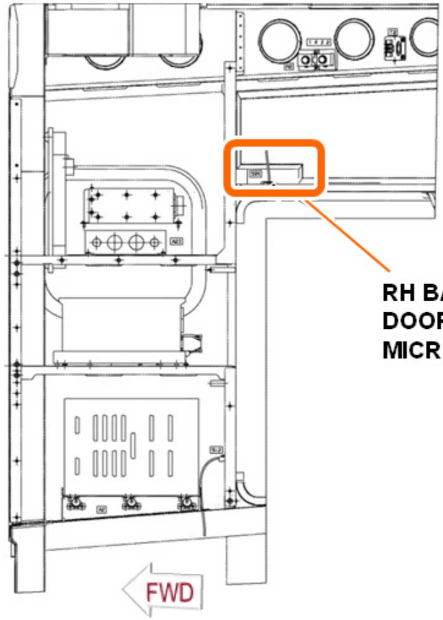
STANDARD AUDIO CONTROL PANEL (PLT OR CPLT)

EMS AUDIO CONTROL PANEL (PLT, CPLT, HOIST OP)

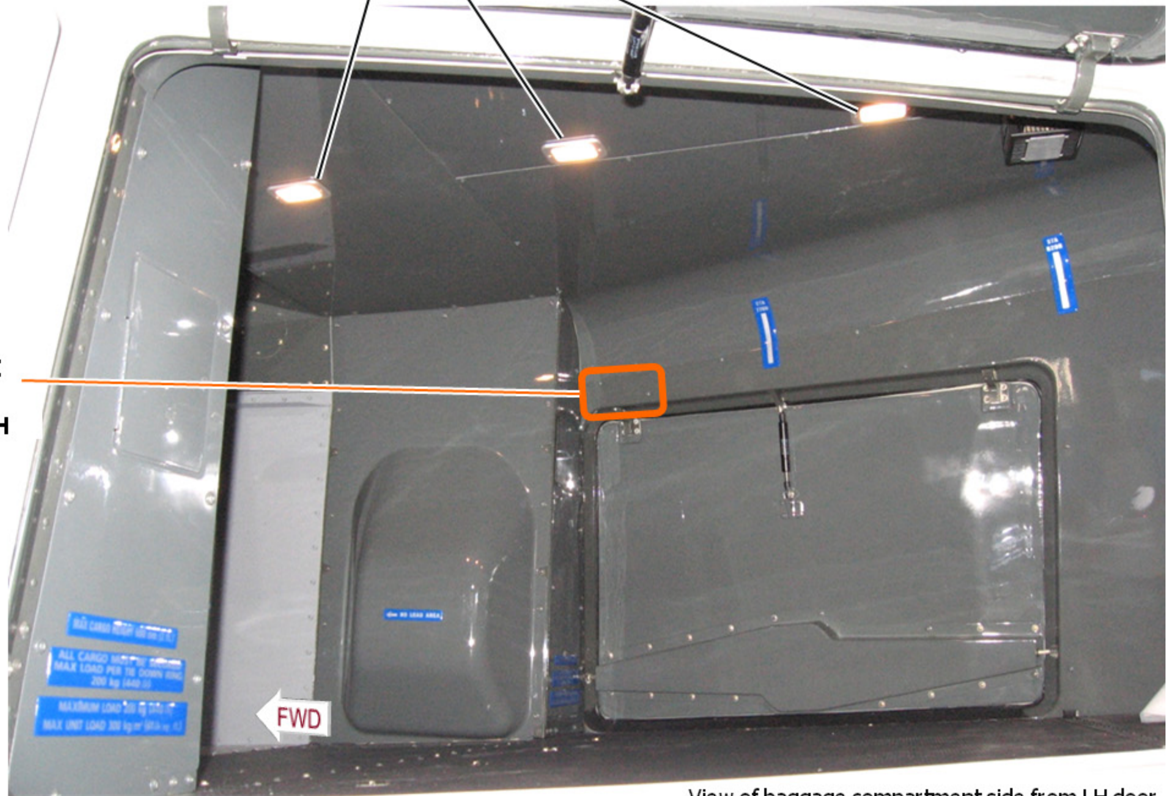
PASSENGER WARNING LIGHTS



View of RH baggage compartment side from interior



NOTE: LH Baggage Door Microswitch is installed symmetrically



View of baggage compartment side from LH door

BAGGAGE COMPARTMENT LIGHTS

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EXTERIOR LIGHTS – GENERAL

The exterior lights comprise:

- anti-collision light
- position lights
- landing lights
- secondary landing lights

All exterior lights can be NVG-compatible (optional kit); in this case a dedicated control panel is installed on the central console.

Other optional exterior lights include:

- second anti-collision light
- strobe lights
- step lights
- rotor lights

ANTI-COLLISION LIGHT(S)

The anti-collision light is a strobe red light that allows the visibility of the helicopter at great distances.
The anti-collision light is installed at the top of the vertical fin.
An optional additional anti-collision light can be installed under the aft fuselage.
The anti-collision light is controlled by the ANTI COLL switch on the overhead console.

POSITION LIGHTS

The position lights provide flight direction information and consist of three lights:

- the tail white position light is installed on the top of the tail fin
- the right green position light is installed either on the RH horizontal stabilizer wingtip or on the RH sponson
- the left red position light is installed either on the LH horizontal stabilizer wingtip or on the LH sponson

The position lights are controlled by the POSITION switch on the overhead console.

LANDING LIGHTS

Two landing lights provide a high-intensity light source suitable for landing and taxi operations in the night.
Landing lights are installed on the right and left sponsons leading edge, inboard position.
The landing lights are controlled by the LDG LT switch on pilot and copilot collective levers; either pilot or copilot can turn on or off the two landing lights.
When the landing lights are illuminated, the LANDING LT ON advisory message is displayed in the CAS window of the MFD.

SECONDARY LANDING LIGHT (SEARCH LIGHT)

The secondary landing light has been designed to provide a mobile and versatile high-intensity light source for landing at night outside airports or helipads and for aerial reconnaissance at night time.
It is installed under the cockpit (forward LH side of bottom fuselage).
The secondary landing light is controlled by the LDG LT2 five-way switch on pilot and copilot collective levers and can be extended up to 120° from the fully retracted position and rotated 360° in either direction.
When the secondary landing light is illuminated, the SEARCH LT ON advisory message is displayed in the CAS window.
Turning the secondary landing light off also causes the automatic stowing of the light.

STROBE LIGHTS (OPTIONAL)

Strobe lights provide high-intensity flashing white light on both helicopter sides and are typically incorporated with the navigation light assemblies.
Strobe lights are controlled by a switch on the overhead console.

STEP LIGHTS (OPTIONAL)

Step lights are installed together with the electrically-retractable step (see chapter 25-00) and provide illumination of the external stepping area for passengers.
The step light is controlled by the UTILITY switch on the overhead console.

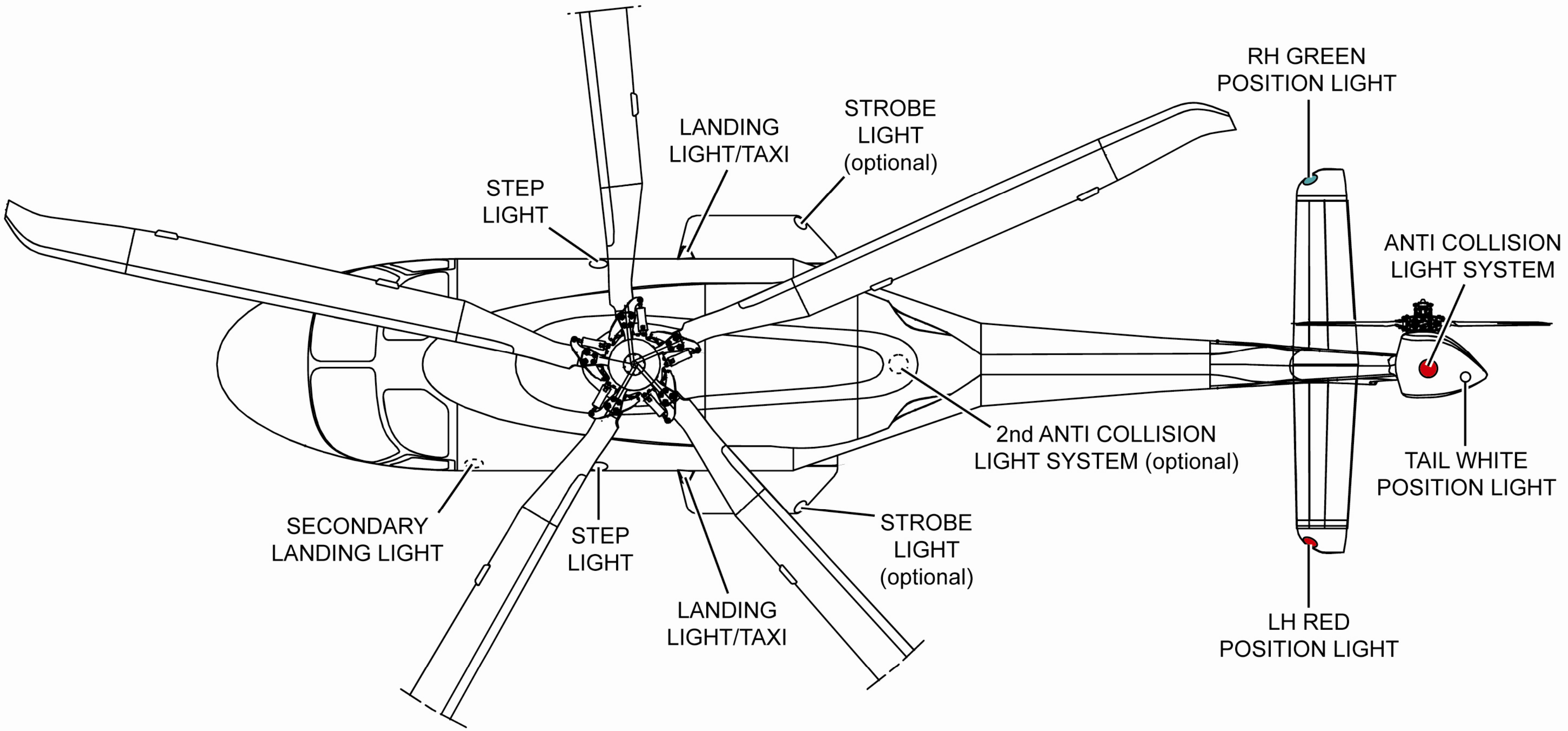
ROTOR LIGHTS (OPTIONAL)

The rotor lights comprise two main rotor lights, two tail rotor lights and the ROTOR LTS switch on the Auxiliary control panel.
The rotor lights installation is intended to illuminate two portions of the main rotor (right side and left side) and the tail rotor. These lights allow the crew to identify, during night time ground and hover condition, the extremity of the main and tail rotor discs.

The ROTOR LTS switch is a three position switch labelled OFF / MAIN / BOTH.

When set to MAIN the main rotor lights are illuminated; when set to BOTH the main and the tail rotors lights are illuminated; when set to OFF the rotor lights are switched off.

The rotor lights are not NVG compatible. If NVG installation is present, then in the NVG COVERT mode the main and tail rotor lights are automatically selected to OFF.



EXTERIOR LIGHTS

EXTERIOR LIGHTS – CONTROLS

1. EXTERNAL LIGHTS – ANTI COLL switch

   OFF …... anti-collision light is off

   ON …….. anti-collision light is on

2. EXTERNAL LIGHTS – POSITION switch

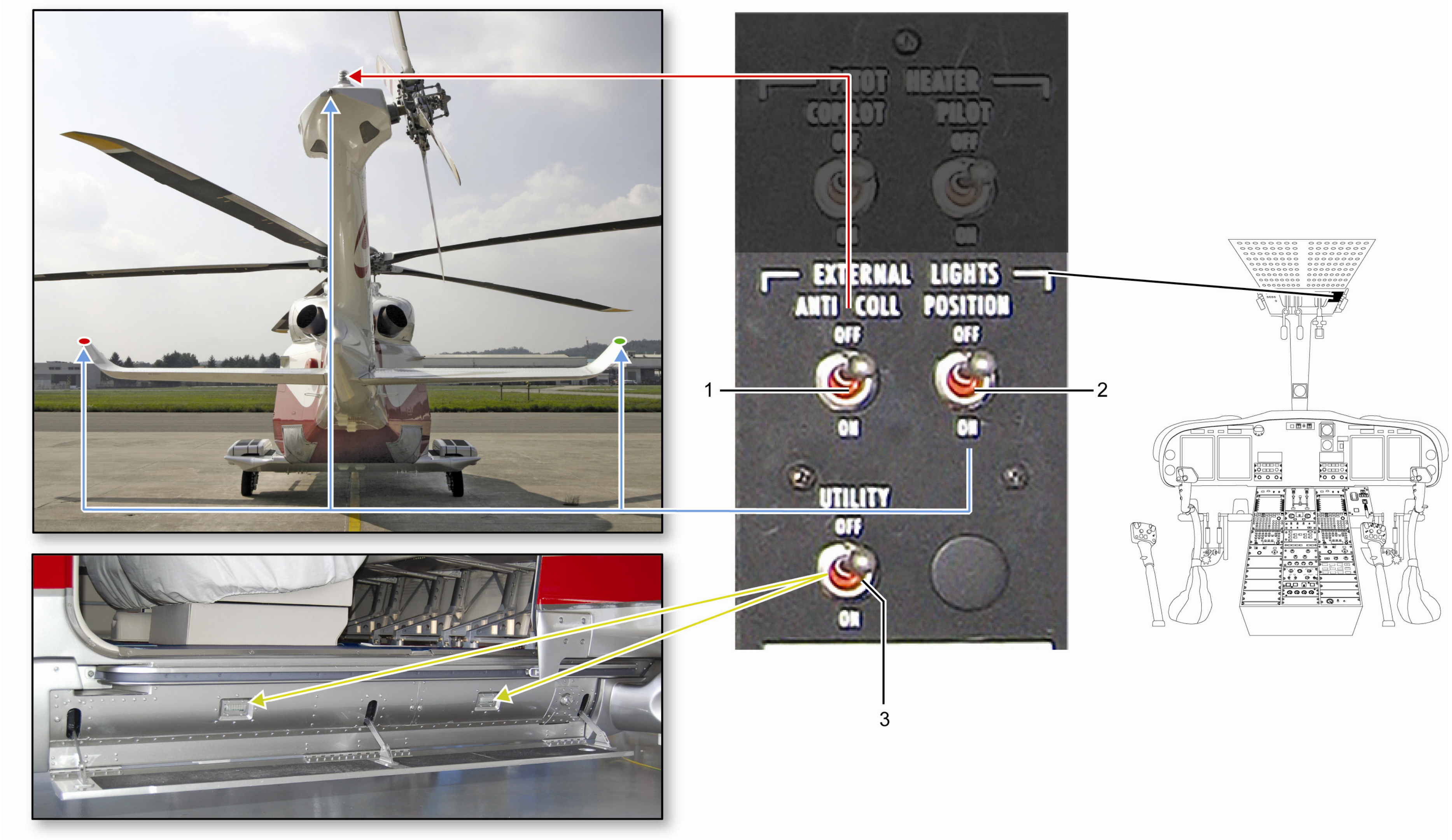
   OFF …... position lights are off

   ON ……. position lights are on

3. EXTERNAL LIGHTS – UTILITY switch

   OFF …... step lights are off

   ON ……. step lights are on



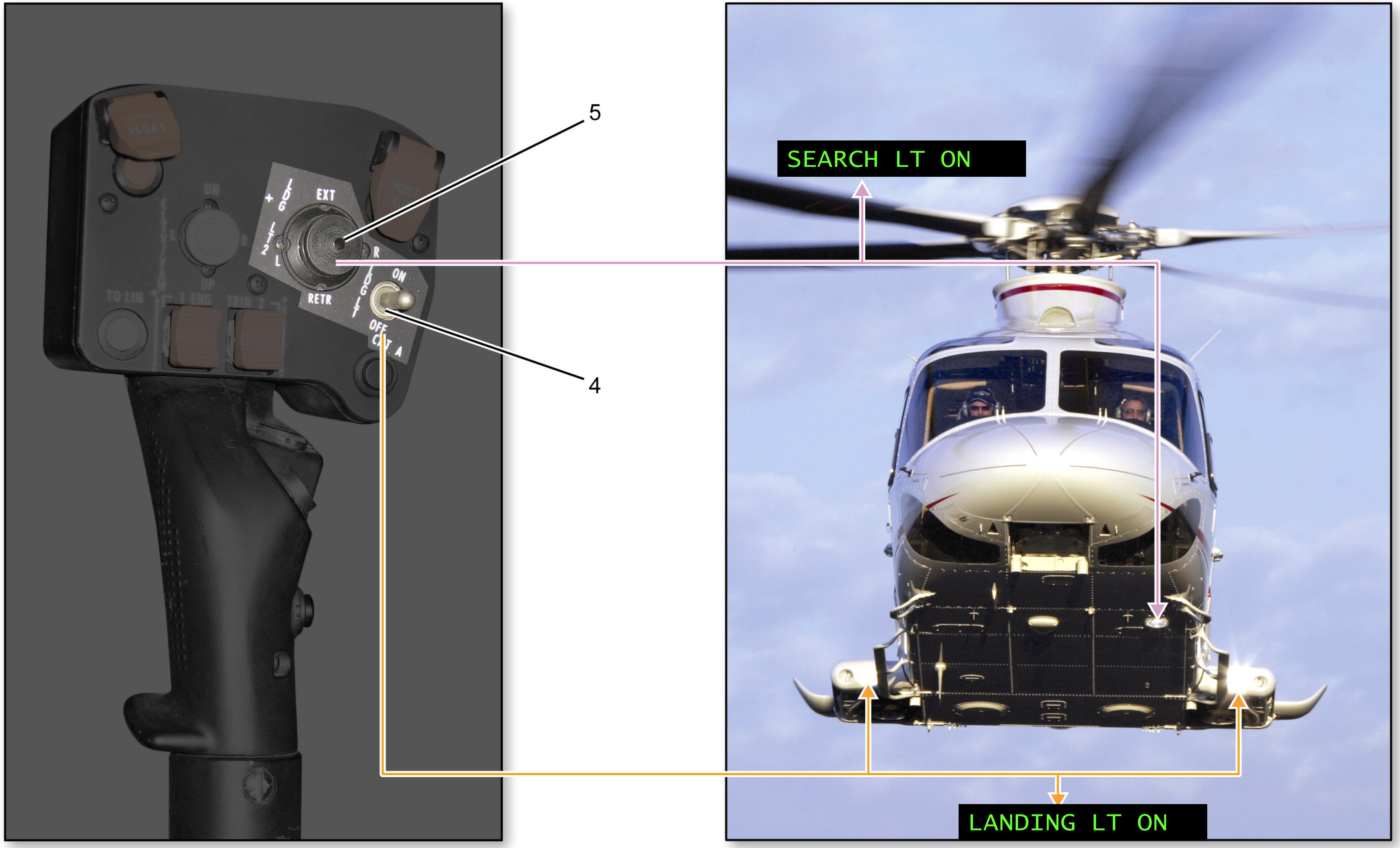
EXTERIOR LIGHTS – ANTI-COLLISION, POSITION AND UTILITY (STEP) LIGHTS CONTROLS

4. LDG LT spring loaded momentary toggle switch

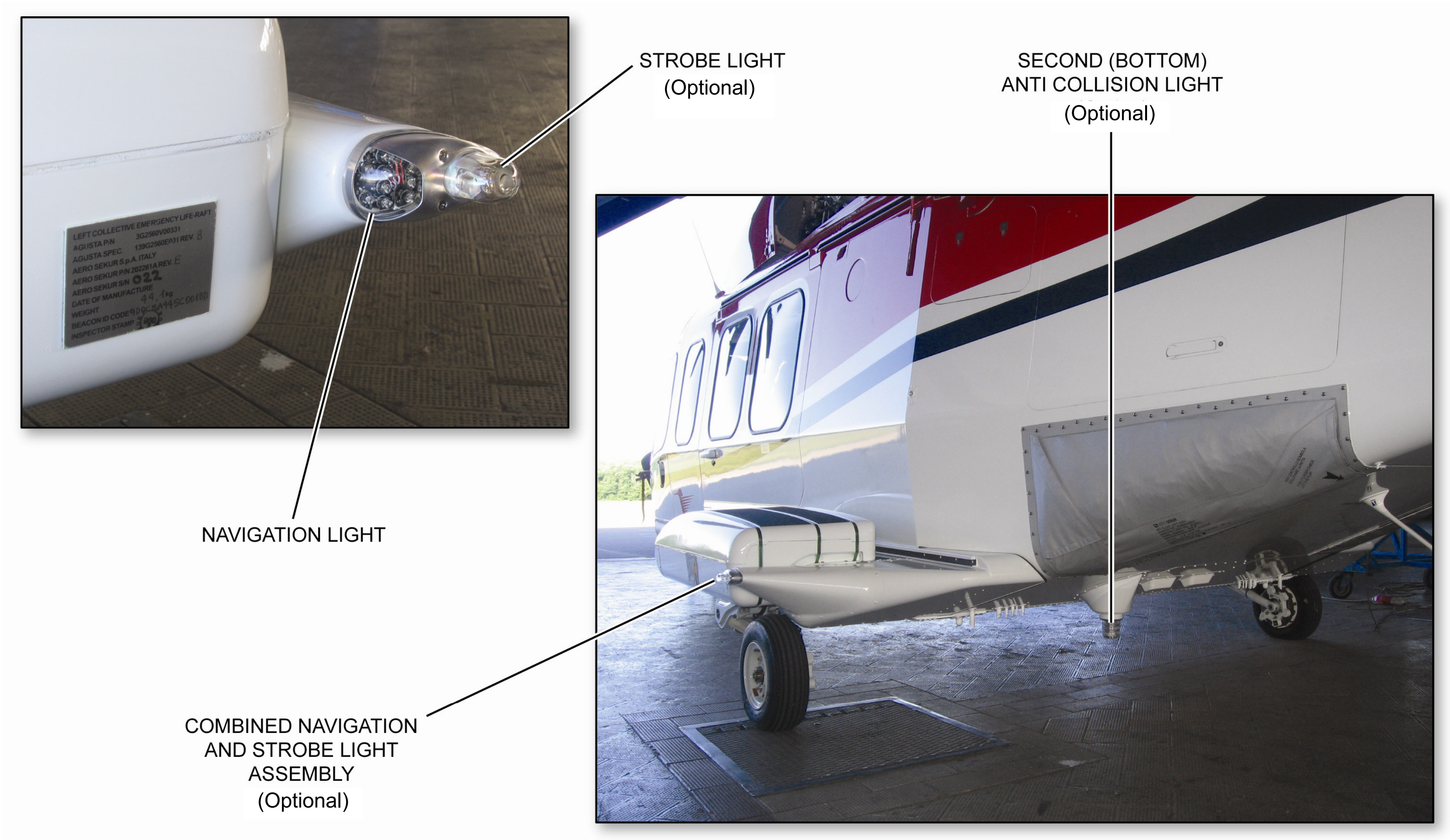
OFF …………….	allows to light off the two lamps
central position ...	inoperative
ON ……………….	allows to light on the two lamps

5. LDG LT2 four way momentary switch and central momentary contact

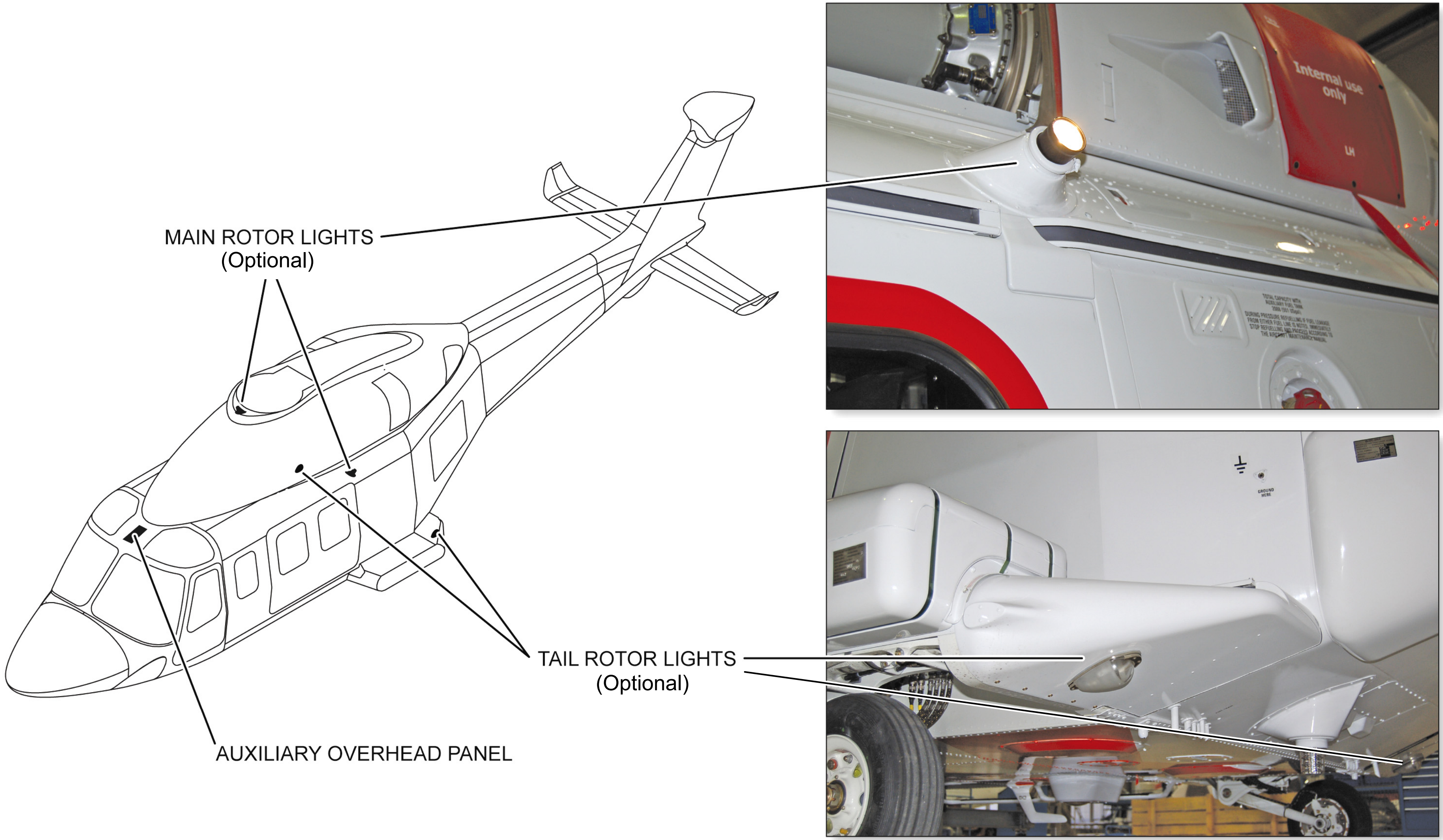
pressed with searchlight lamp extinguished	turns the searchlight lamp on
pressed with searchlight lamp illuminated	turns the searchlight lamp off and stows the searchlight
EXT	allows to extend the searchlight if the lamp is ON only
RETR	allows to retract the searchlight
L	allows to rotate the searchlight to the left
R	allows to rotate the searchlight to the right



EXTERIOR LIGHTS – LANDING LIGHTS CONTROLS



OPTIONAL EXTERIOR LIGHTS (1 OF 2)



OPTIONAL EXTERIOR LIGHTS (2 OF 2)

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EMERGENCY LIGHTS

The emergency lights provide illumination for emergency egress at night and comprise:

- two internal emergency dome lights
- two external emergency lights
- two battery packs

The internal emergency dome lights are installed in the center of the cabin ceiling to provide general lighting in the passenger cabin.
The external emergency lights are installed on the right and left sponsons leading edge, outboard position, to illuminate the ground surface.

All emergency lights are supplied by two rechargeable battery packs installed in the nose compartment.
The emergency lights are controlled by the EMERG switch on the LT control panel on the central console and by a push-button switch located in the passenger area.
The emergency light battery packs are recharged when the EMERG switch is in OFF or ARM and the ESS 1 bus is powered.
With the EMERG switch set to ARM, loss of power on the ESS 1 bus causes the automatic illumination of the emergency lights.

The emergency lights can be manually turned on either by the pilot (EMERG switch at ON) or by the passengers (EMERG LIGHT push-button in the cabin).

EMERGENCY LIGHTS – CONTROLS

1. EMERG switch

   OFF ………...…... emergency lights are OFF and the two batteries are recharging

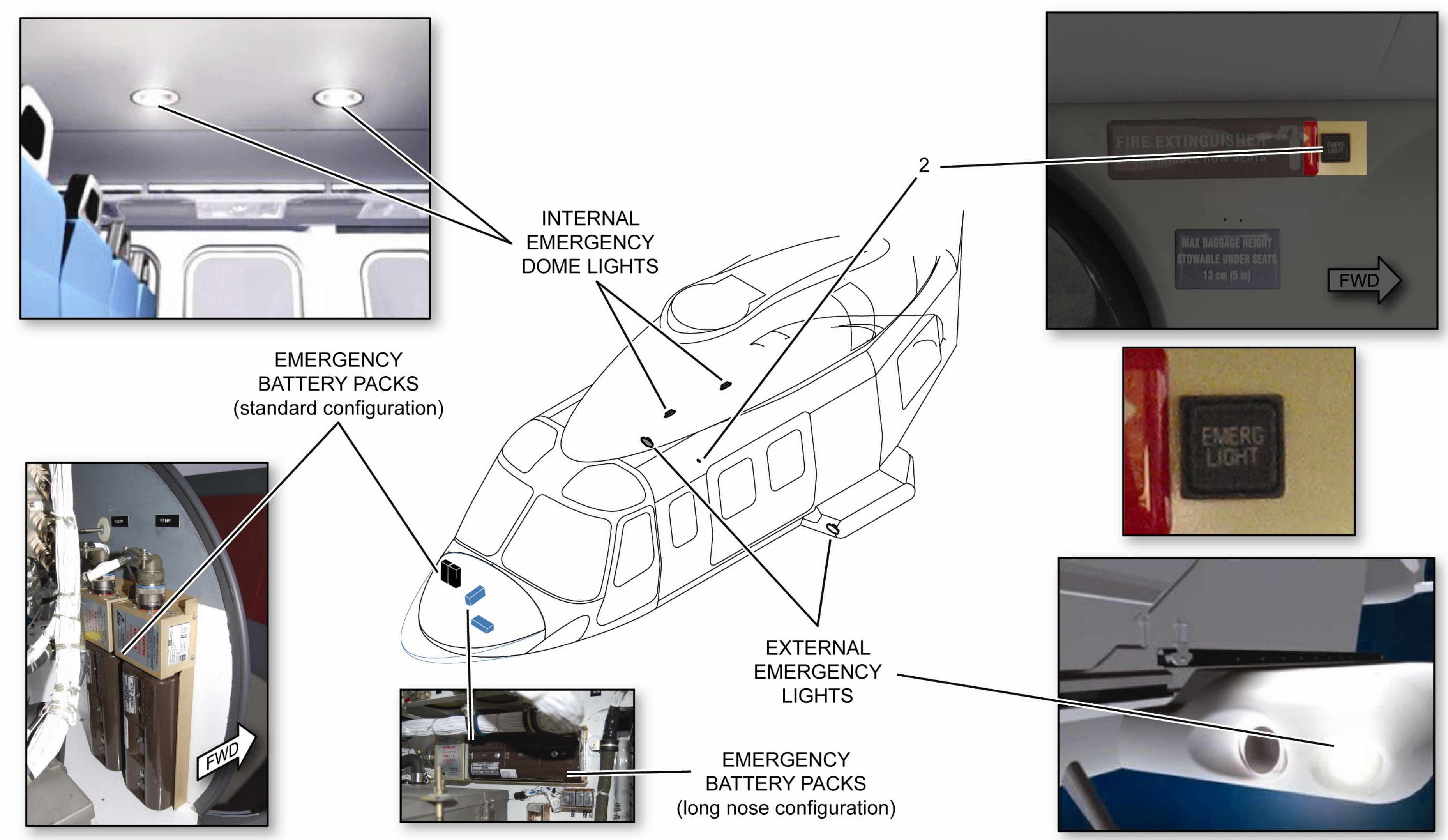
   ARM …..………… emergency lights are OFF and the two batteries are recharging; emergency lights will illuminate automatically upon loss of electrical power from ESS 1 bus

   ON ….…………… emergency lights are illuminated

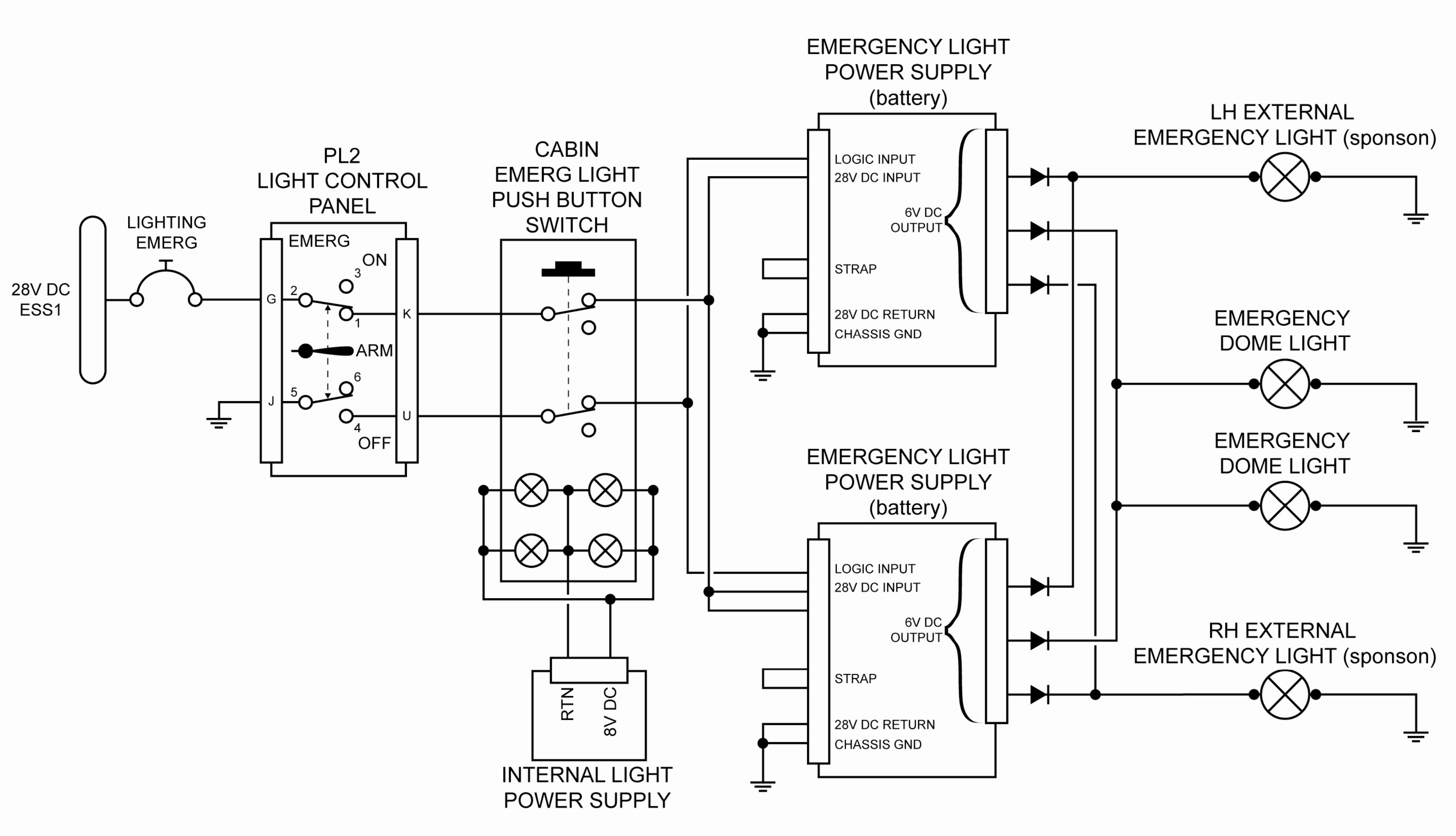
2. EMERG LIGHTS push-button switch (in the cabin)

   Pressed ………… emergency lights are illuminated

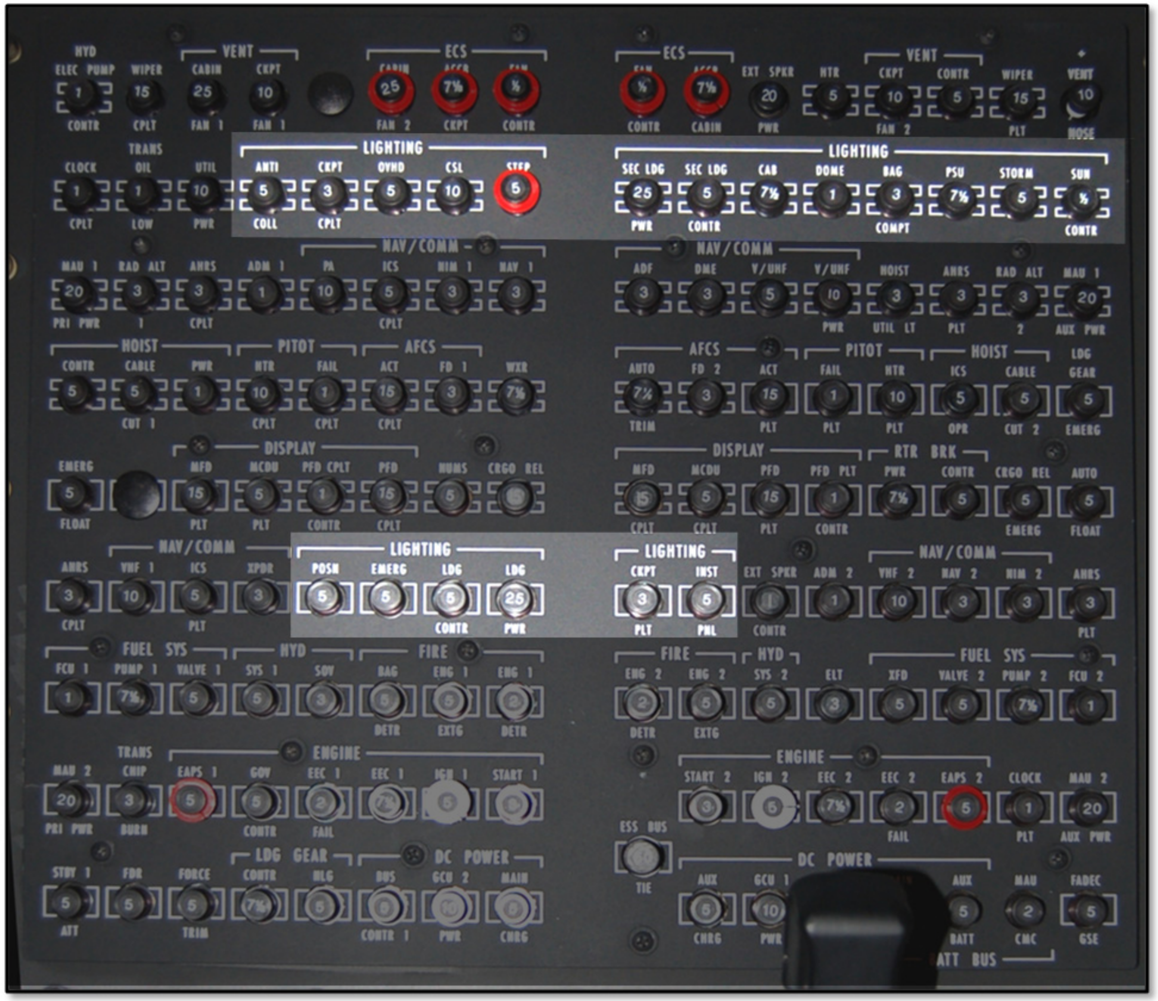
EMERGENCY LIGHTS – CONTROLS (1 OF 2)



EMERGENCY LIGHTS – CONTROLS (2 OF 2)



EMERGENCY LIGHTS SCHEMATIC DIAGRAM



CIRCUIT BREAKER PANEL (OVERHEAD CONSOLE)

LIGHTING CIRCUIT BREAKERS

LIGHTING SYSTEM – CAS ADVISORY MESSAGES

CAS CAPTION	MESSAGE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
LANDING LT ON	LDG LT switched ON		Section 2 NORMAL PROCEDURES
SEARCH LT ON	LDG LT2 switched ON		

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CHAPTER
34
NAVIGATION SYSTEMS

SECTION 00 - GENERAL

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NAVIGATION SYSTEMS – GENERAL

The navigation system is integrated in the PRIMUS EPIC® avionic system and includes the following sub-systems:

- FLIGHT ENVIRONMENTAL DATA

  The system includes two Air Data Systems (ADS 1 and ADS 2) to provide airspeed, barometric altitude, vertical speed and outside air temperature.

- ATTITUDE AND DIRECTION

  The system includes two Attitude and Heading Reference Systems (AHRS 1 and AHRS 2) to provide attitude and heading reference data.

- LANDING AND TAXIING AIDS

  The system includes:

  - One or two Radio Altimeter (RAD ALT) systems
  - Two VOR/ILS/MB (VHF-NAV) system

- INDEPENDENT POSITION DETERMINING (optional)

  The system can determine the helicopter position without using ground stations and/or orbital satellites. The Independent Position Determining system may include:

  - Weather Radar System (WX)
  - Lightning Sensor System (LSS)
  - Enhanced Ground Proximity Warning System (EGPWS)

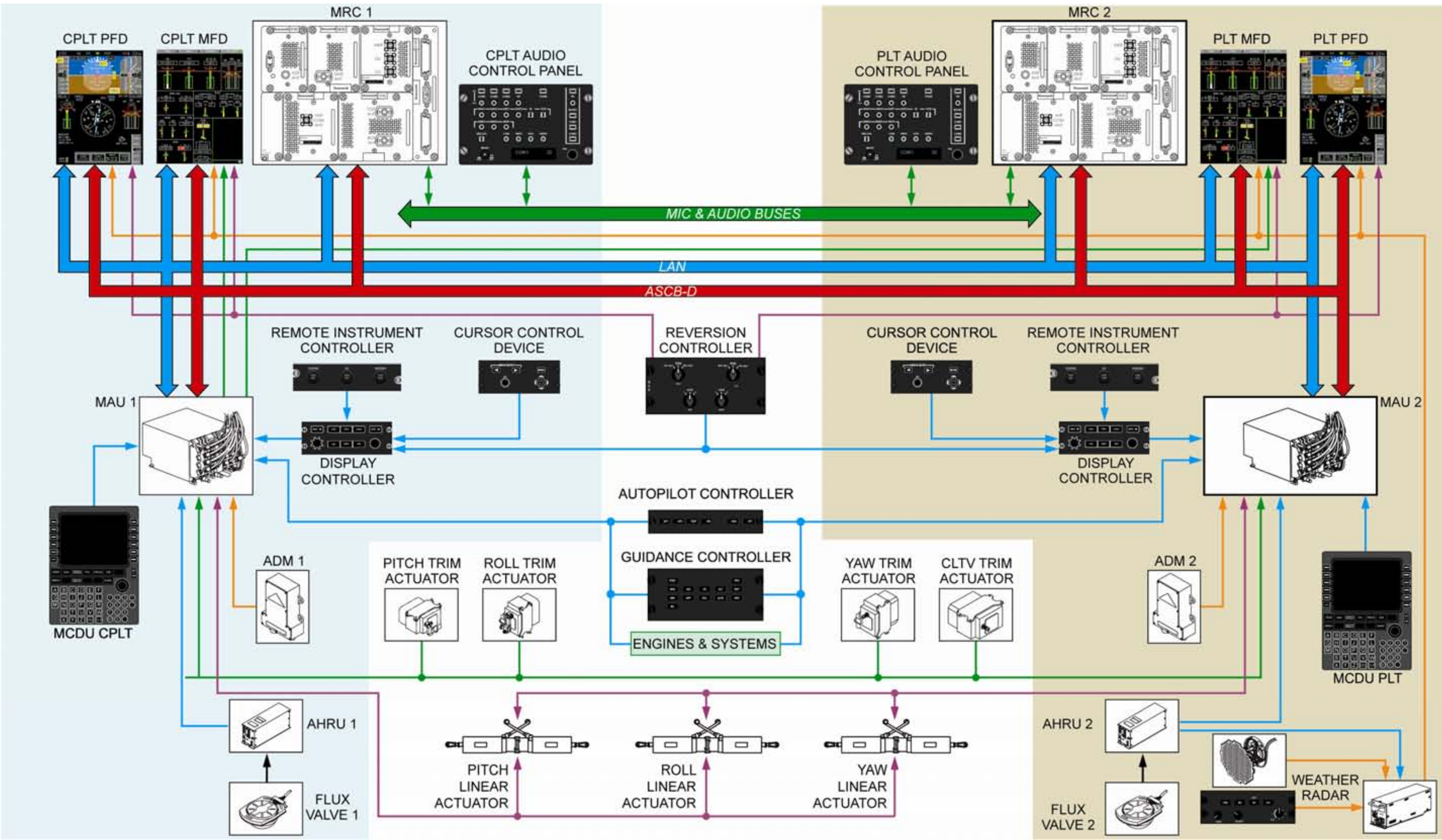
- DEPENDANT POSITION DETERMINING

  The Dependant Position Determining system uses ground stations and/or orbital satellites to find the helicopter position and velocity. The Dependant Position Determining system includes:

  - One or two Distance Measuring Equipment (DME)
  - One or two Air Traffic Control (ATC) Transponder (XPDR)
  - One or two Automatic Direction Finder (ADF)
  - One or two Global Positioning System (GPS)

- FLIGHT MANAGEMENT SYSTEM (FMS)

  The FMS combines the inputs of different aircraft systems (GPS, DME, VOR, AHRS and ADS) to provide navigation, lateral and vertical commands and aircraft performance predictions.



NAVIGATION SYSTEMS – PRIMUS EPIC® GENERAL ARRANGEMENT

AIR DATA SYSTEM (ADS) – GENERAL

The two Air Data Systems (ADS 1 and ADS 2) provide:

- Barometric Altitude (BARO)
- Indicated Airspeed (IAS)
- Vertical Speed (VS)
- Outside Air Temperature (OAT)

Air data are also used by:

- AFCS
- AHRS
- FMS
- Weather Radar (if installed)
- TCAS (if installed)

AIR DATA SYSTEM – MAIN COMPONENTS

The ADS main components include

- two Pitot-Static Probes
- two Air Data Modules (ADM 1 and ADM 2)
- two OAT Probes
- two Alternate Static Sources

PITOT STATIC PROBES

Each Pitot-Static Probe is installed on a forward fuselage side in front of cockpit doors and include:

- one total pressure port
- two static pressure ports
- an electrical heater to prevent ice formation (see chapter 30-00)

A drain hole permits draining any water which has condensed inside the probe.

AIR DATA MODULE (ADM)

Each ADM is a solid state transducer that converts dynamic and static pressure values into digital signals used by the MAU to generate air data parameters such as calibrated airspeed, altitude, vertical speed and OAT.
The Air Data Modules are installed in the nose compartment on the cockpit bulkhead.

OUTSIDE AIR TEMPERATURE (OAT) PROBES

The two outside air temperature probes sense the temperature of the external air and are installed in the bottom forward fuselage.

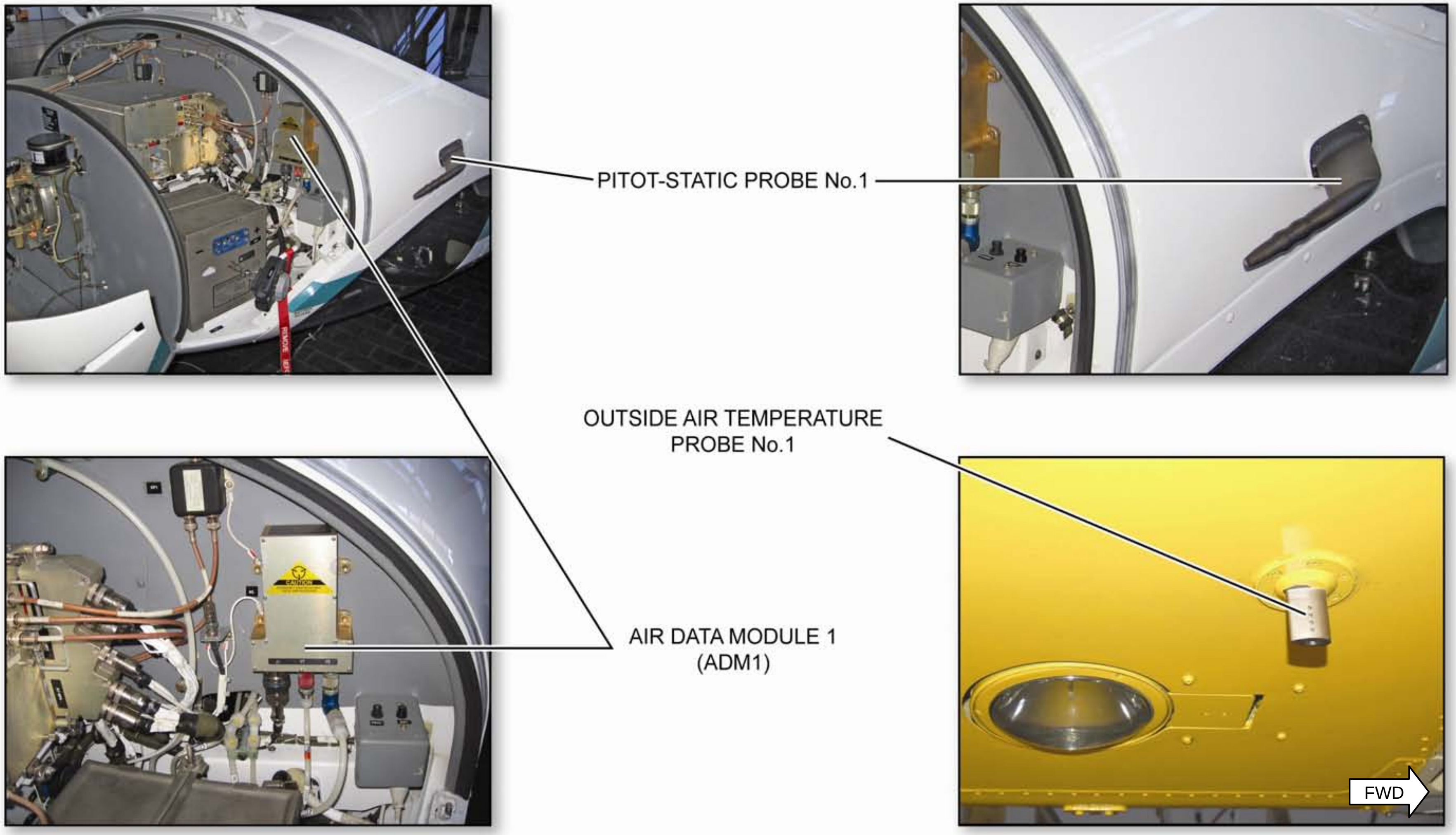
ALTERNATE STATIC SOURCES

The two alternate static sources are static ports located in the overhead panel: one on the LH sidewall (co-pilot) which is part of ADS 1 and the other on the RH sidewall (pilot) which is part of ADS 2.

Each port is integral with a lever-type selector valve that can be operated by the crew. The control is protected by a red guard to prevent inadvertent actuation.

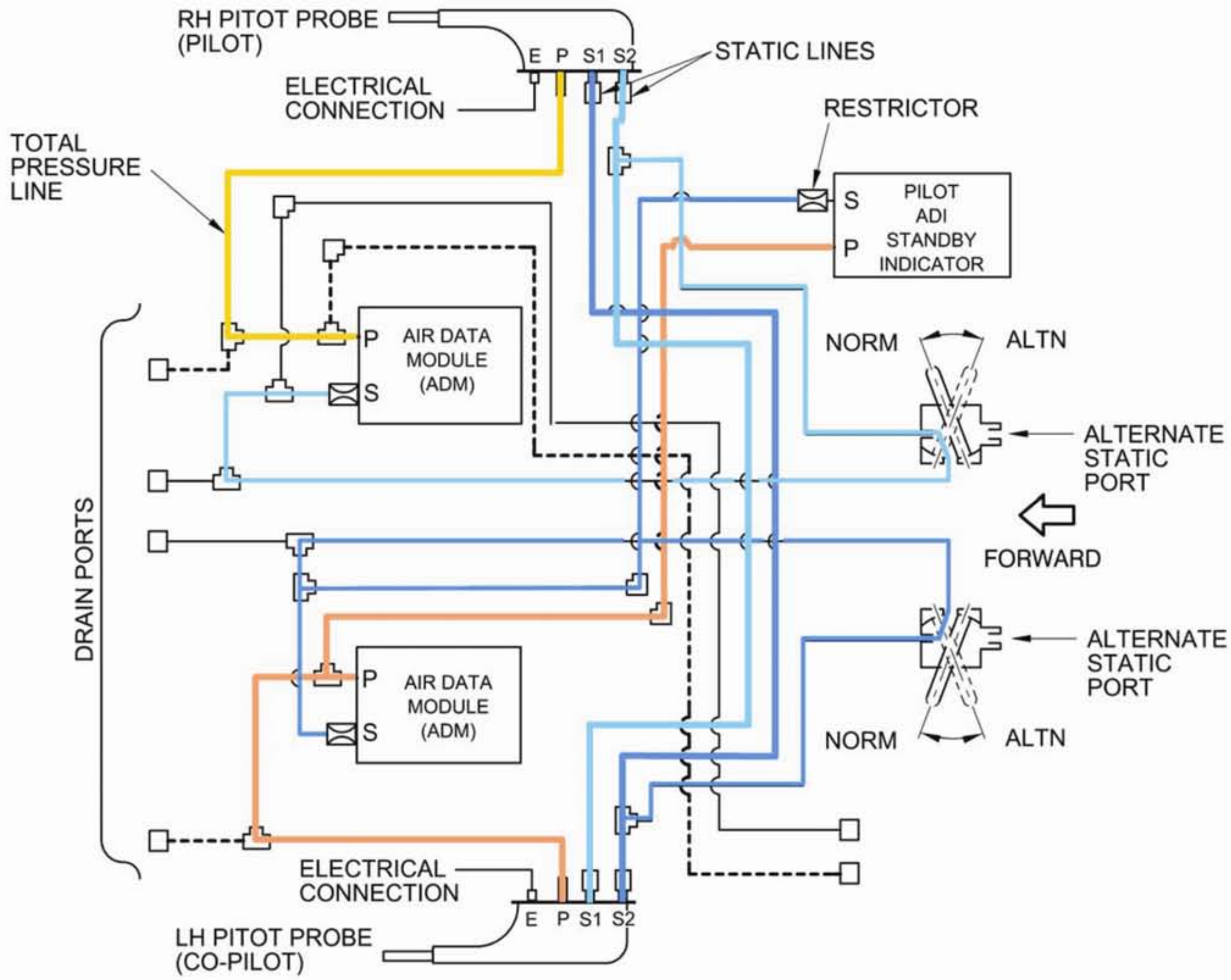
AIR DATA SYSTEM DISPLAY HANDLING

Normal usage of ADS data for display to the crew is “on-side”, i.e. ADS 1 for co-pilot’s DUs and ADS 2 for pilot’s DUs.
In case an ADM fails the 1(2) ADS caution message is shown in the CAS window and relevant data are lost: the on-side display shows failure flags on the airspeed, vertical speed and baro altimeter.
To restore air data display pilot must perform manual reversion: the ADS selector on the Reversion Control Panel (RCP) is to be selected to the non-failed ADS. The selected system supplies all DUs and the ADS1(2) single-source annunciator is displayed on both PFDs.
The ADS annunciator displays which ADS (ADS1 or ADS2) is currently in use for both pilot and copilot a display. The ADS annunciator is not displayed when the ADS sources are from the on-side ADS: in this case the ADS reversion switch on the Reversion Control Panel (RCP) allows to select the ADS source for both pilot and copilot PFD.

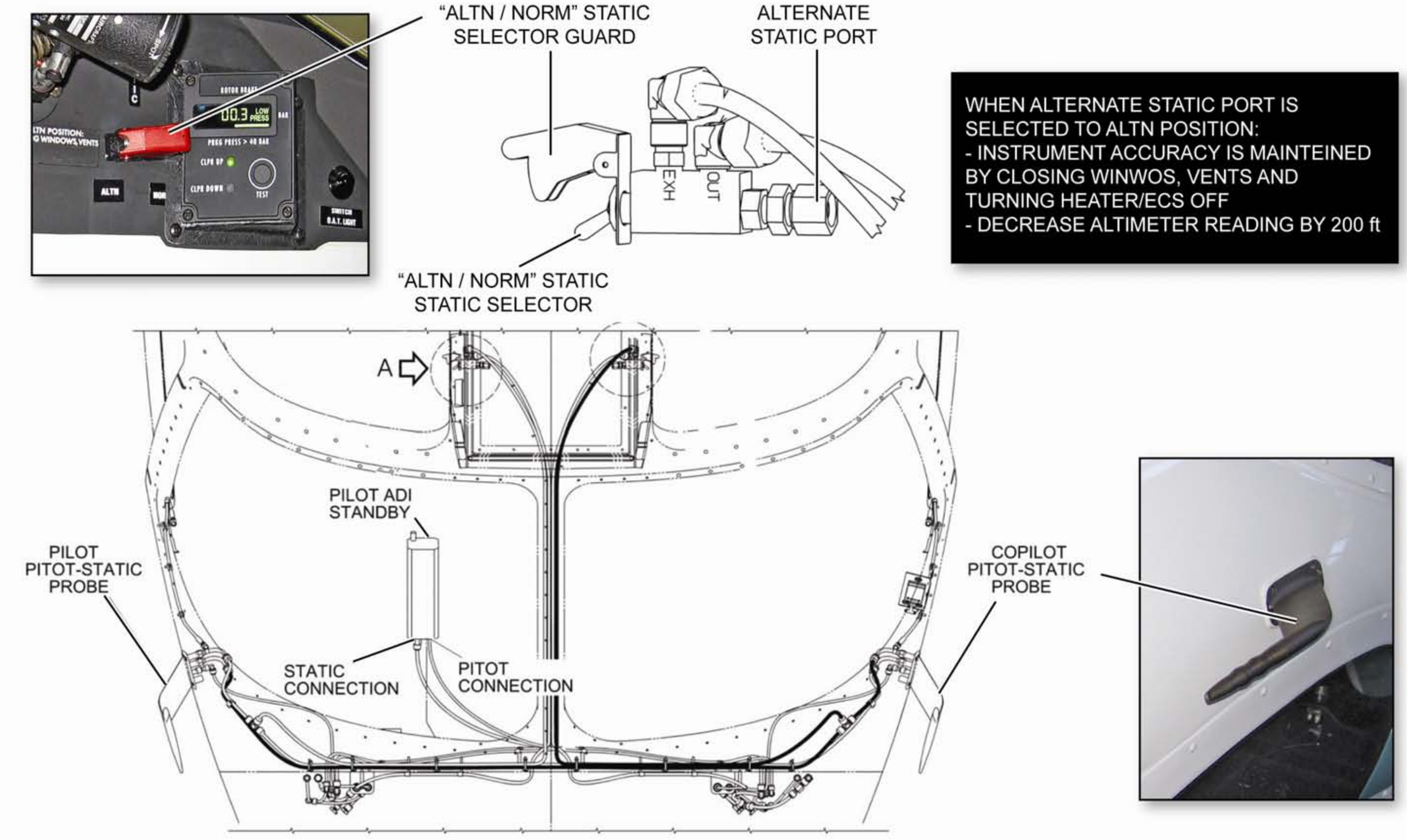


NOTE: RH side components are arranged symmetrically

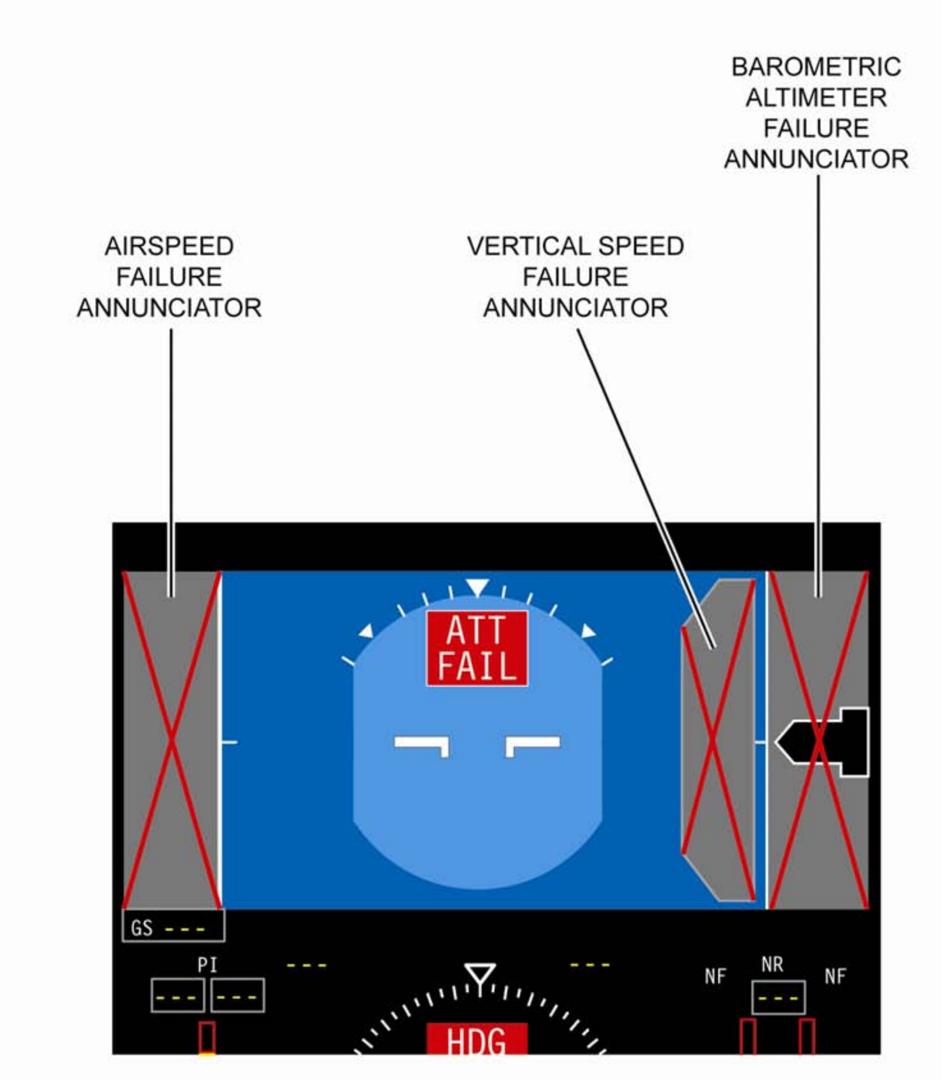
AIR DATA SYSTEMS – MAIN COMPONENTS



PITOT AND STATIC PRESSURE LINES - SCHEMATIC



AIR DATA SYSTEM – PITOT-STATIC LINES AND ALTERNATE STATIC PORTS



AIR DATA SYSTEMS – PDF INDICATORS

AIR DATA SYSTEM - CONTROLS AND INDICATORS

1. ADS switch on RCP (Reversion Control Panel)

   NORM …..... the ON-SIDE ADS data are displayed on each PFD: ADS 1 on PDF1 and ADS 2 on PDF2

   1 ………….. pilot and copilot displays visualize ADS 1 data

   2 ………….. pilot and copilot displays visualize ADS 2 data

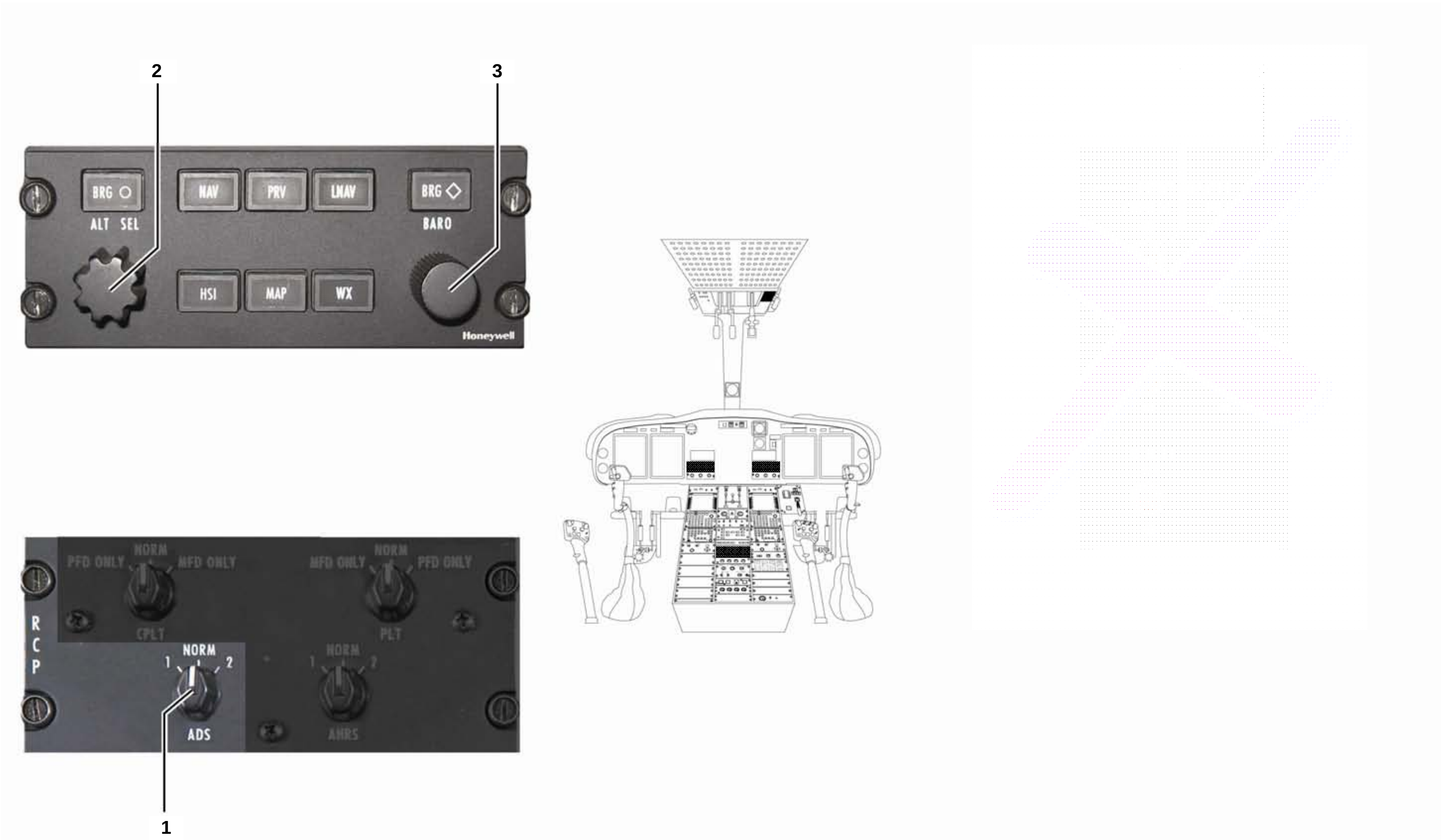
2. ALT SEL knob on DC (Display Controller)

   ……………..... allows to control the setting of the barometric altitude reference on the pilot and copilot PFD altitude tapes. The selected altitude is used for the altitude preselect and altitude alert functions. Rotating the knob clockwise increases the preselect value and counterclockwise rotation decreases the value. The selected altitude drives the altitude intercept function as well as the altitude alert function when in a flight director mode.

3. BARO knob on DC

   ……………..... allows to change the barometric pressure setting on the PFD for the on-side Air Data Computer (ADC). The barometric correction is synchronized between the two air data functions.

   NOTE. The barometric pressure can be set in inches of mercury (IN) or hectopascals (HPA) as displayed at the right of the digital readout.



AIR DATA SYSTEM – CONTROLS AND INDICATORS

AIR DATA SYSTEM – PRINCIPLE OF OPERATION

The LH (copilot) Pitot-Static Probe supplies total pressure to ADM 1 and to the pilot Standby Instrument.

The RH (pilot) Pitot-Static Probe supplies total pressure to ADM 2, only.

Two static pressure lines are obtained by interconnecting a static port on LH probe with one of the RH probe to compensate any lateral unbalance. One line supplies static pressure to ADM 1 and to the pilot Standby Instrument, while the other supplies static pressure to ADM 2, only.

In the event of Indicated Air Speed (IAS) erratic readings when operating in rain with static source, the Alternate Static selector must be set to ALTN position. In this case the air data transducer of the affected line receives static pressure from the alternate static port: instruction for use and correction are given by a placard located near the Alternate Static selector.

AIR DATA SYSTEM INDICATORS

AIRSPEED DISPLAY

- AIRSPEED TAPE

  The airspeed tape slides up or down as airspeed decreases or increases.

- AIRSPEED DIGITAL READOUT

  The digital readout shows in green digits the current airspeed in a fixed window in the center of the airspeed tape. The last digit of the current airspeed indicator rolls in one-knot increments.

  The digital readout is displayed in amber reverse video when the trend vector exceeds V_{NE}.

  The digital readout is displayed in red reverse video and the AIRSPEED aural warning message is played back when the current airspeed exceeds V_{NE}.

- AIRSPEED TREND VECTOR INDICATOR

  The airspeed trend vector indicator is a rectangular magenta thermometer-type bar that travels on the outside edge of the airspeed tape. The bar indicates acceleration as it travels up the tape and deceleration as it travels down. The algorithm for trend is a filter made up from airspeed, sine of pitch, and longitudinal acceleration.

  The airspeed trend vector indicator is not displayed when there is no longitudinal acceleration or IAS < 60 knots or source data is invalid.

- TARGET AIRSPEED BUG and DIGITAL READOUT

  The target airspeed bug is a rectangular magenta bug positioned on the airspeed tape to indicate the selected airspeed for the IAS (Airspeed Hold) mode of Flight Director. The magenta digital readout above the airspeed tape displays the target airspeed value.

  The target airspeed bug and the associated digital readout are only displayed when the IAS mode is engaged in the FD (see chapter 22-00).

  When airspeed is invalid the digital readout shows amber dashes and the airspeed bug is not displayed.

- V_{NE} INDICATOR

  A solid red horizontal line on the airspeed tape indicates the current VNE in AEO conditions. In case of OEI or power-off (autorotation) conditions the indicator shows as a barber pole.

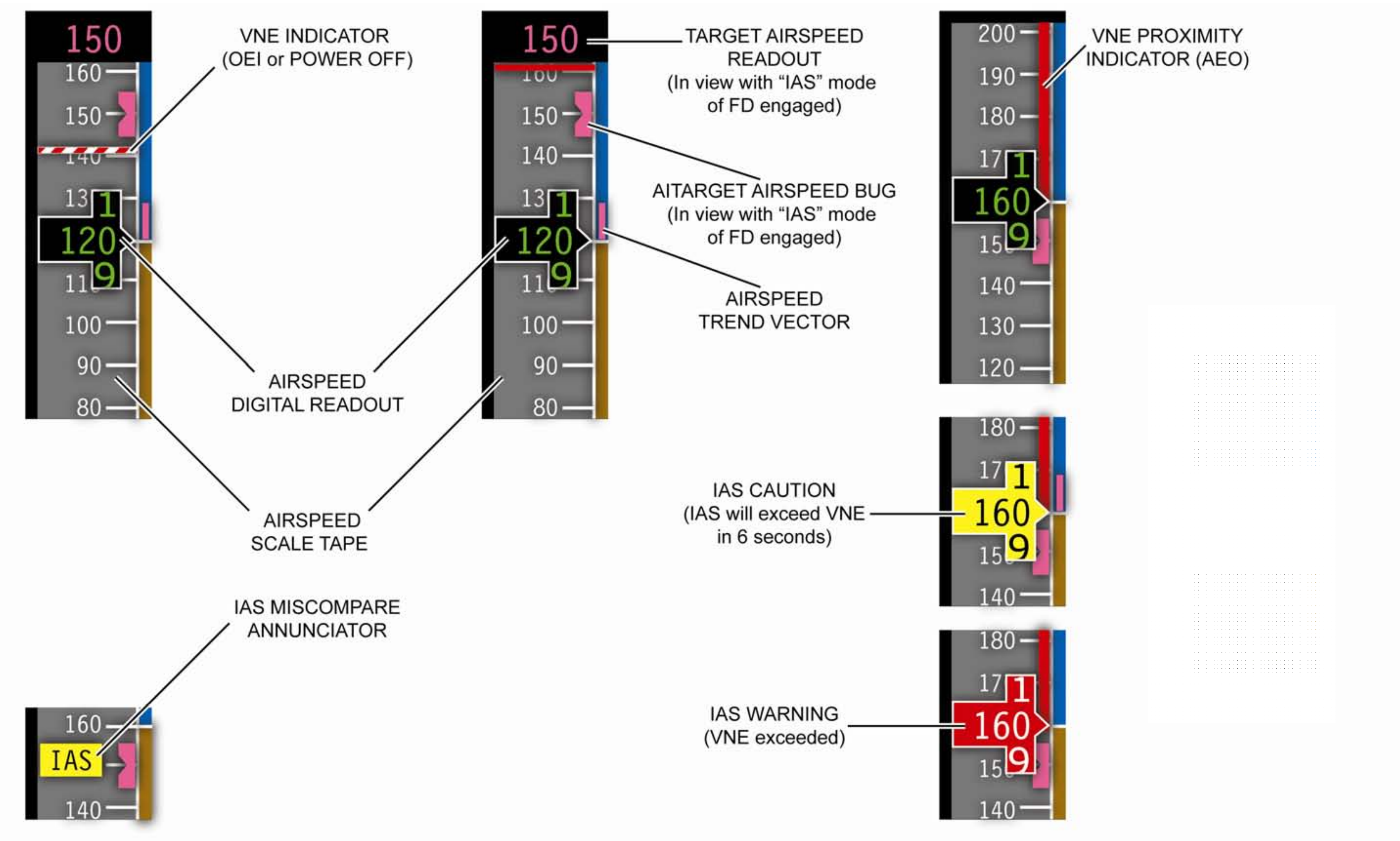
  At 5 knots prior to exceeding V_{NE}, the horizontal line extends vertically upwards becoming a thick red (AEO) or barber pole (OEI and autorotation) vertical bar.

- IAS MISCOMPARE ANNUNCIATOR

  The IAS miscompare annunciator is displayed in the top half of the airspeed tape when indicated airspeed miscompare between ADS 1 and ADS 2 is detected.

- IAS INVALID ANNUNCIATOR

  Loss of valid airspeed data from the selected ADS is annunciated by a red cross that replaces all indications in the airspeed display.



AIRSPEED INDICATOR

VERTICAL SPEED INDICATOR

- VERTICAL SPEED SCALE and POINTER

  The vertical speed scale shows a vertical speed range between ± 3000 fpm.

  A green pointer moves against the scale to indicate current VS.

- VERTICAL SPEED DIGITAL READOUT

  The vertical speed digital readout is displayed at the center of the scale and shows the current vertical speed rounded to the nearest 50 fpm.

  The VS digital Readout is not shown when vertical speed is less than ±300 fpm.

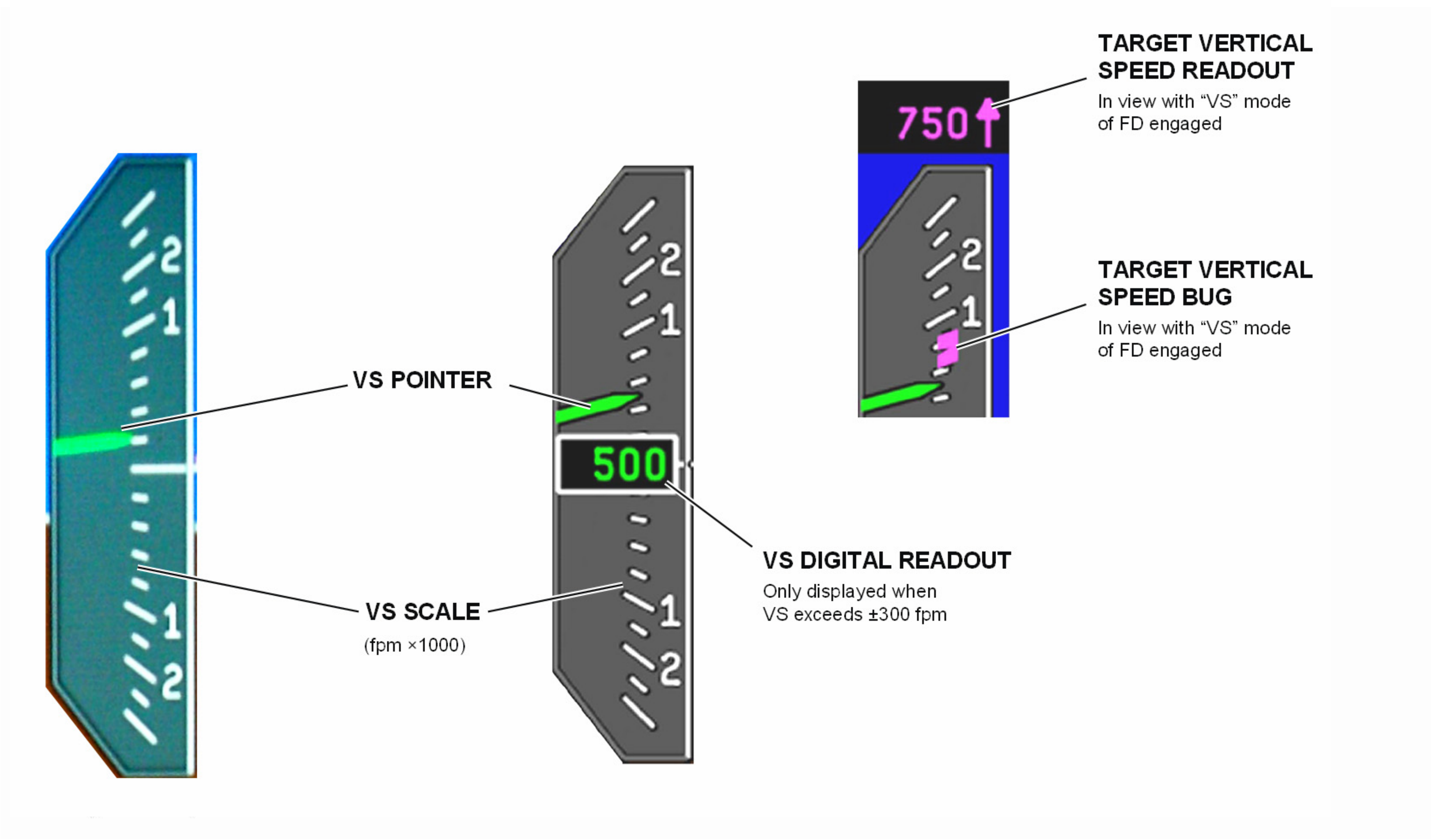
- TARGET VERTICAL SPEED BUG and DIGITAL READOUT

  The target VS bug is a rectangular magenta bug positioned on the vertical speed scale to indicate the selected vertical speed for the VS (Vertical Speed Hold) mode of Flight Director.

  The magenta digital readout above the scale displays the target vertical speed value. Climb and descent speed targets are indicated with an arrow pointing up or down.

  The target vertical speed bug and the associated digital readout are only displayed when the VS mode is engaged in the FD (see chapter 22-00).

  When vertical speed is invalid the digital readout and the vertical speed bug are not displayed.



VERTICAL SPEED INDICATOR

BAROMETRIC ALTITUDE INDICATORS

- BAROMETRIC ALTIMETER TAPE

  The altitude tape slides up or down as altitude decreases or increases and shows a range of ± 550 ft around current aircraft altitude.

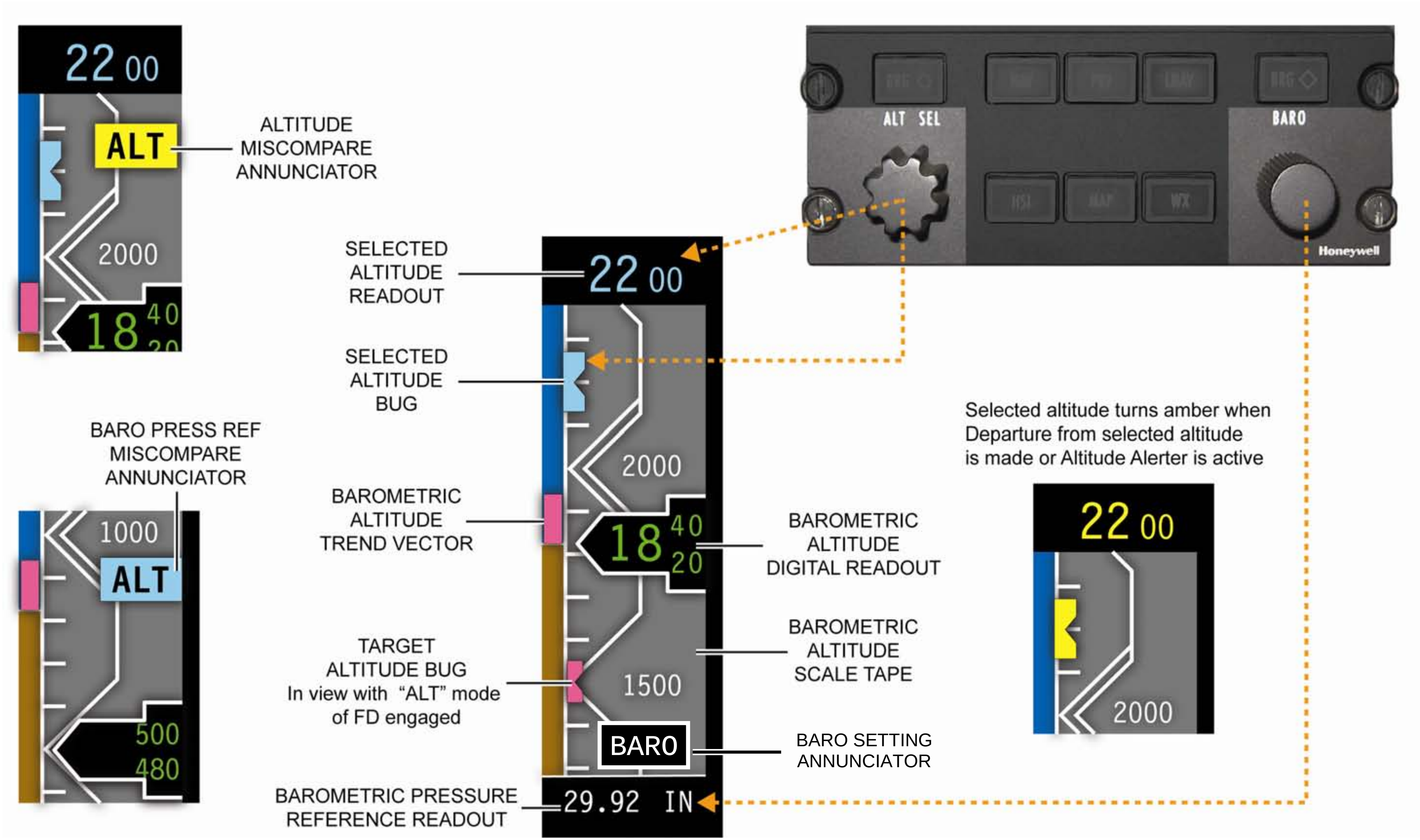
  A double line chevron marks each 1000 ft increment. A single line chevron marks each 500 ft increment.

- BAROMETRIC ALTIMETER DIGITAL READOUT

  The digital readout shows in green digits the current altitude in a fixed window in the center of the altimeter tape. The last two digits roll in twenty-foot increments.

  NEG annunciator stands for altitudes below sea level. A loss of valid altitude from the ADS shows a red cross over the altitude tape, the scale markings and the rolling digital display are not displayed.

- TREND VECTOR INDICATOR
- The altitude trend vector originates at the altitude reference line and indicates what the altitude trend will be in 6 seconds based on the current VS. The trend vector indicator and scale are not displayed when no valid altitude information come from the ADS.
- ALTITUDE PRE-SELECTED DIGITAL READOUT
- A digital readout and the bug of the preselect altitude are displayed in cyan above the altitude tape. When altitude preselect is in Flight Director mode, the bug and digital readout are displayed in magenta. When the altitude alert is active or a departure from altitude has been made, the digital readout and bug turn amber.
- ALTITUDE PRESELECTED BUG
- The altitude preselect bug is a cyan notched rectangle that indicates the preselected altitude set on each PFD. Rotating the ALT SEL knob on the Display Controller moves the bug vertically and changes the altitude preselect value. The rate of altitude pre-selection varies with the rate the knob is turned (the slower the knob is turned the more precise the selection is).
- ALTITUDE REFERENCE BUG AND READOUT
- The altitude reference bug is displayed when the ALT HOLD flight director mode is engaged. The bug is a magenta notched rectangle that is smaller than the altitude select bug. The set point is determined by the priority FD. The altitude reference bug is not displayed when valid information from the ADS and priority flight director is lost. The bug is displayed in amber when the altitude alert function is in an active state.
- ALT MISCOMPARE ANNUNCIATOR
- The pilot is alerted to altitude miscompare by an amber ALT annunciator. The annunciator is removed when altitude miscompare is no longer detected. A cyan ALT annunciator is displayed when BARO ALT miscompare is detected. The annunciator is removed when the BARO ALT miscompare is no longer detected. The amber ALT miscompare annunciator has display priority over the cyan ALT annunciator.



BAROMETRIC ALTITUDE INDICATOR

AIR DATA SYSTEM – CAS CAUTION MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
1(2) ADS FAIL and loss of - AIRSPEED - ALTITUDE - VERTICAL SPEED data on left (right) PDF indicator	Associated ADS system failure On RCP move ADS switch to non failed ADS **ADS 1(2)** illuminates on attitude indicator to highlight both PFDs are using the same ADS source data. Compare frequently PFD data with Standby Indicator.	ADS FAILURE	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES ADS FAILURE

PITOT SYSTEM – LIMITATIONS

Refer to AW139-RFM-4D.

ATTITUDE AND HEADING REFERENCE SYSTEM (AHRS) – GENERAL

The Attitude and Heading Reference System (AHRS) generates attitude and heading data used by the displays, the AFCS, the WX radar and other helicopter systems. The AHRS provides for

- pitch and roll angles
- magnetic heading
- angular rates around aircraft axes
- accelerations

The AHRS main components are

- the Attitude and Heading Reference Units (AHRU)
- the flux valves
- the AHRS control panels (PLT and CPLT)

There are two AHRS on the helicopter named AHRS1 and AHRS2.

Heading information is generated with respect to the Fiber Optic Gyros (FOG) or the magnetic field of the earth as determined by using the MAG/DG switch on the AHRS control panel and displayed on the PFD, along with attitude, where it is used by the pilot for navigation and to maintain flight path direction. True Air Speed input from one or two external digital ADS is used to improve the attitude performance. In addition, AHRS is able to give inertial altitude and vertical speed if augmented by pressure altitude from one or two digital ADS. Utilizing data from the GPS receiver in combination with air data, the AHRS gives additional output data such as ground speed for navigation applications.

The Weight-On-Wheels (WOW) switches tell to the AHRS when the helicopter is on the ground.

AHRS – MAIN COMPONENTS

ATTITUDE AND HEADING REFERENCE UNIT (AHRU)

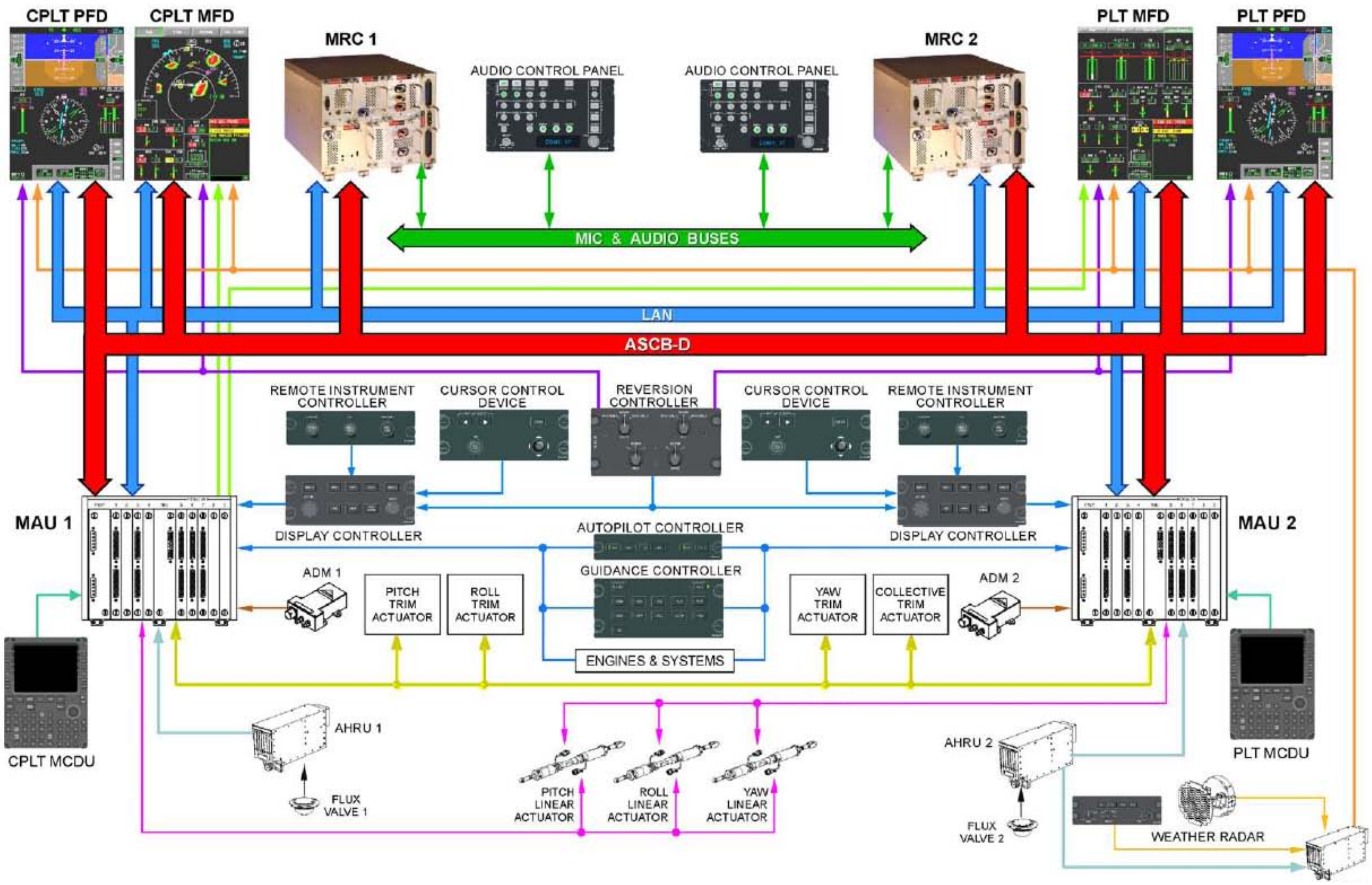
The AHRU is the main electronic assembly of the AHRS. It contains the inertial sensor assembly, which include FOGs and accelerometers. The AHRU computes flight parameters and displays functions. A memory module stores flux valve compensation data and aircraft AHRU orientation data. The AHRU supplies a 28 V DC to flux valve. True Air Speed information is supplied by the air data modules.

FLUX VALVE

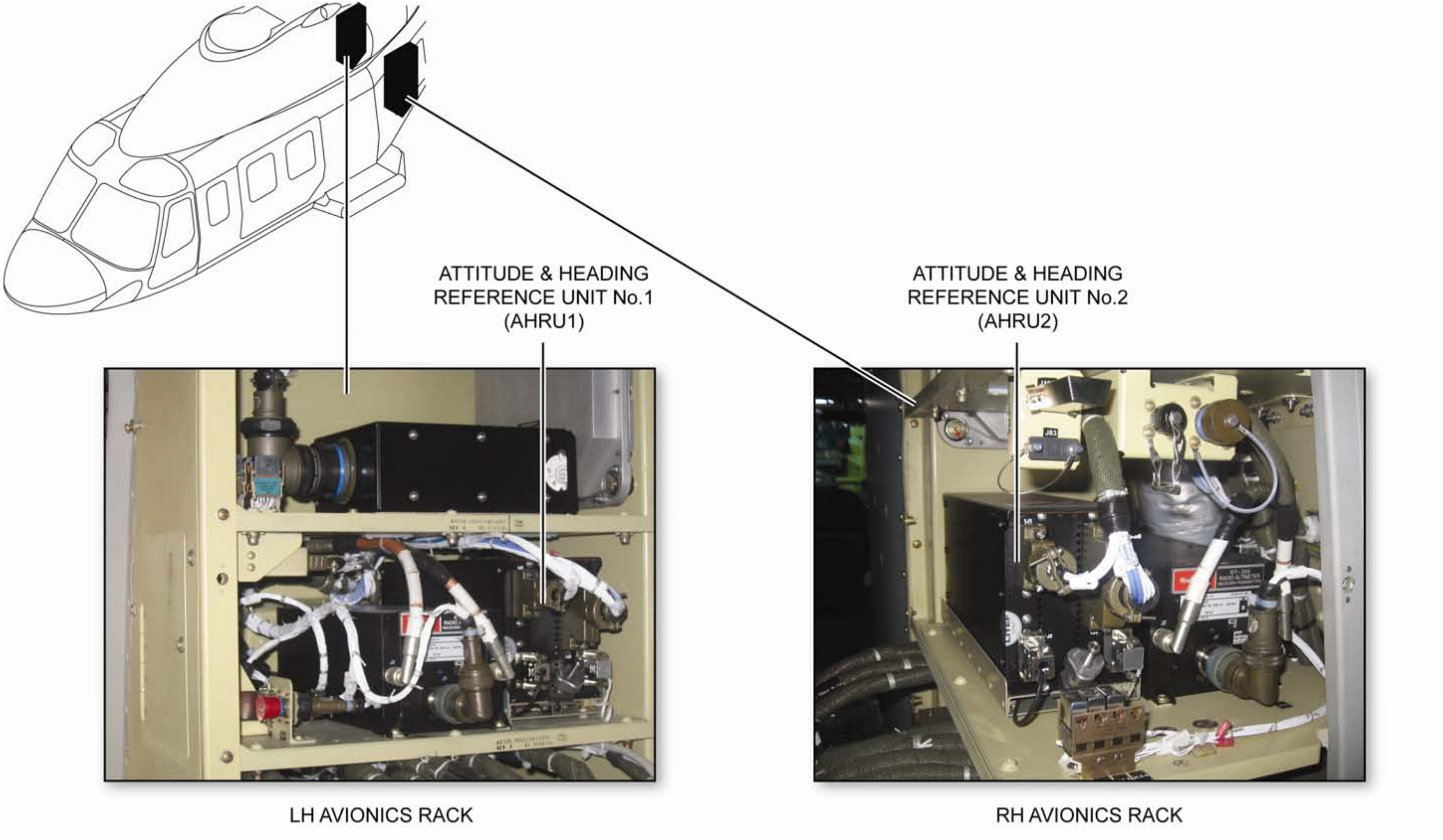
The flux valve consists of a sensitive pendulous element that is free to swing within limits but fixed to the aircraft in azimuth. There are two flux valves one for each AHRU.

AHRS CONTROL PANEL

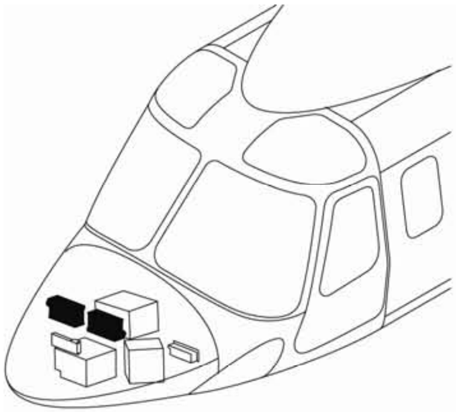
The compass controller enables to control the operation of the on side AHRS.



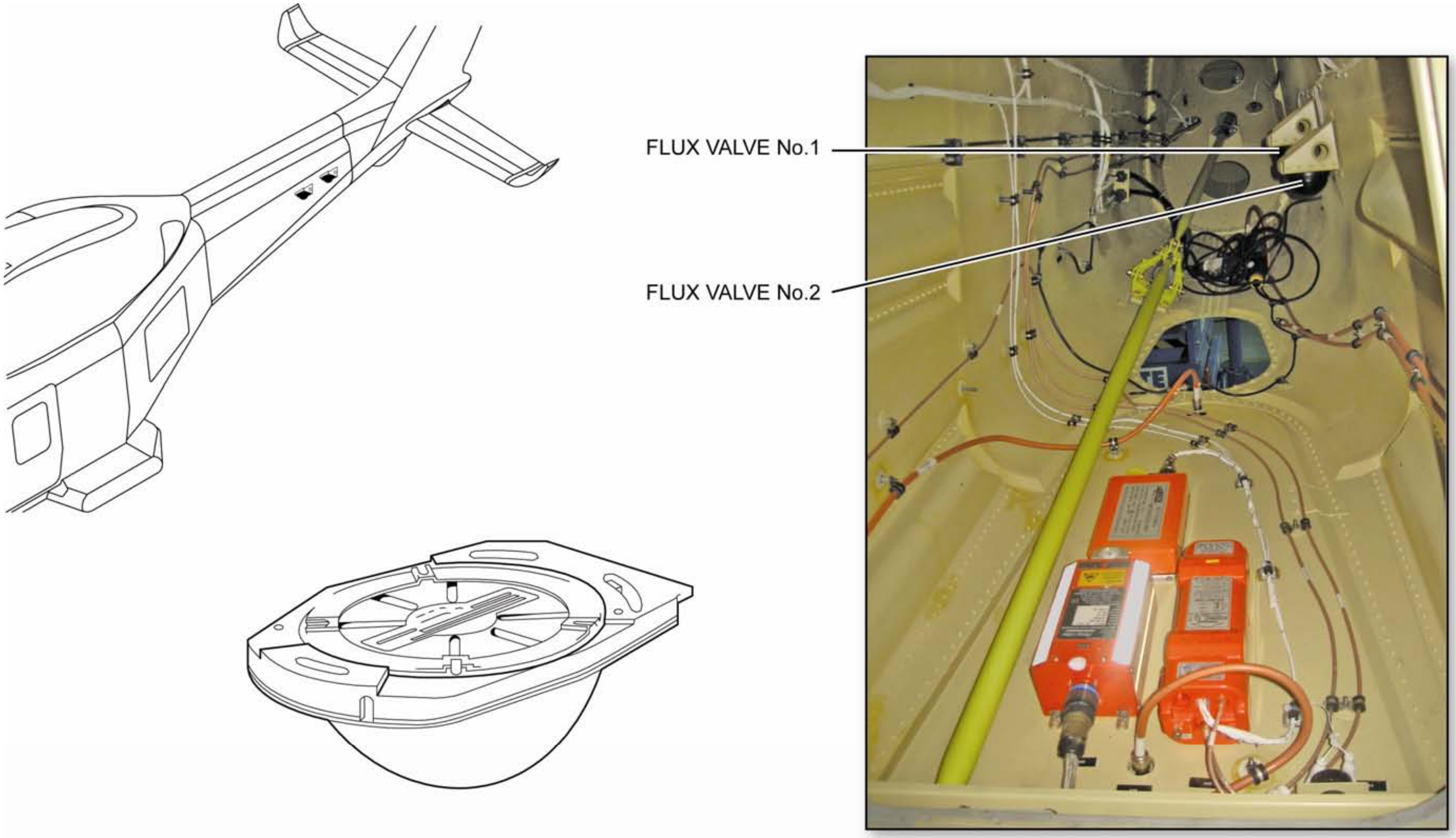
ATTITUDE AND HEADING REFERENCE SYSTEM (AHRS) – GENERAL LAYOUT



AHRS – TYPICAL CONFIGURATION (1 OF 2)



AHRS – LONG NOSE CONFIGURATION (2 OF 2)



AHRS – FLUX VALVE INSTALLATION

AHRS - CONTROLS AND INDICATORS

1. TEST button

pressed …... activates the system self-test (on ground only). The test is disabled during start-up or shut-down and while in flight. The test result are: Pitch = +5°; Roll = +45°; Heading = +15°.

In addition to the recorded attitude and heading data, the **ATT TEST** annunciator is displayed in the ADI and the **HDG TEST** annunciator is displayed in the compass. When invalid data is due to an AHRS test, the **ATT TEST** annunciator is displayed instead of **ATT FAIL**

2. COMPASS annunciator

…………….. the compass annunciator indicates a misalignment of the gyro-compass when the needle is not centered

3. MAG / DG switch

MAG …....... activates the MAG sub-mode. This changes the heading indicator reference from being driven by the directional gyros to being referenced to local magnetic north

DG ……….. slaves the heading indicator to the directional gyros (FOG)

4. SYNC switch (spring loaded)

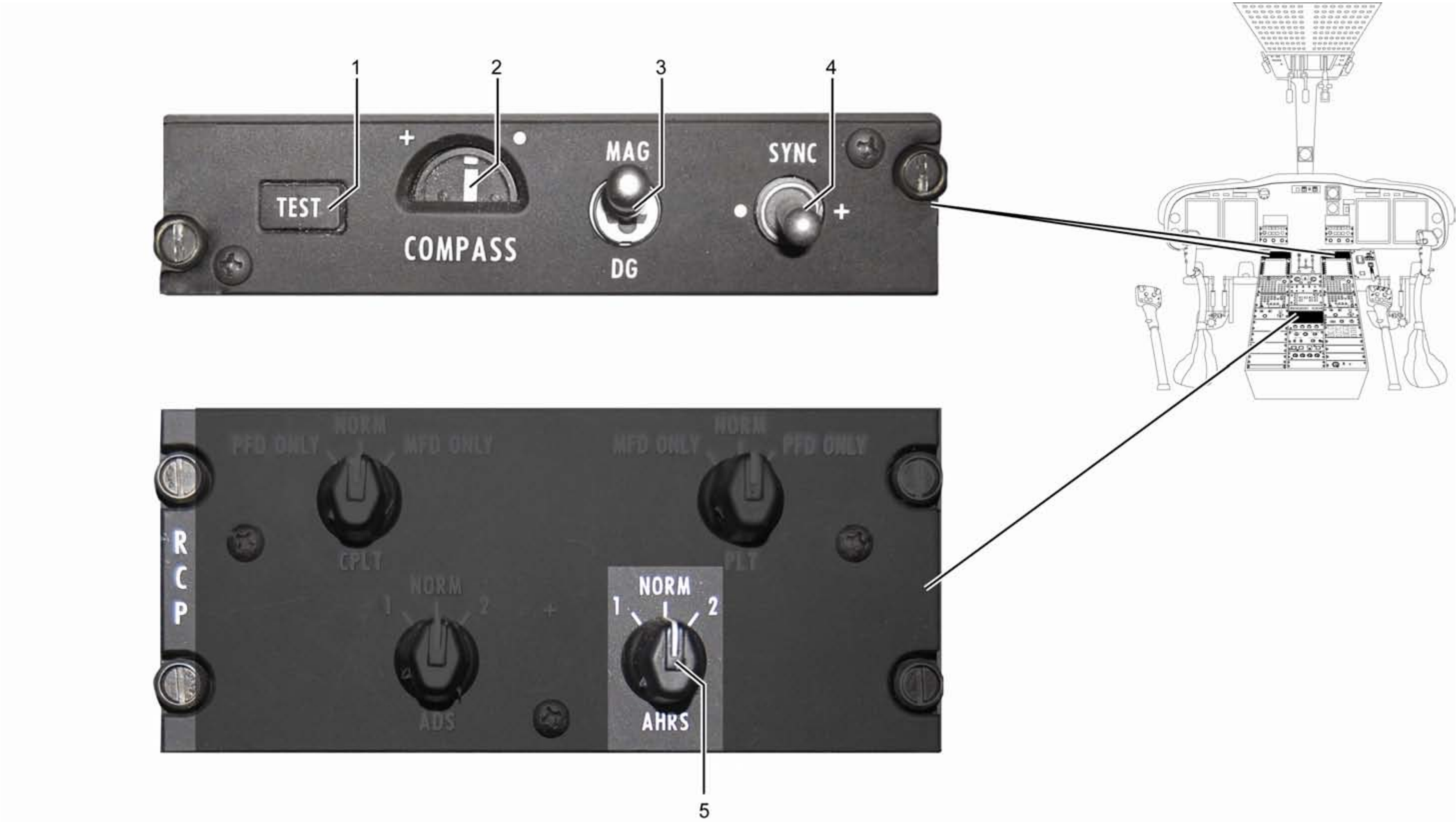
…………..... It is toggled in the direction indicated by the compass annunciator either + or - to align the gyro-compass

5. AHRS switch on RCP (Reversion Control Panel)

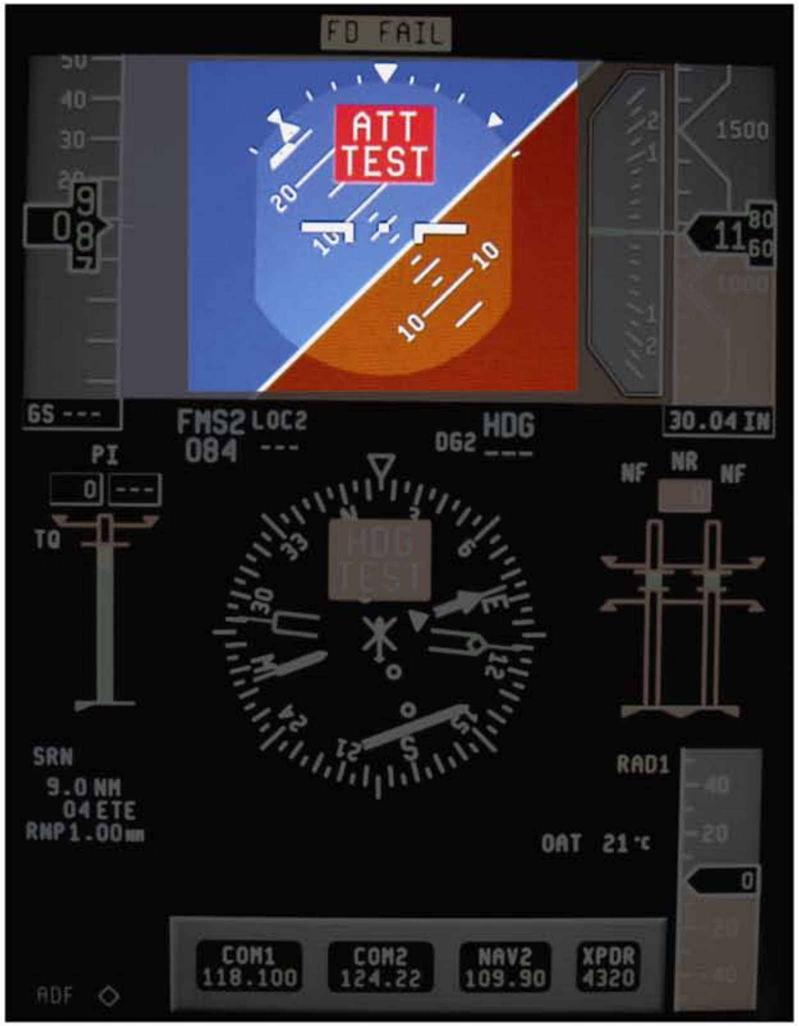
NORM …..... the ON-SIDE ADS data are displayed on each PFD: AHRS 1 on PDF1 and AHRS 2 on PDF2

1 ………….. PDF1 and PDF2 display AHRS 1 data

2 ………….. PDF1 and PDF2 display AHRS 2 data



AHRS – CONTROLS AND INDICATORS (1 OF 2)



AHRS TEST RESULTS (ON GROUN ONLY)

ATTITUDE SOURCE ANNUNCIATOR

AHRS – CONTROLS AND INDICATORS (1 OF 2)

AHRS – PRINCIPLE OF OPERATIONS

The system comprises inertial sensor using Fiber Optic Gyros (FOGs) and accelerometers to compute attitude, heading and flight information.

FOGs and accelerometers are mechanically fixed to the helicopter axes and measure rates of change in pitch, roll, and yaw. A computer puts together the rate of change from the sensors to find and continuously update the values of pitch, roll, and heading.
The flux valves, the two ADS and the Global Positioning Systems (GPS) increase the system capabilities.

ATTITUDE DIRECTION INDICATOR (ADI)

The main properties of the symbols depicted on the ADI are the following:

- ADI SEGMENT

  Displays pitch attitudes between ±90° and roll attitudes through ±180°.

- ARTIFICIAL HORIZON

  The natural horizon is represented as a white horizontal line that joins the bottom edge of the cyan field (the sky) with the top edge of the brown field (the surface of the earth).

- AIRCRAFT REFERENCE SYMBOL

  The symbol is represented in white at the center of the ADI. The position of the symbol in relation to the attitude pitch tape indicates the number of degrees the aircraft is nose-up or nose-down. The aircraft reference symbol indicates bank angle changes, the magnitude of the change (in degrees) and the rate at which the bank angle is changing.

- ATTITUDE PITCH TAPE

  The pitch tape is displayed in the center of the ADI with white linear markings that move up or down behind the aircraft symbol as the pitch attitude changes.

- PITCH ATTITUDE WARNING INDICATOR

  When exists an excessive pitch angle (both nose-up or nose-down) the red warning indicators are displayed. The indicators increase in size as the nose-up or nose-down angle increases.

- ROLL SCALE

  A linear roll scale is visible at the top of the ADI. An inverted triangle marks 0° and the 45° angle of bank.

- ROLL POINTER AND SLIP/SKID INDICATOR

  Bank angle and roll rates are identified by a white segmented triangle called the roll pointer. The bottom segment of the roll pointer is the slip/skid indicator. It identifies a slip or skid by displacing left or right of the top segment of the roll pointer. As long as the aircraft is flown in coordinated flight, the roll pointer and slip/skid

indicator form a white triangle. When a slip or skid is made, the slip/skid indicator moves sideways and turns amber. When the aircraft rate of turn is too great for the existing angle of bank, the aircraft is in a skid. The slip/skid indicator displaces outside the roll pointer away from the turn direction. When the aircraft rate of turn is too slow for the existing angle of bank, the aircraft is in a slip. The slip/skid indicator displaces inside the roll pointer in the direction of the turn. The skid pointer displaces laterally and turns amber when lateral acceleration exceeds 0.1 g. In analog terms, when the skid pointer displaces laterally and turns amber, the aircraft yaw is one ball out of trim. When lateral acceleration, pitch or roll information from the AHRS becomes unreliable, the slip/skid indicator is removed from the PFD and is replaced with an amber X.

ATTITUDE SOURCE ANNUNCIATORS

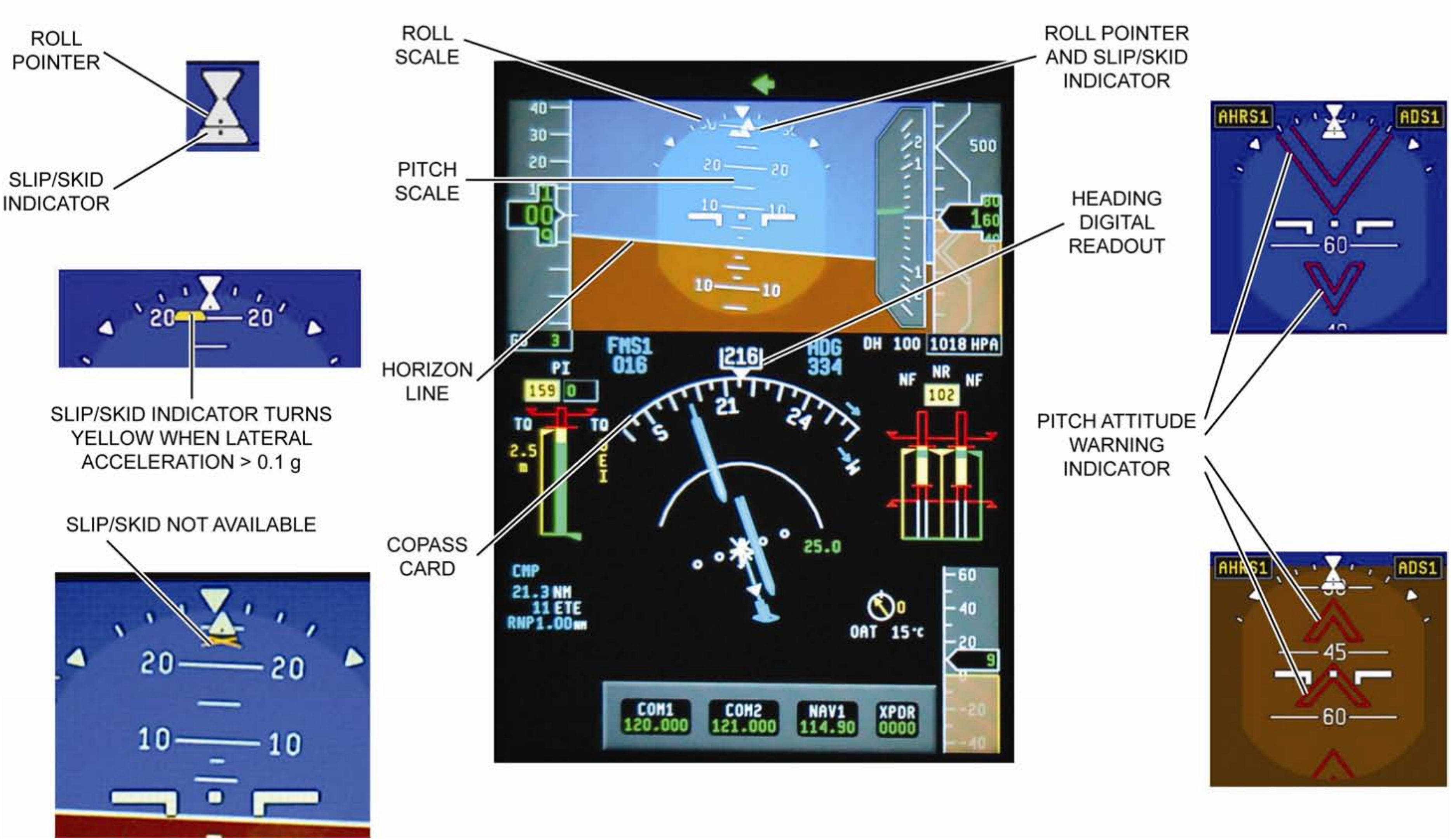
The source of attitude information, that is AHRS1 or AHRS2, is displayed on the left corner of the ADI. The annunciator is not displayed when both PFD are displaying information from the on-side AHRS. When the pilot and copilot PFD are displaying information from the same source, AHRS1 or AHRS2 annunciates in reverse video on each PFD.

ATTITUDE MISCOMPARE ANNUNCIATORS

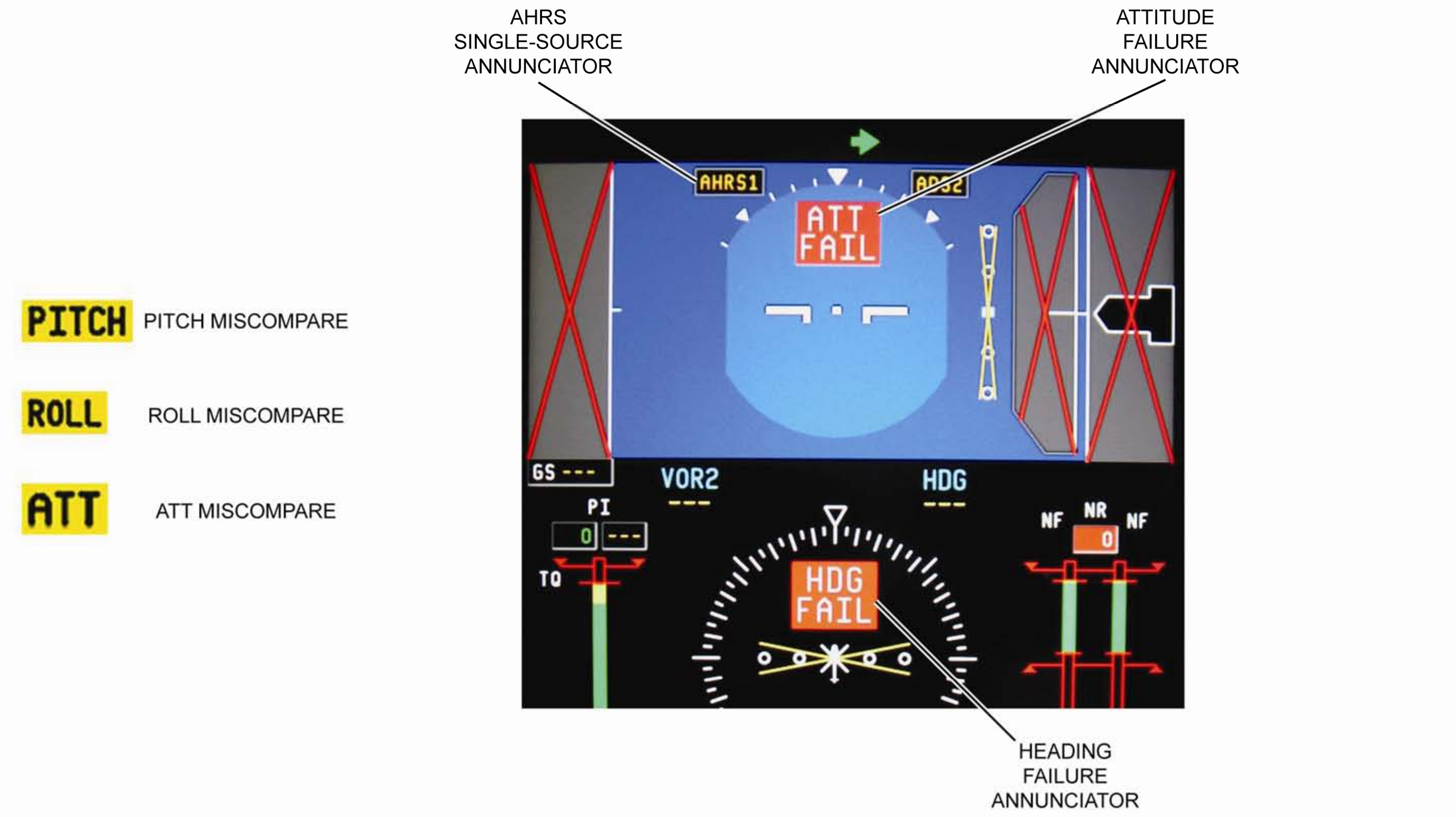
Attitude data from the two AHRS are continuously compared so that, when a performance discrepancy between the two AHRS is found, the monitor warning system alerts the pilot by annunciating one of the following miscompare messages:

- PITCH miscompare is displayed when the pitch attitude from the two AHRS disagrees by 5°
- ROLL miscompare is displayed for a roll attitude miscompare of greater than 6°
- ATT miscompare: is displayed when both pitch and roll thresholds are exceeded. When attitude information is unreliable, the ATT miscompare is removed from the PFD

A loss of valid pitch or roll information from the AHRS is indicated on the display when the pitch tape, roll pointer and flight director bars are not displayed. In addition, the boxed ATT, PITCH or ROLL annunciators are not displayed while the attitude sphere turns cyan and the ATT FAIL annunciator is displayed.



ATTITUDE SOURCE ANNUNCIATORS



ATTITUDE MISCOMPARE ANNUNCIATORS

ATTITUDE AND HEADING REFERENCE SYSTEM – CAS CAUTION MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
1(2) AHRS FAIL **AVIONC FAULT** **AFCS DEGRADED** **1(2) AP FAIL** + aural message **ATT FAIL** **HDG FAIL** and loss of attitude, heading and slip skid data on left (right) PFD	Associated AHRS failure and subsequent 1(2) AP failure	AHRS FAILURE	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES AVIONICS

RADIO ALTIMETER – GENERAL

The Radio Altimeter (RAD ALT) system uses Frequency-Modulated Continuous Wave (FMCW) signals to provide

- radio height
- low height awareness

Radio altitude data are shown on the PFD (Compass mode, Arc mode and Reversion mode). The receiver-transmitter sends digital radio altitude data to the MAU through an ARINC 429 data bus. The MAU changes the digital radio altitude data to a format that can be shown on the PFD.

In a dual radio altimeter system, the receiver-transmitter no.1 supplies the radio altitude data for the co-pilot's PFD and the receiver-transmitter no.2 supplies the radio altitude data for the pilot's PFD.

RADIO ALTIMETER – MAIN COMPONENTS

The RAD ALT main components are

- two radio altimeter transceivers (no.1 and no.2)
- four antennas

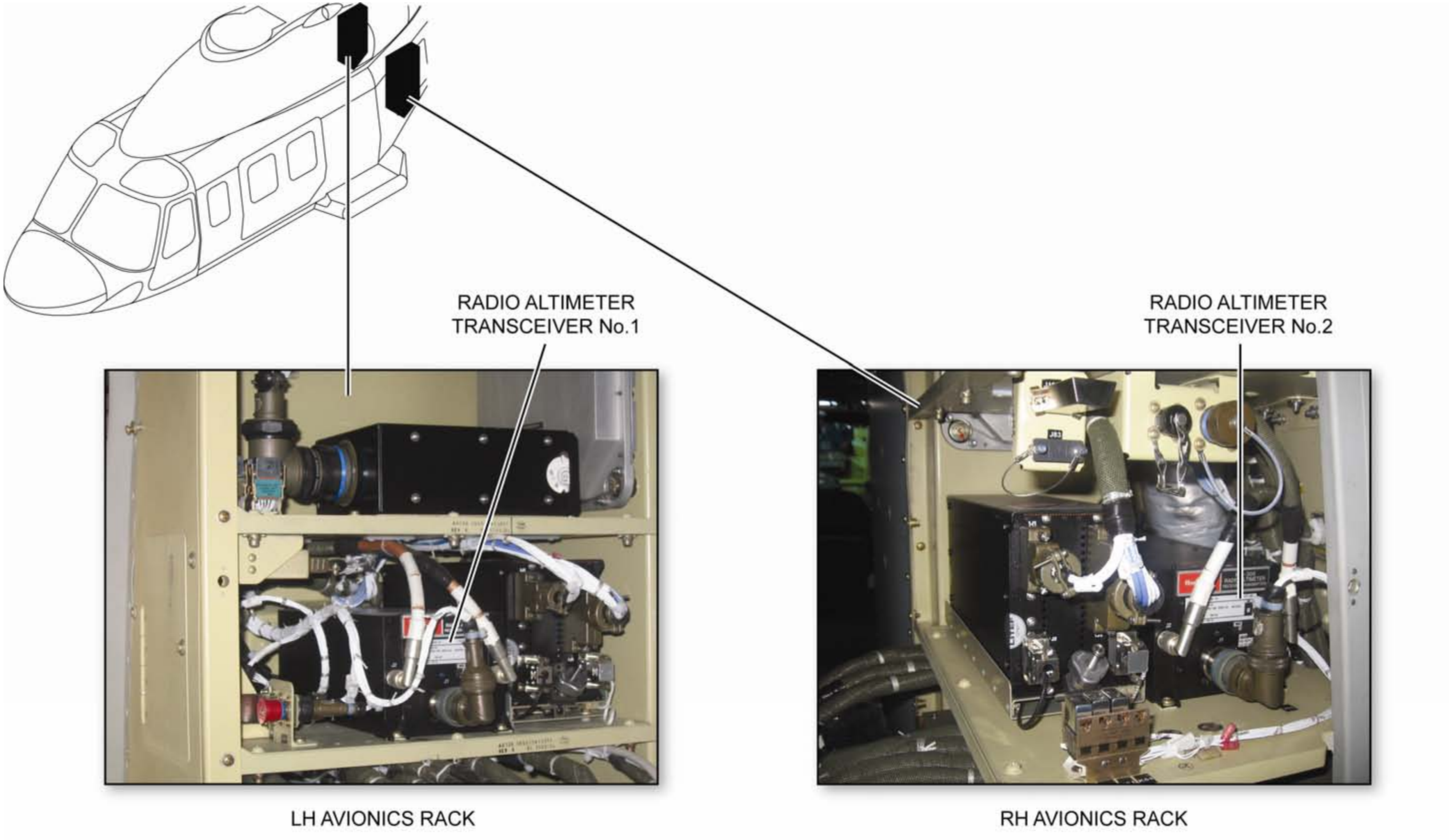
The connections between the two systems are shown in the simplified diagram.

RADIO ALTIMETER TRANSCEIVERS

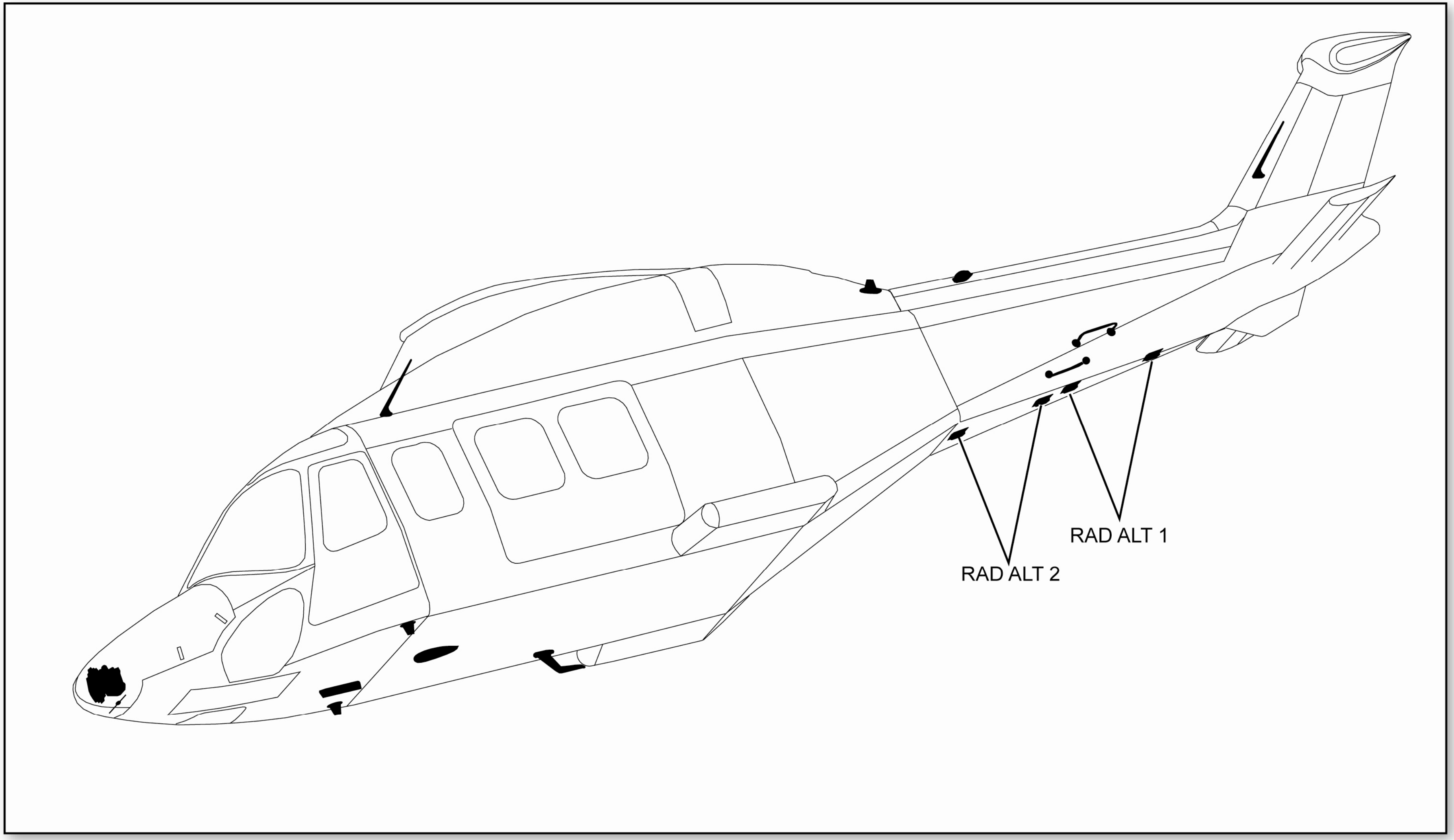
The transceivers contain the 28 V DC power supplies, the transmitter and the receiver circuits. The receiver-transmitter transmits 4300 MHz FMCW signals through the transmit antenna and receives FMCW signals through the receive antenna. The receiver-transmitter measures the time interval between the transmitted and received FMCW signals to calculate the aircraft's altitude AGL (Above Ground Level).

ANTENNAS

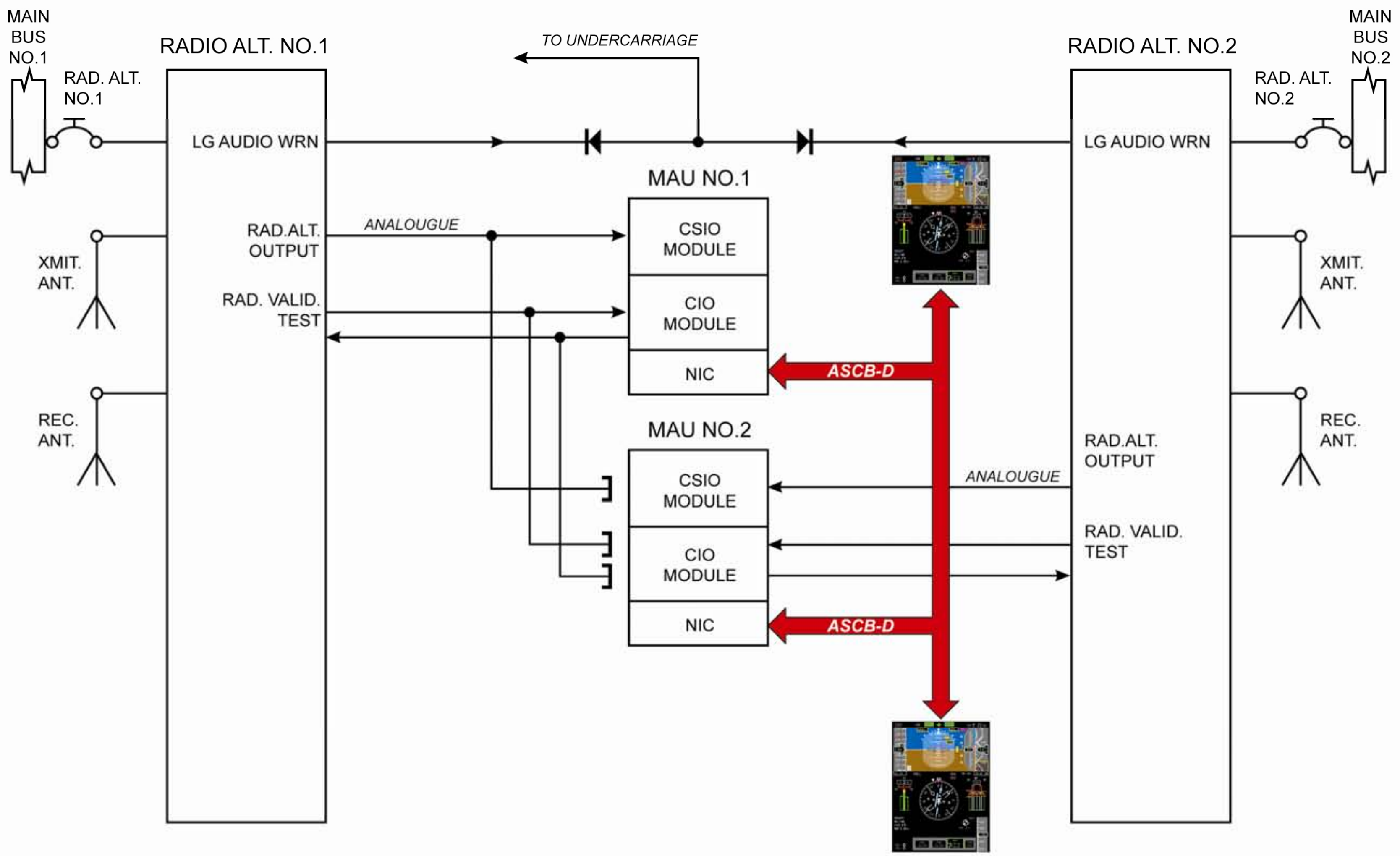
The receive and transmit antennas are low profile, rectangular-shaped antennas installed on the bottom of the tail boom. The location of the antennas lets each antenna to operate at the normal limits of pitch and roll.



RADIO ALTIMETER INSTALLATION



RADIO ALTIMETER ANTENNAS



RADIO ALTIMETER - SCHEMATIC

RADIO ALTIMETER - CONTROLS AND INDICATORS

1. DH knob on Remote Instrument Controller (RIC)

   ……………….. when rotate, it allows to change the decision height display on the on-side PFD from 20 to 2500 ft

2. PUSH TEST push-button

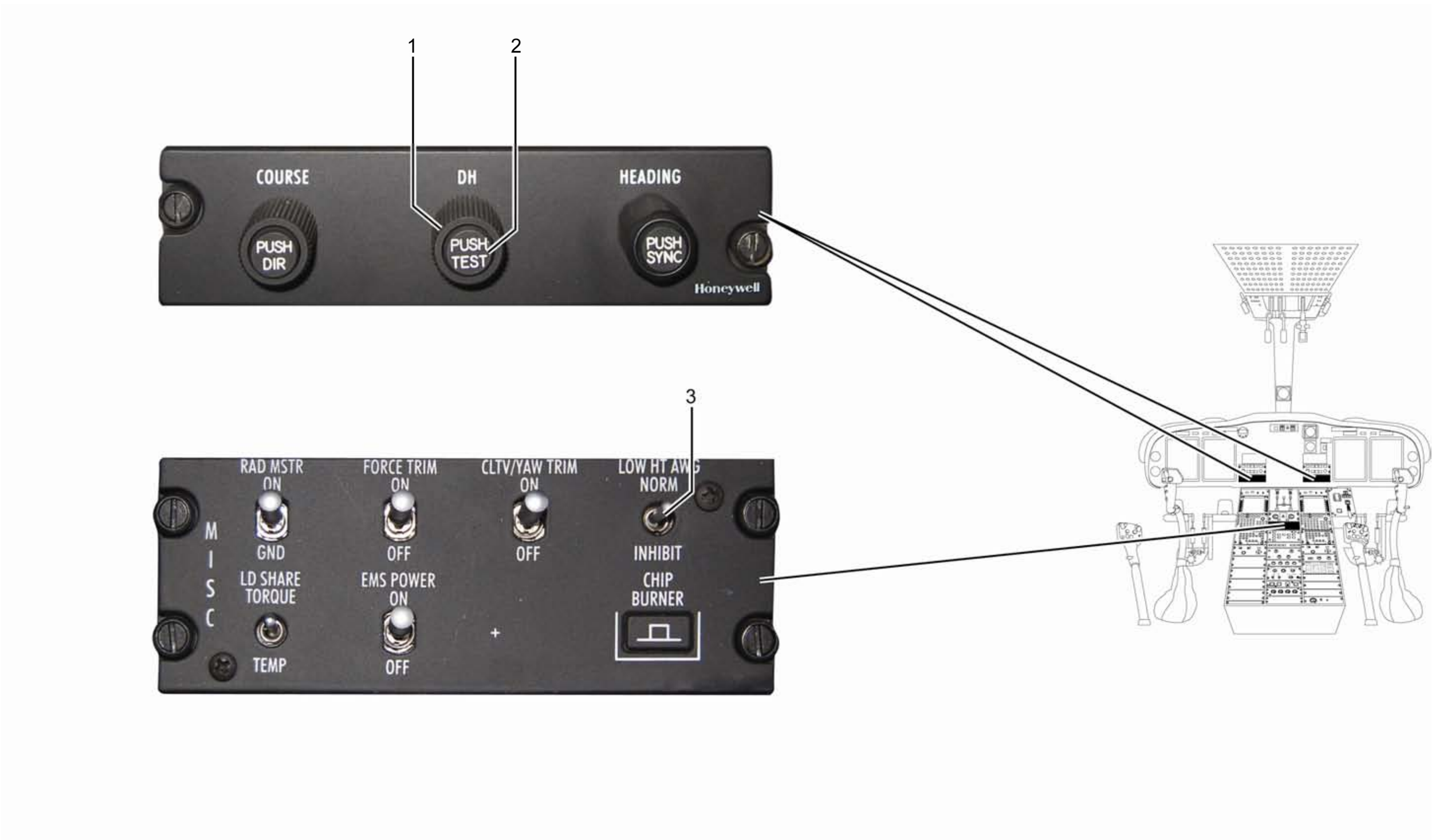
   PRESSED …. execute the built-in test for the radio altimeters. The display pointer indicates 100 ± 10 ft and a TEST flag in inverse video (displayed on the radio altimeter tape) is represented until the button is released. Also the amber messages RAD1 and RAD2 are visible during the test

   NOTE The test can be done both on ground and in flight.

3. AWG switch on MISC control panel

   NORM ..…….. the warning message ONE FIFTY FEET is audible if the helicopter descends below 150 ft AGL

   REGRADE …. the warning message ONE FIFTY FEET is suppressed.



RADIO ALTIMETER – CONTROLS AND INDICATORS

RADIO ALTIMETER TEST RESULT

RADIO ALTITUDE INDICATORS

- RADIO ALTITUDE DIGITAL READOUT

  A digital readout for radio altitude is displayed for altitudes less then 2500 ft. If the radio altitude tape is displayed, the digital readout is incorporated into the tape pointer. The radio altitude information is removed from view above 2500 ft AGL. If the tape is not present, the digital readout is displayed below the triple tachometer display. Brown shading of the RA tape is present below zero (0) ft. Gray shading is present above zero (0) ft.

- RADIO ALTITUDE SOURCE SELECTION

  The PFD automatically selects the source for the radio altitude data. Each PFD displays radio altitude data from the on-side MAU, if that data is valid. If the on-side radio altitude data is not valid, the radio altitude data come from the off-side MAU.

  NOTE: When one radio altimeter is installed, it is wired to both MAU. When two radio altimeters are installed, each is wired to the on-side MAU.

- RADIO ALTITUDE SOURCE ANNUNCIATOR

  When two radio altimeters are installed and a PFD is displaying data from the off-side radio altitude sensor, the sensor is annunciated as either RAD1 or RAD2.

- RADIO ALTITUDE MISCOMPARE

  When a radio altitude miscompare is detected by the monitoring software, a RAD2 to RAD to RAD2 annunciator flashes for the first 6 seconds and then shows RAD steadily above the radio altitude tape. Once the miscompare is no longer detected, the annunciator is removed. The miscompare function is performed whether there are one or two radio altimeters installed. When only one radio altimeter is installed, the miscompare function compares the same data from the two MAU.

CAUTION

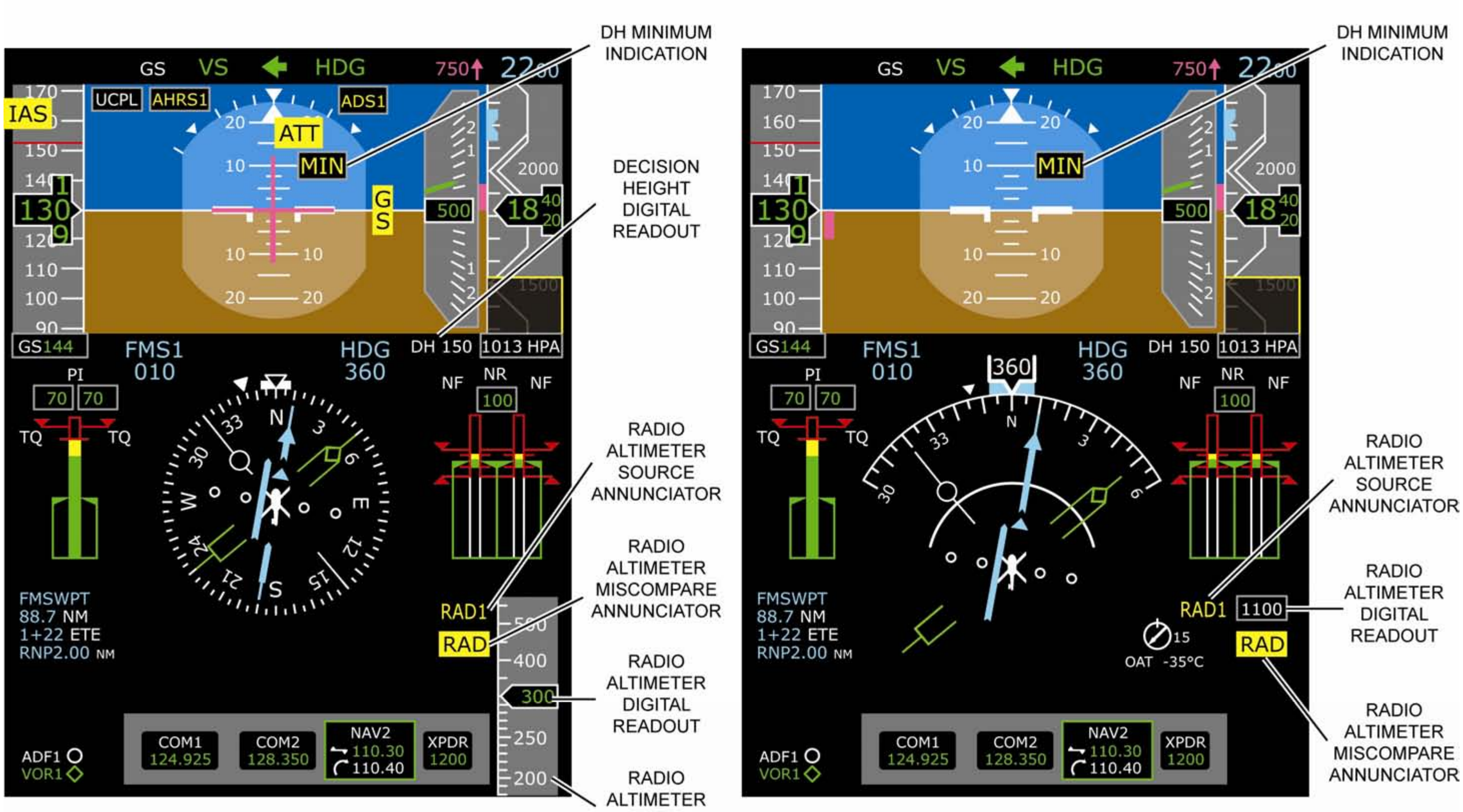
If radio altitude information is lost, a RAD annunciator replaces the radio altitude digital readout in the center of the altitude tape.

CAUTION

The radio altitude display during the radio altitude test procedure is reliable.

WARNING

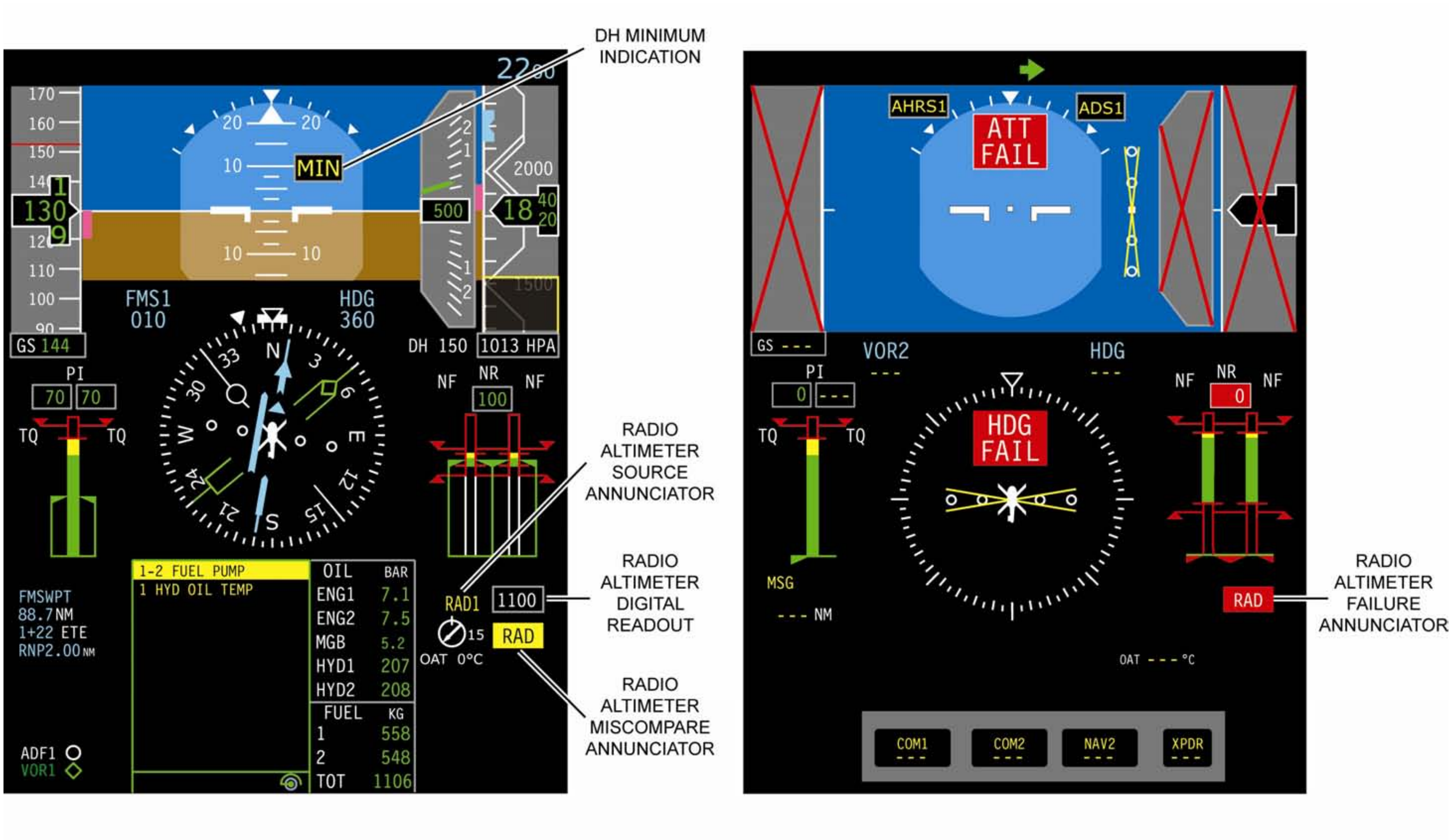
The radio altitude measures absolute altitude above terrain directly beneath the helicopter. It does not measure absolute altitude above terrain in front of the helicopter.



PRIMARY FLIGHT DISPLAY - COMPASS MODE

PRIMARY FLIGHT DISPLAY - ARC MODE

RADIO ALTIMETER INDICATORS (1 OF 2)



RADIO ALTIMETER INDICATORS (2 OF 2)

DECISION HEIGHT (DH)

The DH is displayed as a digital readout located to the right. The DH is set by the Display Controller. The data range corresponds with the radio altitude range (< 2500 ft). The DH display is removed for settings below 20 ft.

NOTE. A loss of valid DH setting from the DC function results in the DH display is amber dashed.

DH MINIMUM INDICATION

During descent, when radio altitude is equal to DH + 100 ft, an empty black box with an amber outline is displayed in the upper right of the attitude sphere. When radio altitude is equal to or less than the DH setting, a MIN annunciator appears in the black box. The MIN annunciator shares the location with the VTA (Vertical Track) annunciator. The MIN annunciator is inhibited on the ground and through climb out until radio altitude is greater than DH + 100 feet.

NOTE. A loss of valid radio altitude information on ASCB-D bus or valid DH setting from the DC inhibits the MIN annunciator.

BAROMETRIC ALTIMETER LOW ALTITUDE ALERT DISPLAY

The Low Altitude Awareness Display (LAAD) is an area of brown shading with amber line limits. The LAAD advances or recedes vertically along the altitude tape with changes in absolute altitude below 550 ft AGL. All data for the LAAD is derived from the radio altitude.

DECISION HEIGHT

HORIZONTAL SITUATION DISPLAY

The horizontal situation display consists of elements that give essential information to pilot using graphics, text and symbols. The Horizontal Situation Indicator (HIS) has three distinct formats as selected by the Display Controller (DC):

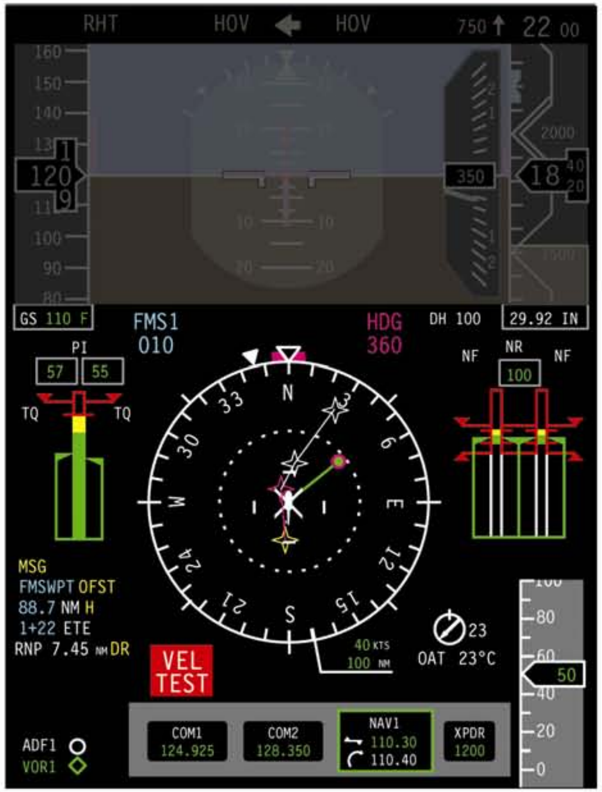
- the FULL COMPASS FORMAT presents a 360° display overlaid with navigation data given by the bearing pointers and course deviation indicator
- the HOVER FORMAT presents the aircraft longitudinal and lateral velocities with the FMS map on a 360° display. The hover display mode gives the pilot a display of the aircraft longitudinal (along the heading) and lateral (across the heading) velocities and a flight director velocity reference bug. The hover display permits overlay of the FMS map data to give situational awareness during the hover. The hover page is selected by toggling the HSI button until the hover page is displayed. It is automatically displayed when either the hover (HOV) or mark on target (MOT) flight director modes are selected
- the ARC FORMAT shows a section of the full compass that is 45° either side of the current heading. It has navigation data supplied by bearing pointers and the CDI. There are four functions that can be displayed in the ARC mode that cannot be displayed in the full compass mode. They are weather, TAWS, TCAS, and flight plan.

Color	Condition
Cyan	Selected NAV source when not coupled to the flight director. Color priority No. 3.
Magenta	Selected NAV source when coupled to the flight director. Color priority No. 2.
Amber	Source annunciators when pilot and copilot have selected the same NAV source. Color priority No. 1.

NAV source color scheme

Refer to PRIMUS EPIC® PILOT'S GUIDE for a complete description of the three display formats.

FULL COMPASS FORMAT



HOVER DISPLAY FORMAT

ARC FORMAT

HORIZONTAL SITUATION INDICATOR (HSI) – DISPLAY FORMATS

HORIZONTAL SITUATION INDICATOR (HSI)

The main properties of the symbols depicted on the HSI are the following:

- AIRCRAFT SYMBOL

  A fixed aircraft symbol helps to visualize the position of the aircraft in relation to horizontal navigation information. In the full compass display the aircraft symbol is in the center of the compass.

- HEADING AND DIGITAL READOUT

  The aircraft heading is indicated by movement of the compass rose with respect to the lubber mark at the top of the compass. Left turns rotate the compass rose clockwise (CW) while right turns rotate the compass rose counterclockwise (CCW). In the arc compass mode, the aircraft heading is numerically displayed inside the heading indicator.

- LUBBER MARK

  The aircraft heading is indicated by movement of the compass rose with respect to a lubber mark at the top of the compass. In full compass mode, an reversed triangle is displayed at the top of the compass to give a lubber. In arc mode, a box with a digital readout of current heading is displayed. The bottom of the box is notched to indicate the lubber line. The notch and the inverted triangle are designed to fit inside the heading bug when they are aligned.

- HEADING SOURCE

  The heading source is annunciated when both pilots have the same source selected, or when directional gyro (DG) mode is selected. When the on-side heading source is displayed on each PFD, no source annunciator is displayed unless the DG mode has been selected. In this case, DG 1 annunciates the copilot PFD and DG 2 annunciates the pilot PFD. Selecting AHRS1 with the reversion controller has MAG1 or DG 1 displayed on each PFD. Selecting AHRS2 has MAG2 or DG 2 displayed on each PFD. Amber dashes replace the digital readout when the selected heading source becomes unreliable. The heading source annunciator is displayed following an AHRS failure.

- HEADING SELECT BUG

  The heading bug is a rectangle with a triangular notch that move 360° around the compass rose to select a desired aircraft heading. When heading select is engaged, the heading bug is displayed in magenta; while a cyan heading bug indicates the flight director is not coupled. The heading select knob of the Remote Instrument Controller (RIC) allows to change the position of the heading bug along the compass rose. When the flight director is in heading (HDG) mode, commands are generated that align and maintain the aircraft on the heading selected by the bug.

- YAW HEADING HOLD REFERENCE BUG

  The yaw heading hold reference bug is displayed on the HSI heading scale tape when the flight director indicates

that the yaw heading hold is functional. A loss of valid heading information from the AHRS or a loss of valid yaw heading hold reference information from the priority autopilot causes the yaw heading hold bug to be removed.

- HEADING DIGITAL READOUT

A digital readout shows the heading currently indicated by the bug. The heading bug and digital readout are magenta, alerting the pilot that the flight director HDG mode is engaged and reliable. The cyan heading bug and digital readout alert the pilot that the flight director HDG mode is not engaged.

- DRIFT ANGLE POINTER

The white, triangular, drift angle pointer is displayed to indicate ground track angle necessary to maintain course. A loss of reliable heading information from the AHRS, or track angle information from the FMS removes the drift angle pointer.

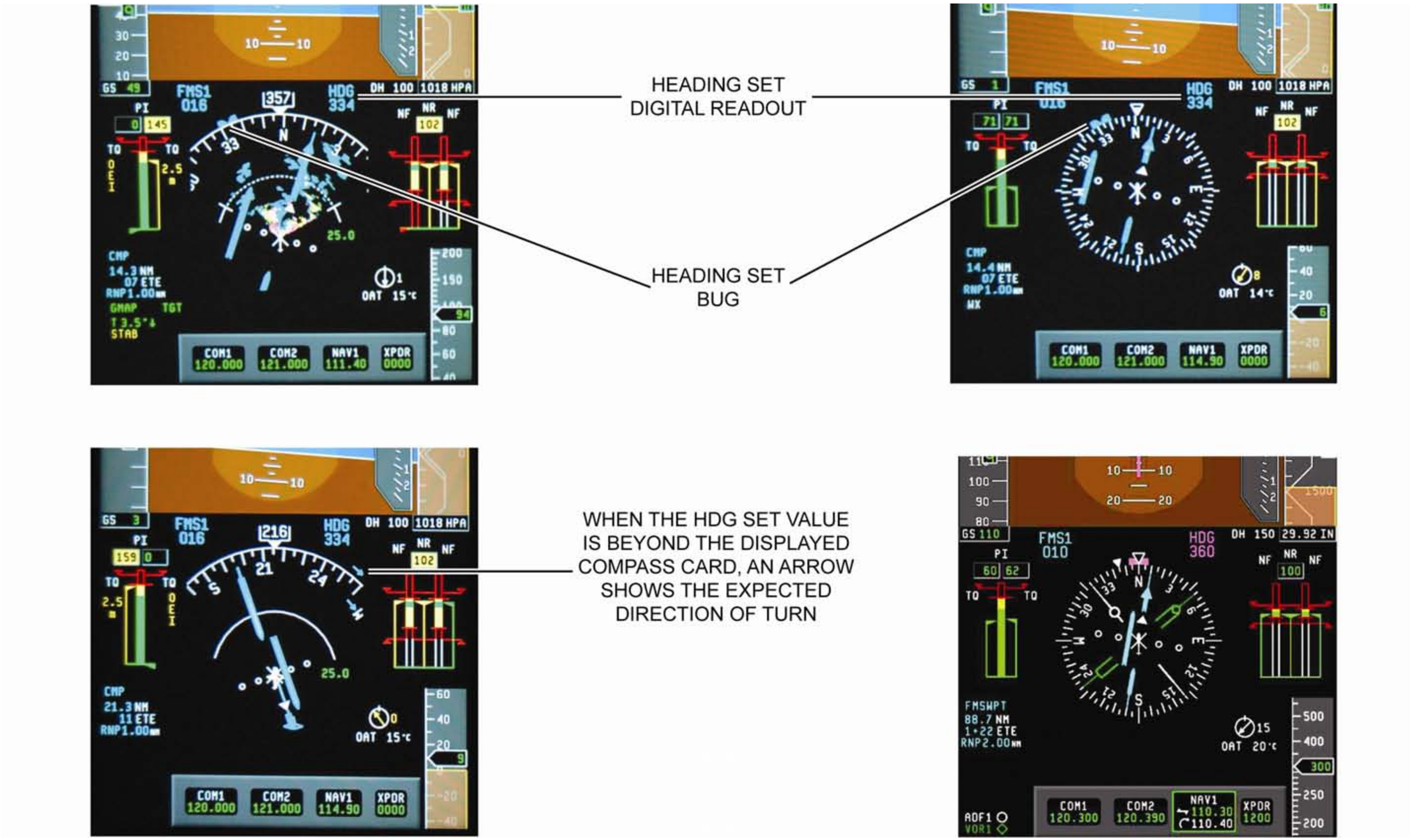
- HEADING MISCOMPARE ANNUNCIATOR

A 10° miscompare between the two AHRS displays the HDG miscompare annunciator. The annunciator is removed when a heading miscompare is no longer detected. A loss of reliable heading information from the AHRS removes the HDG miscompare annunciator.

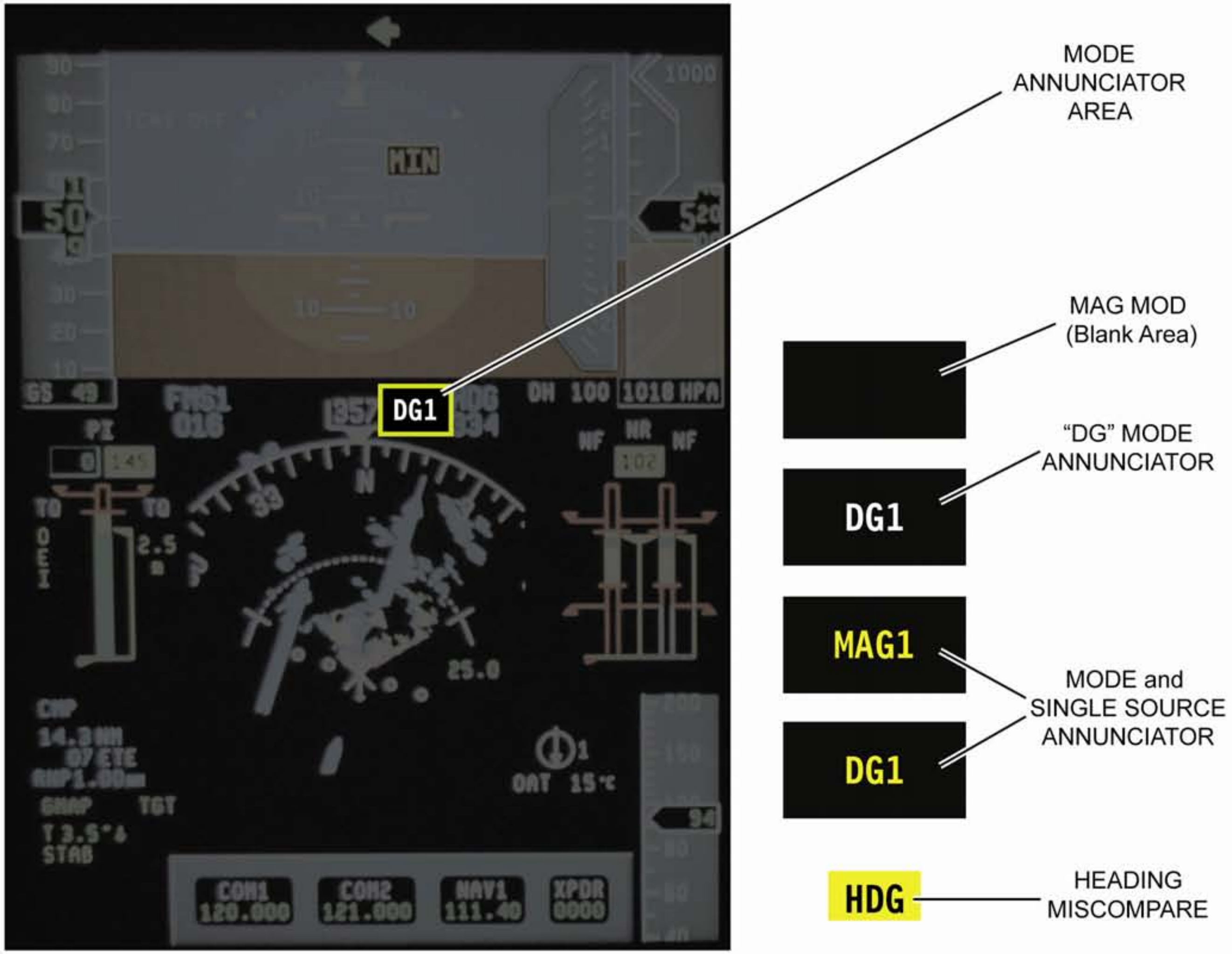
- COURSE POINTER

The course pointer is a segmented needle that is superimposed over the HSI compass display. It represents a selected short-range navigation course (CRS) or FMS desired track (DTK) and the helicopter position relative to it. The dots are always white, and the pointer is cyan when the flight director is not coupled, magenta when the flight director is coupled.

The course pointer pivots around a center point that corresponds to the center of the compass card. The center position is marked by the aircraft symbol. When the FMS is selected, the course pointer is controlled by the FMS, and it is set to the desired track. Once a course is selected, the needle turns with the compass card and aligns with the lubber mark when the helicopter heading is the same as the selected course. The course pointer can be rotated 360° clockwise and counterclockwise using the course select knob on the on-side Remote Instrument Controller (RIC). The tail segment of the needle falls on the reciprocal of the course indicated by the head of the needle or course pointer.


HEADING SET
DIGITAL READOUT
HEADING SET
BUG
WHEN THE HDG SET VALUE
IS BEYOND THE DISPLAYED
COMPASS CARD, AN ARROW
SHOWS THE EXPECTED
DIRECTION OF TURN

HEADING SET

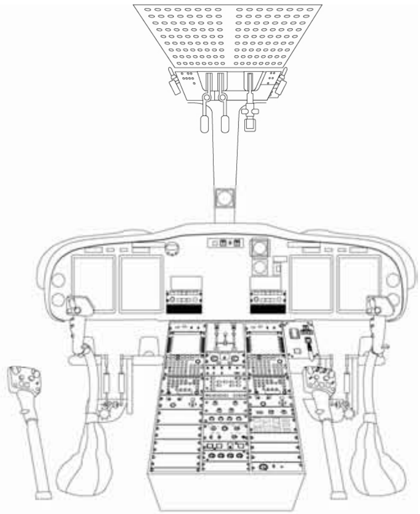


HEADING ANNUNCIATORS

HEADING - CONTROLS AND INDICATORS

1. HEADING knob

PUSH SYSNC button	when pressed synchronizes the selected heading bug to present heading of the coupled attitude and heading reference system (AHRS) or to the heading of the on-side AHRS when the flight director is not coupled. The heading select bug on the two displays are synchronized to the same heading value
	NOTE.
	The HEADING knob controls the heading select bug for both HSI. When the map page is selected on the MFD, the heading bug is displayed.
…………………………...….	When rotated, sets the HDG SET value in step of 1°



HEADING – CONTROLS AND INDICATORS

BEARING POINTER SELECTION

Two different bearing pointers can be displayed on the compass arc. When selected, the bearing pointer operates as a radio magnetic indicator (RMI). They are presented on the display by pushing either the BRG circle or BRG diamond buttons on the display controller. The circular white (O) bearing pointer is used for copilot side sources. The diamond green (◊) bearing pointer is used for pilot side sources.

The display unit receives bearing information from the on-side and cross-side VOR, ADF, and FMS. VOR (magnetic) and FMS (true) data are heading card referenced (absolute) and ADF data is case referenced (relative). When displaying the magnetic referenced heading card against an FMS, the bearing pointer is compensated for magnetic variation.

The circle bearing button allows the selection of various sources following the sequence:

OFF → VOR1 → ADF1 → FMS1→ OFF

The circle bearing pointer is always white. The active source for the circular white bearing pointer annunciator is displayed in white on the lower left corner of the PFD.

The diamond bearing button allows the selection of various sources following the sequence:

OFF → VOR2 → ADF → FMS2→ OFF

The diamond bearing pointer is always green. The active source for the diamond green bearing pointer annunciator is displayed in green on the lower left corner of the PFD.

PRIMARY NAVAID SELECTION

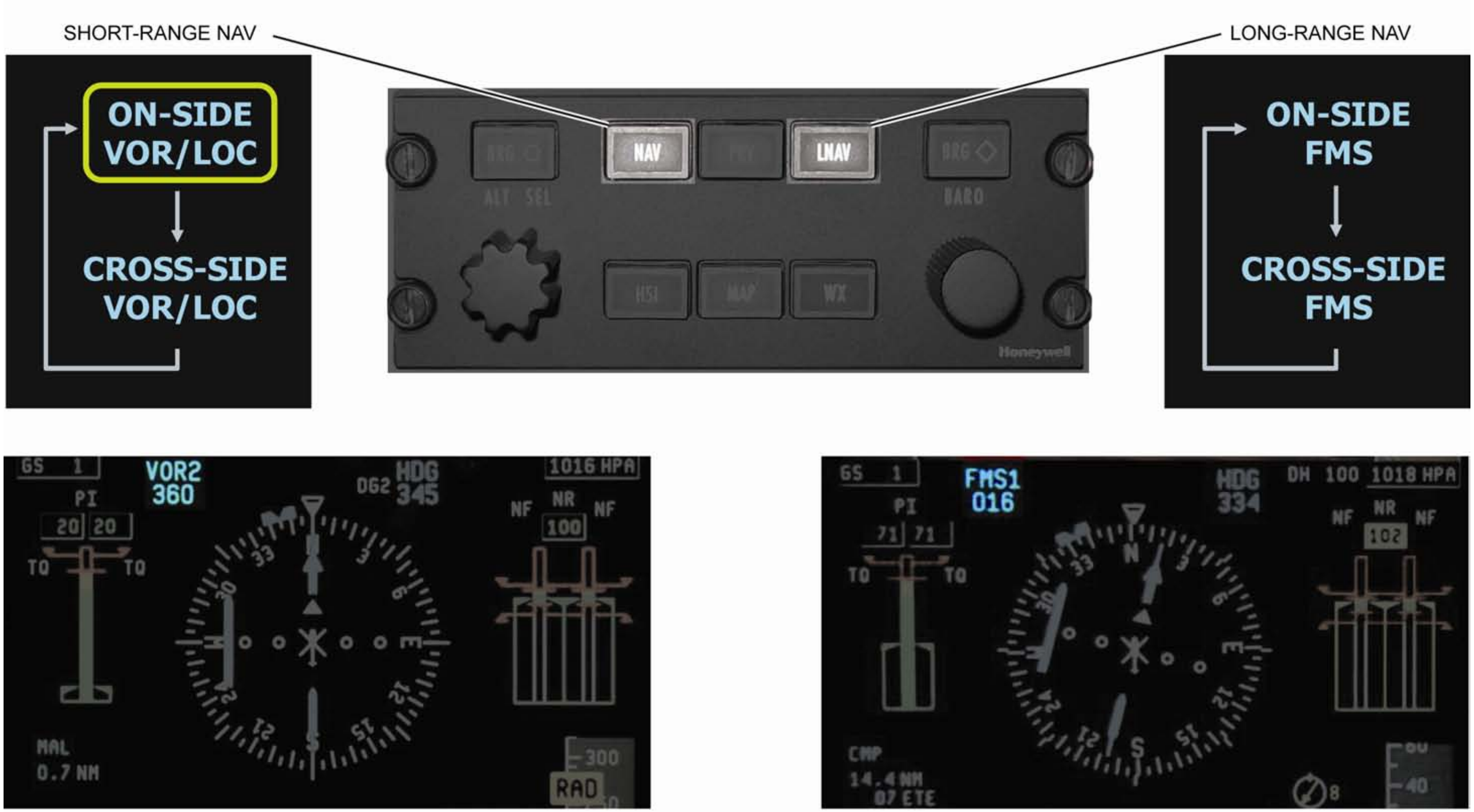
The VOR navigation mode gives short-range navigation aid. The course pointer selects a desired fly to or fly from radial of the VOR station. The VOR navigation mode permits the flight director to give lateral guidance in the roll axis to track the radial selected using the course pointer on the selected PFD.

The NAV button allows to toggle through the available short-range navigation sources displayed on the on-side PFD following the sequence:

ON-SIDE VOR/LOC → CROSS-SIDE VOR/LOC → ON-SIDE VOR/LOC

The LNAV lateral mode intercepts, captures, and tracks the active leg of the FMS flight plan displayed on the on-side PFD. The LNAV button allows to toggle through the available long-range navigation sources displayed on the on-side PFD following the sequence:

ON-SIDE VOR/LOC → CROSS-SIDE VOR/LOC → ON-SIDE VOR/LOC



PRIMARY NAVAID SELECTION

AUTOMATIC DIRECTION FINDER (ADF) – GENERAL

The ADF system supplies data for in-flight navigation, terminal navigation, and area guidance. A narrow band mode is used to reduce noise during navigation. A wide band mode is used to improve clarity when listening to voice signals.
The optimal VOICE and bearing reception are under the following modes:

- ANTENNA (ANT): receives ADF station signal and does not compute bearing
- ADF: receives ADF station signal and computes relative bearing to station
- VOICE: opens IF bandwidth for improved audio fidelity and does not compute bearing
- BEAT FREQUENCY OSCILLATOR (BFO): adds a beat frequency oscillator for reception of CW signals.

The ADF audio is transmitted from the digital audio bus to each audio panel in the system.

When the ADF system is used with the electronic flight instrument system (EFIS) display, the ADF system gives the radio-bearing relative to the rotorcraft heading.

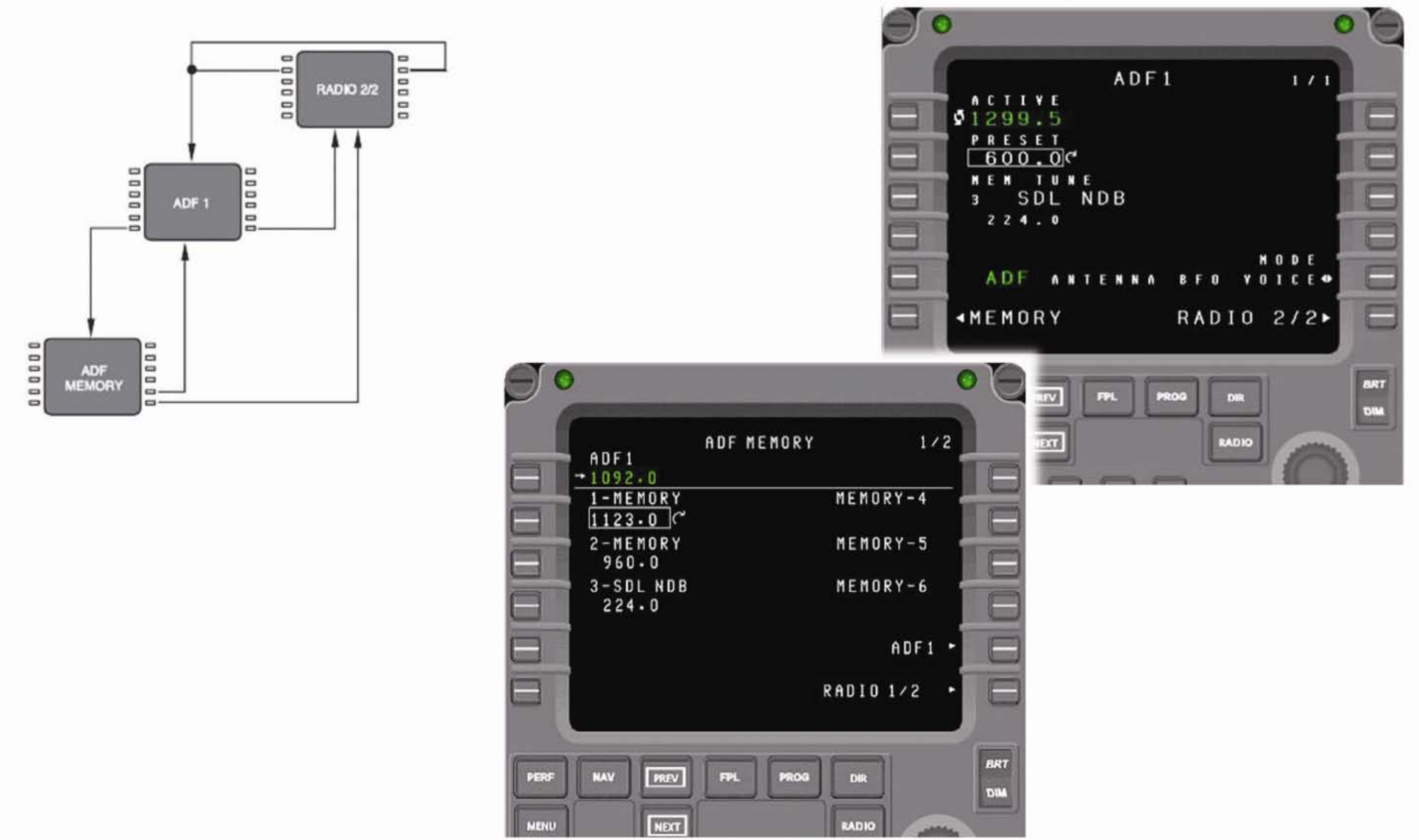
The ADF main components are

- one ADF module (installed is in the MRC 2 pilot side)
- one ADF antenna

When the second ADF is installed, it is identified as ADF 1 and the originally installed ADF is identified as ADF 2 and is installed in the MRC 1 (copilot side). The second ADF antenna is installed on the bottom center fuselage, behind the first ADF antenna.

The pages associated with the ADF system are shown in the figure.

AUTOMATIC DIRECTION FINDER (ADF)



ADF DETAILS AND MEMORY PAGES

VOR / ILS / MB (VHF-NAV) AND DME – GENERAL

The navigation and communication subsystem includes:

- VOR/ILS (VHF Omni-Directional Radio Range /Instrument Landing System) navigation
- DME (Distance Measuring Equipment) navigation

The Modular Radio Cabinets (MRC) contains the VOR/ILS (VIDL), the DME and the Network Interface Module (NIM). The NIM gives processing functionality and interface with the ASCB databus. The NIM interfaces with the audio panels by way of the digital audio and microphone buses. The radios are controlled using the MCDU and PFD radio tuning functions. The pilot intercom function and the control of audio from the radios are given by the audio panels.

The VHF-NAV provides for:

- VOR lateral deviation
- TO/FROM flag
- VOR bearing and audio
- ILS (LOC+GS) deviations and audio
- MARKER BEACON annunciator and audio

The DME is connected to the display system and to the Flight Management System through the Avionic Standard Communication Bus (ASCB-D). The digital audio bus transmits the DME identification audio signals to each audio panel. The DME operates with radio pulses in the frequency range from 960 MHz to 1215 MHz. The airborne system transmits from 1025 MHz to 1150 MHz to a ground station. The receiver frequency range is from 962 MHz to 1213 MHz.

The DME provides for

- distance from the ground station
- ground speed and time-to-station

VHF-NAV AND DME – MAIN COMPONENTS

The VHF-NAV main components are

- two VHF-NAV modules (inside MRC 1 and MRC 2)
- one VOR/LOC antenna
- one GS antenna
- one MB antenna
- three antenna couplers

The DME main components are

- one DME module (inside MRC 2)
- one DME antenna

VOR (VHF OMNI-DIRECTIONAL RADIO RANGE) INDICATORS

- NAV SOURCE

The selected navigation source legend is positioned at the upper left of the compass card immediately above the digital course readout. The NAVAID available for display are: VOR1, VOR2, FMS1, FMS2, LOC1, and LOC2.

The NAV source color scheme is the following:

Color	Condition
Cyan	Selected NAV source when not coupled to the flight director. Color priority No. 3.
Magenta	Selected NAV source when coupled to the flight director. Color priority No. 2.
Amber	Source annunciators when pilot and copilot have selected the same NAV source. Color priority No. 1.

VOR1 legend is displayed when navigation data is from the number one side. VOR2 is displayed when navigation data is from the number two side.

The VOR1 or VOR2 legend is displayed when VOR/LOC mode is selected and the NAV source is not tuned to a localizer frequency.

When the NAV source is tuned to a localizer frequency, the LOC1 or LOC2 annunciator is displayed. The FMS1 or FMS2 annunciator is displayed when the FMS is selected as the NAV source.

- TO/FROM INDICATOR

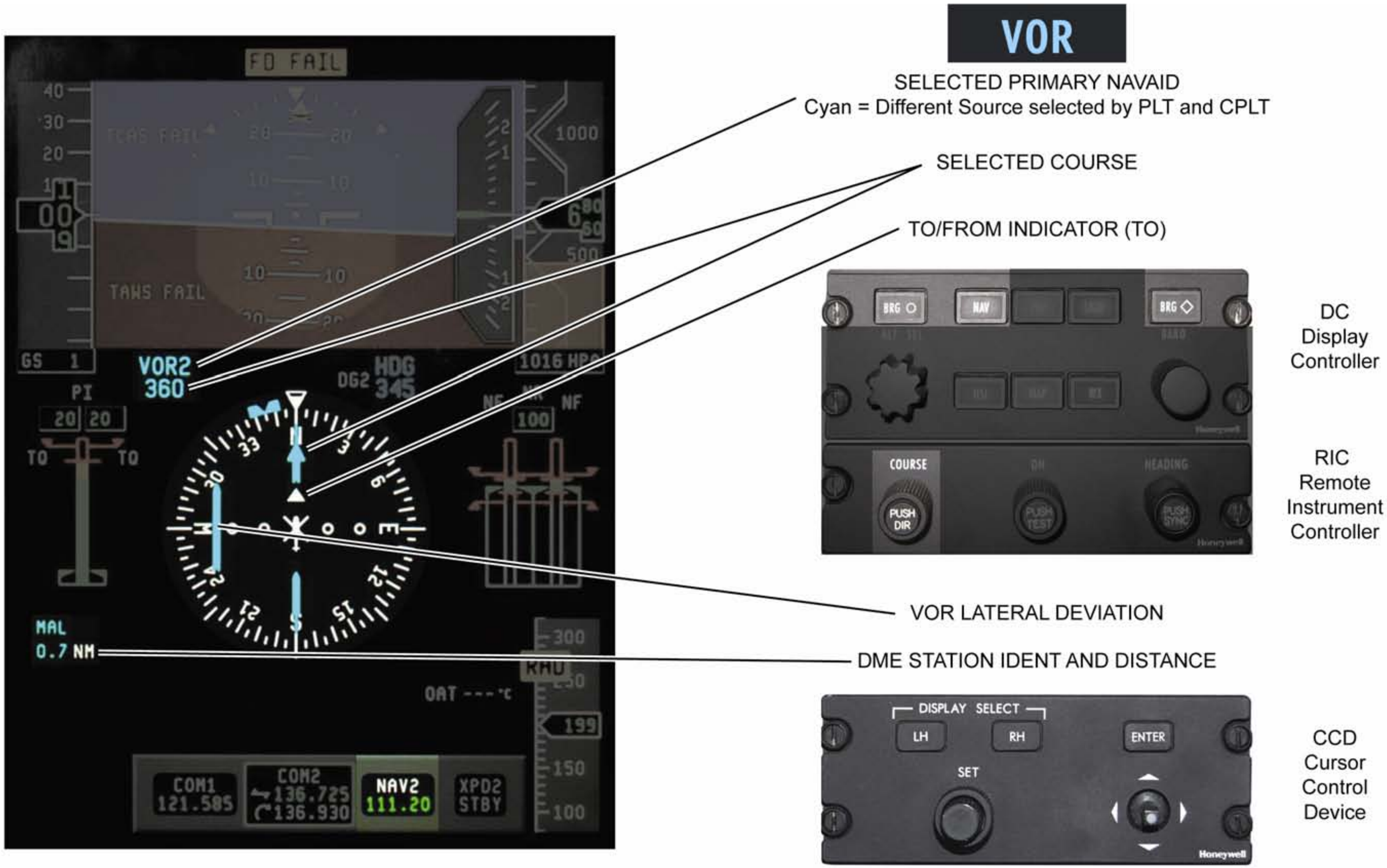
The TO/FROM indicator is a white triangle superimposed on the center line of the course pointer. It is positioned at the nose (TO) or tail (FROM) of the helicopter symbol, and moves with the course pointer. The DU determines whether the helicopter is flying TO or FROM the navigation source when tuned to a VOR. A loss of reliable heading information from the AHRS, or a loss of reliable bearing information from the NAV receiver does not display the TO/FROM indicator or when the FMS is selected does not display the TO/FROM indicator.

- VOR DEVIATION

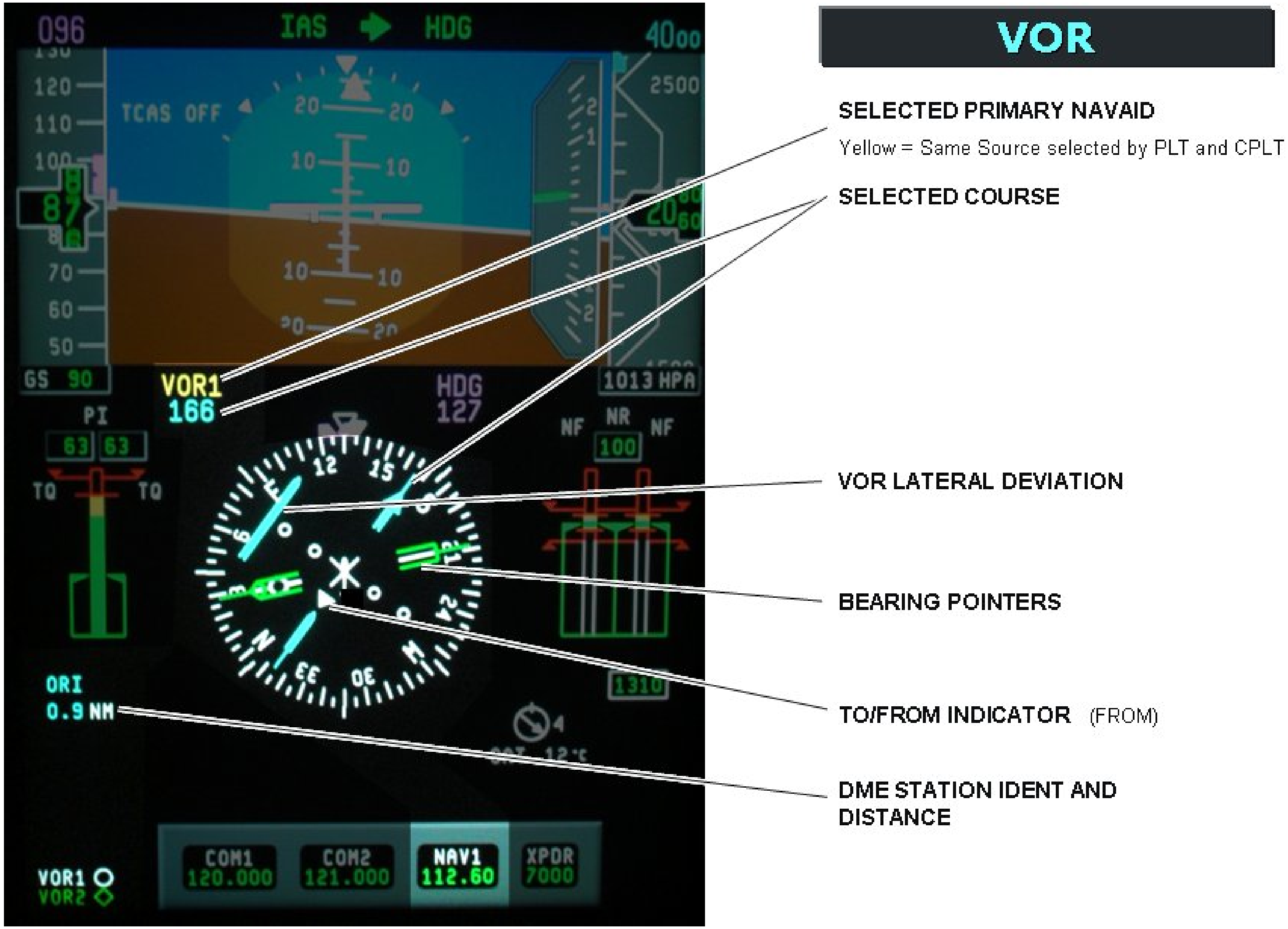
For VOR deviation the pointer parks at ± 20°.

- TO WAYPOINT DISTANCE READOUT

When a VOR/LOC receiver is selected as the primary NAV source and a valid DME station is available, the corresponding DME distance is displayed. When the DME is tuned to a station that is not collocated with the selected VOR, an H is annunciated adjacent to the DME distance to indicate that it is in DME hold mode and is not synchronized with the VOR.



VOR INDICATORS (1 OF 2)



VOR INDICATORS (2 OF 2)

INSTRUMENT LANDING SYSTEM / MARKER BEACONS

- VERTICAL DEVIATION DISPLAY

  The vertical deviation scale is displayed on the right side of the ADI when a localizer is tuned and identified. In this instance, the scale is the glideslope for the selected Instrument Landing System (ILS) approach. When tuned to an identified ILS frequency, the pilot can use the display controller to select glideslope information for display on the vertical deviation scale from either the on-side or cross-side ILS. The scale consists of a rectangle with two dots above and below it where each dot represents a graduation of 1.5° above or below glideslope on an ILS approach.

  The display of the glideslope deviation pointer and scale are not displayed when the localizer back course (BC LOC) is active or armed on the flight director. The glideslope deviation pointer or bug is a truncated cyan triangle that represents the position of the aircraft on the ILS approach in relation to the glideslope.

  The rectangular box on the horizon line represents the position of the aircraft. The glideslope bug shows the pilot where the glideslope is in relation to the helicopter. When the bug is above the box on the display, the helicopter is below glideslope and viceversa. Using the position of the bug in relation to the dots, the pilot can estimate the number of degrees the aircraft is above or below the glideslope. When the bug is immediately adjacent to the box, the aircraft is on glideslope.

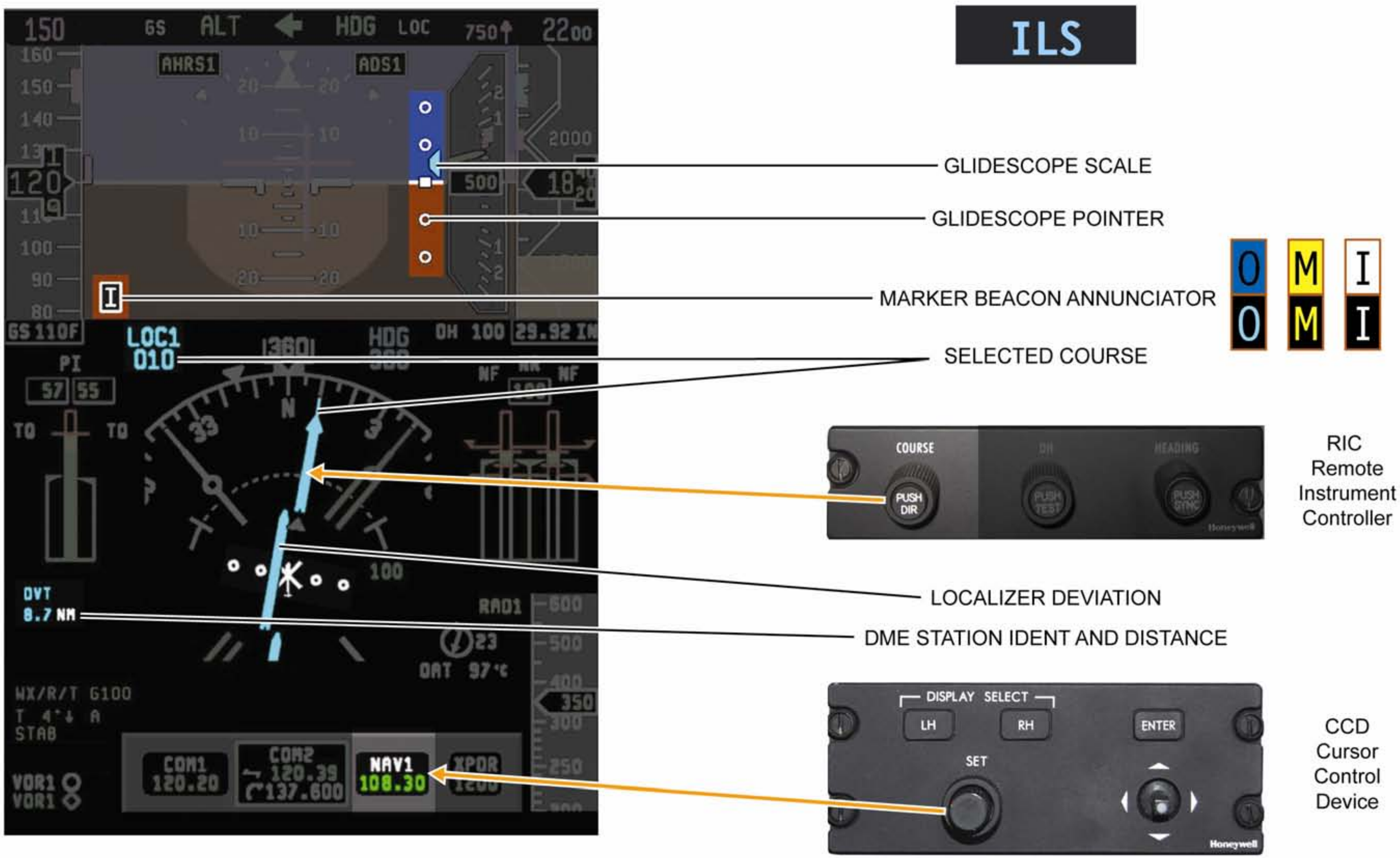
  In helicopters equipped with a flight director, the bug is magenta when the flight director is coupled.

- GS MISCOMPARE

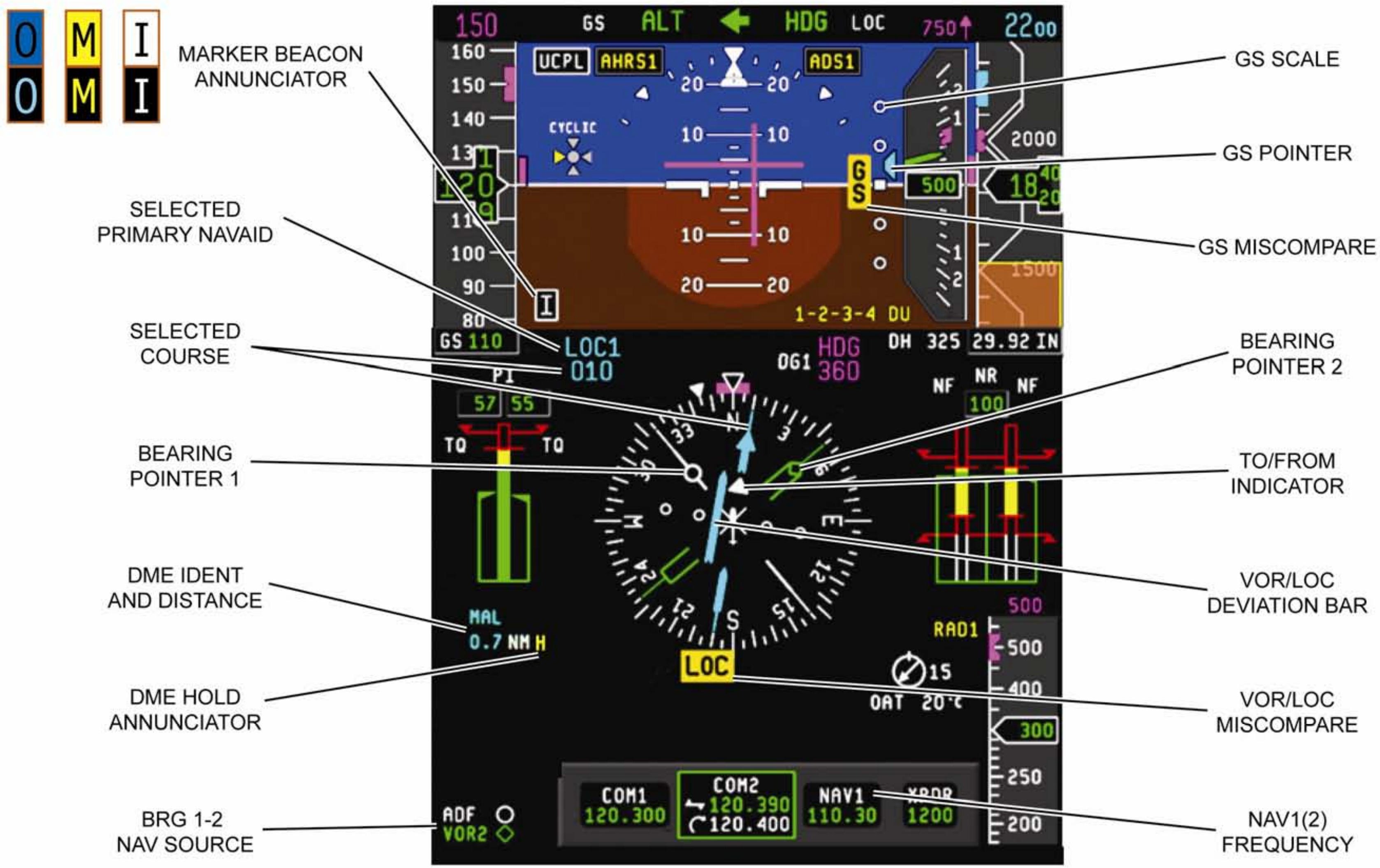
  When a glideslope miscompare is detected, the system alerts the pilot by displaying the GS miscompare annunciator. The loss of valid information from either NAV receiver does not display the GS annunciator.

- MARKER BEACON ANNUNCIATORS

  OUTER (O), MIDDLE (M) and INNER (I), marker beacon annunciators are displayed on the ADI and are controlled by the selected NAV receiver (VOR/LOC). When the selected navigation source is not from the VOR/LOC, the marker beacon generates from the on-side NAV receiver. The pilot is alerted to marker beacon passage by the beacon annunciator.



ILS INDICATORS



VOR / ILS / MB / DME INDICATORS (1 OF 2)

VOR / ILS / MB / DME INDICATORS (2 OF 2)

FLIGHT MANAGEMENT SYSTEM (FMS) – GENERAL

The Flight Management System (FMS) is made by Honeywell. The function of the FMS is to give flight planning capability, navigation information and flight performance data. The FMS manages flight details from takeoff to touchdown. These details include Standard Instrument Departures (SID), Standard Terminal Arrival Routes (STAR) and non precision approaches with missed approaches.

The FMS combines inputs of other aircraft systems (GPS, DME, VOR, AHRS, ADS) to output navigation, lateral and vertical commands and aircraft performance predictions. FMS uses GPS as long range sensor and DME/DME and VOR/DME as short range sensors.

- Primary functions of FMS are position computation and flight planning. These functions work with the associated guidance in both lateral and vertical axes. The FMS Database is essential to these functions in order to easily retrieve, navaids, airways, procedures, airports, other navigation data and store and retrieve waypoints and flight plans.
- Secondary function of FMS are performance including fuel management, time estimates for the flight, vertical navigation estimates, stored flight plans and other functions.

Provided the FMS is receiving usable signals, it has been demonstrated capable of and has been shown to meet the accuracy specifications of:

- VFR/IFR en route domestic, terminal, and instrument approach (GPS, VOR, VOR-DME, TACAN, NDB, NDB-DME, RNAV) operation using the WGS-84 (or NAD 83) coordinate reference datum in accordance with the criteria of AC 25-11, AC 00-314, AC 20-130A, FMS PS 7028826.

Satellite navigation data is based upon use of only the NAVSTAR Global Positioning System (GPS).

Refer to the Honeywell PRIMUS EPIC® PILOT'S GUIDE for a detailed description of the FMS.

FMS DATA BASE

The FMS Database consists of two parts: a navigation database and a custom (or pilot defined) database.

The Navigation Data Base (NDB) cannot be changed by the pilot but, using the custom database, the pilot can customize the FMS by defining waypoints (WP) and flight plans (FP).

NAVIGATION DATA BASE

The navigation database contains worldwide coverage of navaids, airways, Standard Instrument Departure / Standard Terminal Arrival Route procedures, approach procedures, airports, heliports and runways as following:

- navaids (VOR, DME, ILS, MLS, TACAN, NDB)
- airports and heliports
- runways and airways
- SID and STAR
- approaches
- named intersections
- Outer Markers

CUSTOM DATA BASE

The custom database consists of pilot defined WP (up to 1.000) and stored FP (up to 3.000).

FLIGHT PLAN - GENERAL

A flight plan is presented as a series of legs that are sequenced according to their occurrence during the flight. The legs are bounded by waypoints.

During flight, the active flight plan automatically sequences so that the first leg of the active flight plan is the active leg that is referenced to the guidance parameters. Normally, the FMS sequences before the waypoint for an inside turn when the aircraft is on or close to on course: a flyover option is also available to the pilot. If the aircraft is not on course, the normal sequence occurs no later than a point abeam of the waypoint. Some waypoints have unique sequence criteria.

For example, a holding fix is a flyover waypoint. The holding fix should be over flown before entering or exiting holding. Some waypoints in SID and STAR procedures also have unique sequence criteria.

The FMS is programmed to automatically comply with these requirements. There are situations where the sequence criteria cannot be satisfied by the FMS. Under these conditions, the pilot must perform the sequence manually to assist the FMS.

NAVIGATION OPERATION DESCRIPTION

The FMS navigation function is responsible for tuning the NAV and DME radios. The FMS chooses the best NAVAID to tune on the radios for VOR/DME or DME/DME radio position updates.

LATERAL GUIDANCE (LNAV) - GENERAL

Lateral guidance (LNAV) is the function in the FMS that sends commands to the flight guidance computer to laterally steer the aircraft. LNAV guides the aircraft along a flight path set by the pilot including flight plans, departures, holding patterns and approaches.

LNAV maintains the aircraft within an airway or protected airspace. LNAV general rules are the following:

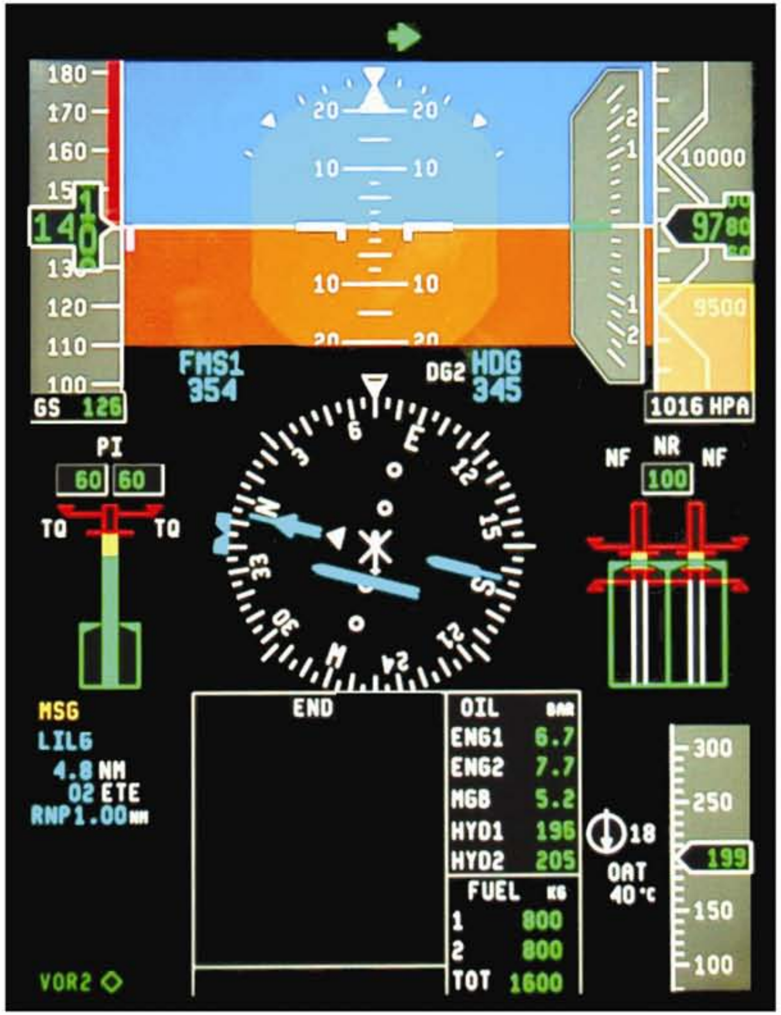
- the FMS must be selected as the navigation source.
- a minimum of one leg must be defined for LNAV calculations.
- LNAV is available for all phases of flight.
- leg sequencing is performed with inside turns between two legs.
- LNAV bank angles are normally as for rate 1. Roll rate is 3° per second.
- the distance shown for each leg of the flight plan accounts for the distance travelled due to the change in course from one leg to the next.
- a lateral track alert (waypoint identifier flashing) is given for each waypoint sequence. The alert is given 30 seconds before starting a turn.

VERTICAL GUIDANCE (VNAV) - GENERAL

Vertical guidance (VNAV) gives a complete vertical profile for the entire flight and is integrated with the lateral flight plan. Using FMS VNAV, the operator can define vertical profile information that may be manually flown by pilot. VNAV cannot be coupled to the FD.

VNAV altitude profile must be flown manually. VNAV computes TOC (Top Of Climb) and TOD (Top Of Descent) points and gives a vertical deviation for path descents. VNAV general rules are the following:

- for VNAV operation, performance initialization and activation of altitude selector (ALT SEL) is required.
- it is recommended to set the altitude selector at the altitude cleared by ATC
- for climbs, only TOC information is provided on the MFD. The TOC is the location at which the aircraft should complete the climb.
- for descents TOD information is provided on the MFD and VPATH (vertical flight path) on the PFD. On MCDU a BOD (Bottom of descent) information is also presented. The TOD is the location at which the aircraft should commence a descent. The BOD is the location at which the aircraft should complete the descent.
- when the altimeter is adjusted to display height above the ground (Q_{FE}) rather than sea level, VNAV should not be used.



PFD Symbology – Full Compass HSI

- DESIRED TRACK
- CROSS TRACK DISTANCE BAR
- TO/FROM
- BEARING
- GROUND SPEED
- ACTIVE WPT IDENT
- WPT DISTANCE
- ETE (Estimated Time Enroute)
- RNP (Required Navigation Performance)
- WIND VECTOR
- ACTUAL TRACK
- Annunciators

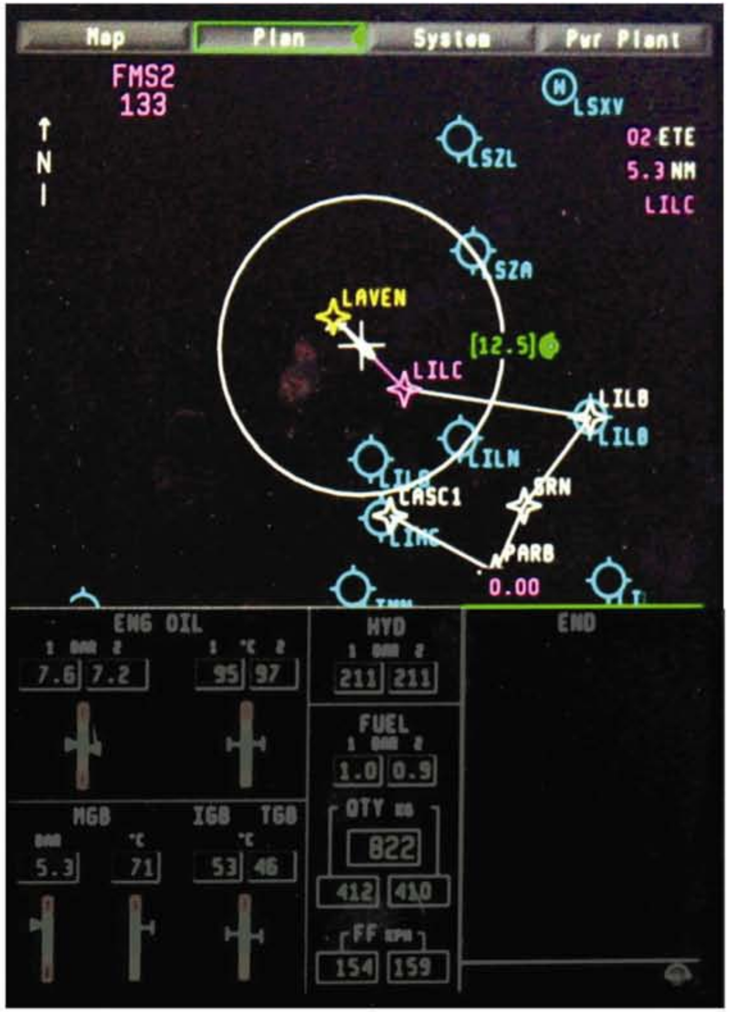
FMS – FULL COMPASS HSI

PFD Symbology – ARC Mode HSI

- Same as Full Compass HSI
- Flightplan

FMS – ARC MODE HSI

MFD Symbology : MAP Format FPL



MFD Symbology : PLAN Format FPL

MFD SYMBOLOGY

A VPATH will be shown during a descent if:

- a vertical constraint is entered or,
- an approach is activated.

WARNING

The pilot must not fly VPATH indication unless:
- when under ATC control
- flying a published approach

or

- during VMC Day conditions

CAUTION

When the vertical profile includes two consecutive slopes with different angles, a discontinuity in the vertical indicator may be shown at the same tansition.

VNAV

FMS DISPLAYS AND CONTROL – GENERAL

Flight management system displays and Flight management system data are presented on the MFD (MAP page), PFD (HSI) and Multipurpose Control Display Unit (MCDU). Displayed data includes:

- a map presentation that shows:
  - radio navigation aids
  - airports
  - waypoints on the active flight plan
  - legs, patterns and approach profiles of the active flight plan
- FMS mode annunciations

Electronic maps integrate route map data with auxiliary navigation data to display the aircraft's situation at any time.

Electronic displays integrate map data with weather radar displays, terrain maps and electronic charts. Control panels used in FMS operation are :

- MCDU (Multipurpose Control Display Unit)
- RIC (Remote Instrument Controller)
- DC (Display Controller)
- GC (Guidance Controller)

MCDU - GENERAL

The MCDU has a color display used to highlight important information but color assignments are not coordinated with MFD and PFD colours.

Assigned Color	Parameter
Cyan	Vertical, Performance, and Atmospheric Data
Green	Lateral, Modes
Yellow	Warnings, FROM Waypoint, Flight Plan Names
Magenta	TO Waypoint
White	Names, Titles and scratchpad messages
Red	Failures

The MCDU display has 14 lines each one of 24 characters. The first line is the *title line* and the fourteenth line is the *scratchpad* that is a working area where the pilot can enter or verify data before line selecting the data into the proper position. Alphanumeric entries are made to the scratchpad using the keyboard. As each key is pushed, the character is displayed in the scratchpad. Information in the scratchpad does not affect the FMS until it is moved to another line on the display. Data is retained in the scratchpad throughout all mode and page changes.

Functions keys are presented directly below the screen. These keys access primary functions, index and page selection.

- PERF Key. Displays page 1 of the performance index. The pilot can select any of the index functions by pushing the respective line select key.
- MENU Key. Displays the MCDU menu page which accesses the maintenance and status information.
- NAV Key. Displays page 1 of the navigation index. The pilot can select any of the index functions by pushing the respective line select key.
- PREV/NEXT Keys. The specific page and number of pages in a particular function or menu display are shown in the upper right corner of the display. Page changes are made by pushing the PREV and NEXT keys. The keys can be held down for repeated page changing.
- FPL Key. Displays the first page of the active flight plan. If no flight plan is entered, the pilot can perform the following: manually create a flight plan; select a stored flight plan; create and store a flight plan.
- PROG Key. Displays the first progress page. This mode shows the current status of the flight. The first progress page displays distance to, the Estimated Time Enroute (ETE) and fuel projection for the TO waypoint, the NEXT waypoint and destination. It also displays the current NAV mode (i.e. GPS, DME-DME, VOR-DME, AHRS), the required and estimated navigation performance, and the navaids that are presently tuned.
- DIR Key. Displays the active flight plan page with the DIRECT, PATTERN, and INTERCEPT prompts. If other than the active flight plan page is displayed when pushing the button, the first page of the flight plan is displayed. If the active flight plan is already displayed when pushing the button, the display remains on the same page with prompts displayed. DIRECT is the primary function. PATTERN and INTERCEPT must be selected at 6L or 6R, respectively.
- RADIO Key. Pushing the RADIO function key results in the display of the RADIO 1/2 page. From this location and also the RADIO 2/2 page, the pilot can tune various radios including COM1, COM2, NAV 1, NAV 2, ADF and set Transponder code.
- FUNCTION KEY PAGING. Function key paging is an option that, when activated (via FLIGHT CONFIG), can be used to advance function pages using additional pushes of the function key instead of the NEXT key. For example, when initially selecting a function key, the MCDU displays page 1 for that function key. Pushing the same function key again displays page 2 for that function.
- LINE SELECT KEYS (LSK). Line select keys are six on each side of the MCDU displays. Data is selected to a line from the scratchpad or viceversa using the line select keys. These keys are identified from top to bottom as 1L through 6L on the left side and 1R through 6R on the right side. The line select keys are the most often used keys on the MCDU. Typical LSK functions are:
  - Direct Access Prompts/Function Selects In the case of an index display, the line select keys are used to select functions from the index. In displays other than index, the bottom line select keys (6L, 6R) are primarily used for direct access to other functions in

the FMS. The FMS prompts the pilot at 6L and 6R for the most likely functions to be selected. An example is the ARRIVAL prompt that is displayed on the active flight plan pages when within 200 NM of the destination. These types of prompts reduce the number of key strokes in order to minimize pilot workload. The pilot can also access functions through the main navigation and performance indices.

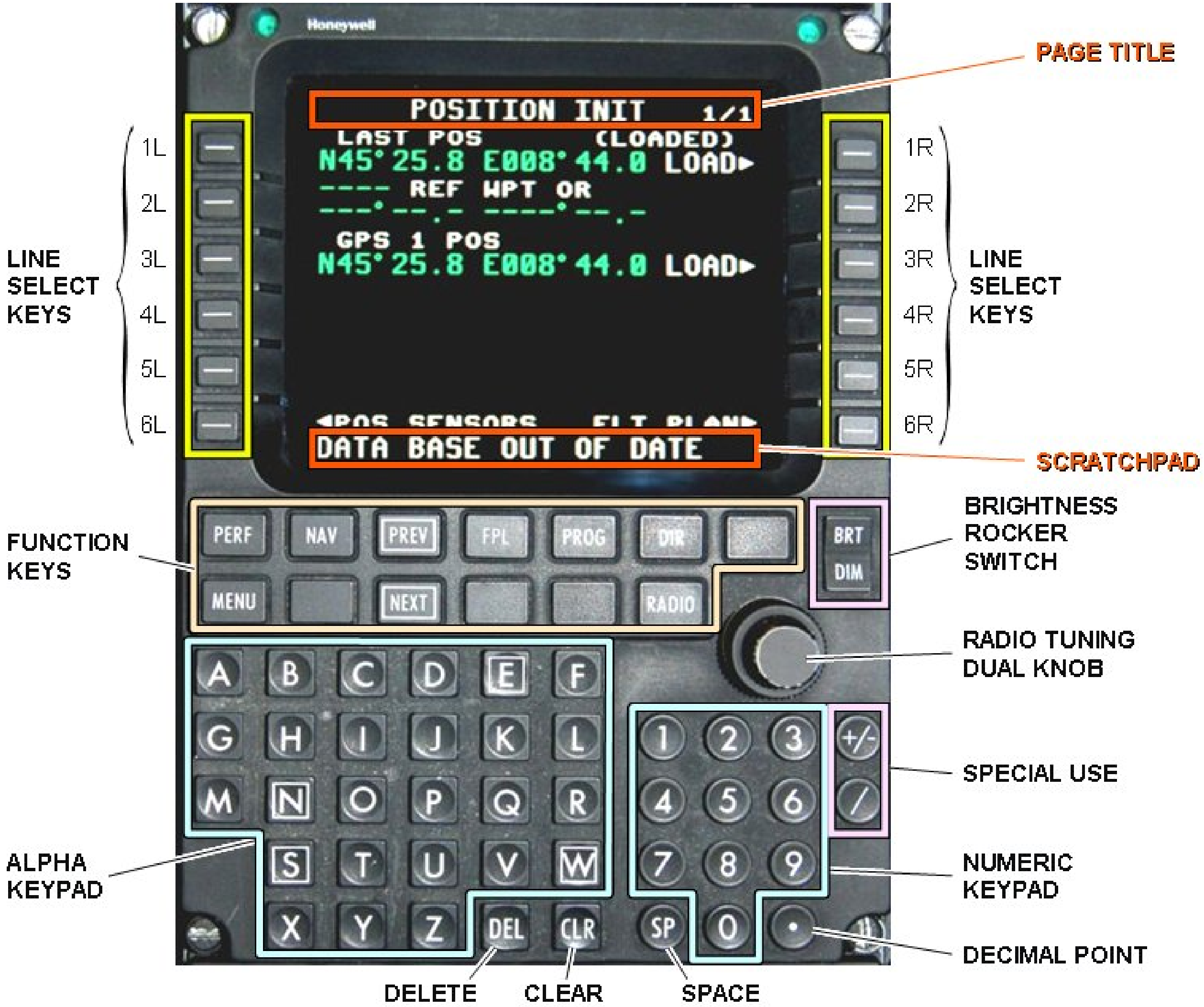
- Transfer Line Data to Scratchpad. If the scratchpad is empty, pushing a line select key transfers the respective line data to the scratchpad.
- Transfer Scratchpad Data to Line Fields. Once data has been entered into the scratchpad either through line selection or manual keyboard entry, it can be transferred to any of the allowable line select fields on a page. To transfer the data, push the key adjacent to the line where the scratchpad data is intended.

- ALPHANUMERIC KEYBOARD. Is used by the pilot for input to the FMS. The alphanumeric keys make entries to the scratchpad only. The following are each represented with a key on the MCDU:
  - letters of the alphabet
  - the numbers 0-9
  - the decimal
  - the plus/minus
  - the space
  - the slash

  The SP (space) key is used to insert a space following a character in the scratchpad. The +/- (Plus/Minus) key is used to enter a - or + in the scratchpad. The initial push of the +/- key results in a - being entered. A subsequent push changes the – to a +. Continued pushing of this key toggles the +/- display.
- CLR KEY. This key performs the following functions:
  - When a message is displayed in the scratchpad, pushing the CLR key deletes the message.
  - When a scratchpad entry begins with an asterisk (*) or pound sign (#), pushing the CLR key removes the entire entry.
  - When an alphanumeric entry is made in the scratchpad, one character is cleared from the scratchpad (from right to left) each time the CLR key is pushed. If the CLR key is held down after the first character is cleared, other characters are cleared, one at a time, until the key is released.
- DEL Key. The DEL key is used to delete items from the FMS. When the DEL key is pushed, *DELETE* is displayed in the scratchpad. The DEL key can be line selected to delete waypoints or other items displayed in the MCDU data fields. When there is a message displayed, the delete operation is inhibited. Delete is also used to return default values after entries have been made. The DEL key can also be used in the scratchpad edit mode. With a dash (-) at the end of the scratchpad

entry, pushing the DEL key deletes the entire scratchpad entry.

- BRIGHTNESS CONTROL. Both manual and automatic (photo sensor) brightness controls are used to increase or decrease the MCDU display brightness. When manually selected, a bright/dim bar is displayed in the scratchpad. The bright/dim bar level is controlled by pushing BRT or DIM. Following manual adjustment, the photo sensors monitor the ambient light and maintain the brightness level of the MCDU display over various lighting conditions. Note that the brightness can be adjusted during evening hours such that, during daylight hours, the display cannot be seen.



FMS CONTROLS – MCDU

FLIGHT MANAGEMENT SYSTEM – CAS CAUTION MESSAGES

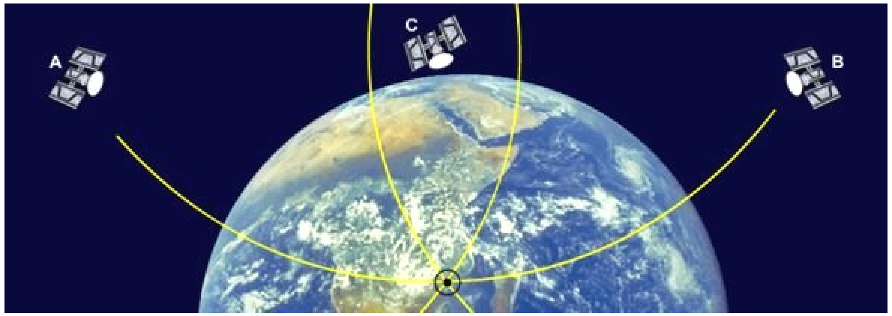
CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
1(2) FMS FAIL	Failure of flight management system. FMS navigation not available	FLIGHT MANAGEMENT SYSTEM FAILURE	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES AVIONICS

GLOBAL POSITIONING SYSTEM – GENERAL

The Global Positioning System (GPS) supplies data for flight guidance and other information during flight. The GPS installed on the AW139 helicopter is a 12-channel GPS receiver that receives from the NAVSTAR GPS satellite constellation.

The GPS interface the Air Data System (ADS) and the Flight Management System (FMS). The GPS modules have the primary function of determining the aircraft position from the signal codes. The output data includes:

- three dimensional aircraft position and velocities
- satellite position
- pseudo range
- delta range data



GPS – MAIN COMPONENTS

The GPS main components are:

- one receiver (that is a GPS module inside MAU 2)
- one antenna

The GPS receiver uses the Commercial Access (C/A) code of the NAVSTARGPS satellite constellation and can operate when selective availability (SA) is activated and deactivated.

GPS – CONTROLS AND INDICATORS

The performance of the GPS can be monitored by the GPS 1 STATUS pages (1/2 and 2/2) and PREDICTIVE RAIM page (1/1). (RAIM = Receiver Autonomous Integrity Monitor calculations).

Predictive RAIM (PRAIM) calculates the estimated value of the Horizontal Integrity Limits (HIL) at some future place and time. The FMS can interrogate the PRAIM function of the GPS through the ARINC 429 interface. However, RAIM integrity performance requirements cannot be selected with the GPS.

GPS CONTROLS AND INDICATORS

GPS – PRINCIPLE OF OPERATION

The GPS receiver operations are transparent to the crew. Each GPS has RAIM (Receiver Autonomous Integrity Monitor) outputs for the current position and time in the form of Horizontal and Vertical Integrity Limits (HIL and VIL). In order to compute RAIM, the GPS must have a minimum of five satellite signals. The FMS does not accept GPS data unless a valid RAIM figure is available.

Therefore the GPS RAIM function assures the integrity of the data because the GPS RAIM function can detect satellite failures. It isolates and removes failed satellites when it is tracking a sufficient number of satellites for measurement redundancy.

The FMS uses predictive RAIM to determine the integrity levels at specific locations/times to support a non precision approach and the flight planning activities of the pilot.

The GPS has the following types of RAIM predictions:

- destination
- alternate waypoint

The destination and alternate waypoint predictions are made at specific locations or they are the Estimated Time Of Arrival (ETA) when the FMS makes the request for flight planning purposes. Satellites can be manually deselected or enabled for destination and alternate waypoint predictions.

GLOBAL POSITIONING SYSTEM – CAS CAUTION MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
GPS FAIL	GPS system failure	GPS FAIL	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES AVIONICS
1(2) FMS/GPS MSCP	Miscompare between FMS and GPS position data	FMS/GPS MISCOMPARE	
FMS/GPS MSCP UNAVL	FMS/GPS miscompare function not available due to FMS or GPS DATA INVALID	FMS/GPS MISCOMPARE UNAVAILABLE	

ATC TRANSPONDER – GENERAL

The Air Traffic Controller (ATC) Transponder (XPDR) module has the encoding and decoding capability to operate in Mode S that allows digital addressing of individual aircraft and it is fundamental for the operation of the Traffic Collision Avoidance System (TCAS).

TCAS is an independent airborne system that does not rely on air traffic control (ATC) for control or coordination for traffic separation. It is designed to act as a backup to the ATC system and the see and avoid principle. It detects unsafe traffic conflicts with other transponder-equipped aircraft and assists the flight crew in avoiding intruders inside a protected airspace. This is done by interrogating surrounding aircraft with Mode A, Mode C, and Mode S transponders, tracking the responses, and issuing advisories to the flight crew of the vertical separation from intruders.

ATC XPDR – MAIN COMPONENTS

The ATC XPDR main components are:

- the XPDR module (inside MAU 2)
- the XPDR antenna

ATC XPDR – PRINCIPLE OF OPERATIONS

The ATC XPDR modes of operation are:

- STANDBY: ready but not replying
- ALT OFF: Transponder modes A and S, no altitude reporting
- ALT ON: Transponder modes A, C and S, altitude reporting enabled
- TA: TCAS is enabled, if installed

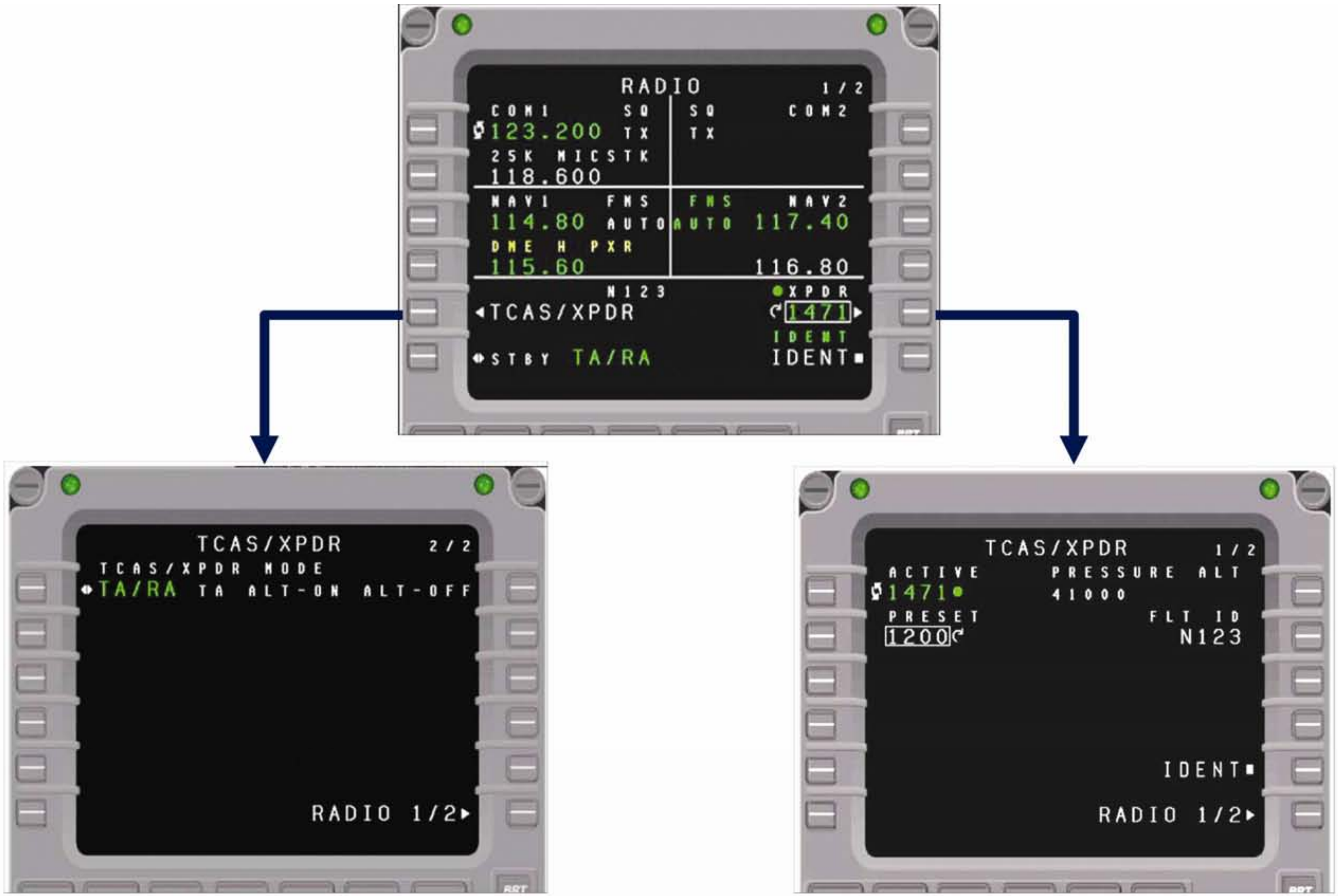
ATC XPDR – CONTROLS AND INDICATORS

The Electronic Display System (EDS) receives data from the XPDR module through the Network Interface Module (NIM) on the Avionics Standard Communications Bus (ASCB-D). The EDS displays on the Radio NAV window of the PFD the transponder code, indicating which XPDR is active. When the selected transponder is in standby mode the transponder code is replaced with the STBY annunciation.

In the RADIO page 1/2 of the MCDU, XPDR code, status and operating mode are displayed.

The TCAS/XPDR 1/2 page displays XPDR code (active and preset), selection (XPDR1 and XPDR2), barometric altitude data and flight ID.

The TCAS/XPDR 2/2 page displays the transponder operating mode selection.



ATC XPDR CONTROLS AND INDICATORS

ADI STANDBY INDICATOR – GENERAL

One ADI STANBY used as backup contains an inertial measurement cluster that eliminates the need for an external gyro system. It has no moving parts and it could replace all the AHRS gyro functions in case of failure.

The ADI STBY provides for

- Pitch Angle
- Roll Angle
- Slip/Skid (Lateral acceleration)
- Heading
- Airspeed
- Barometric Altitude
- Vertical Speed

The ADY STBY receives and displays

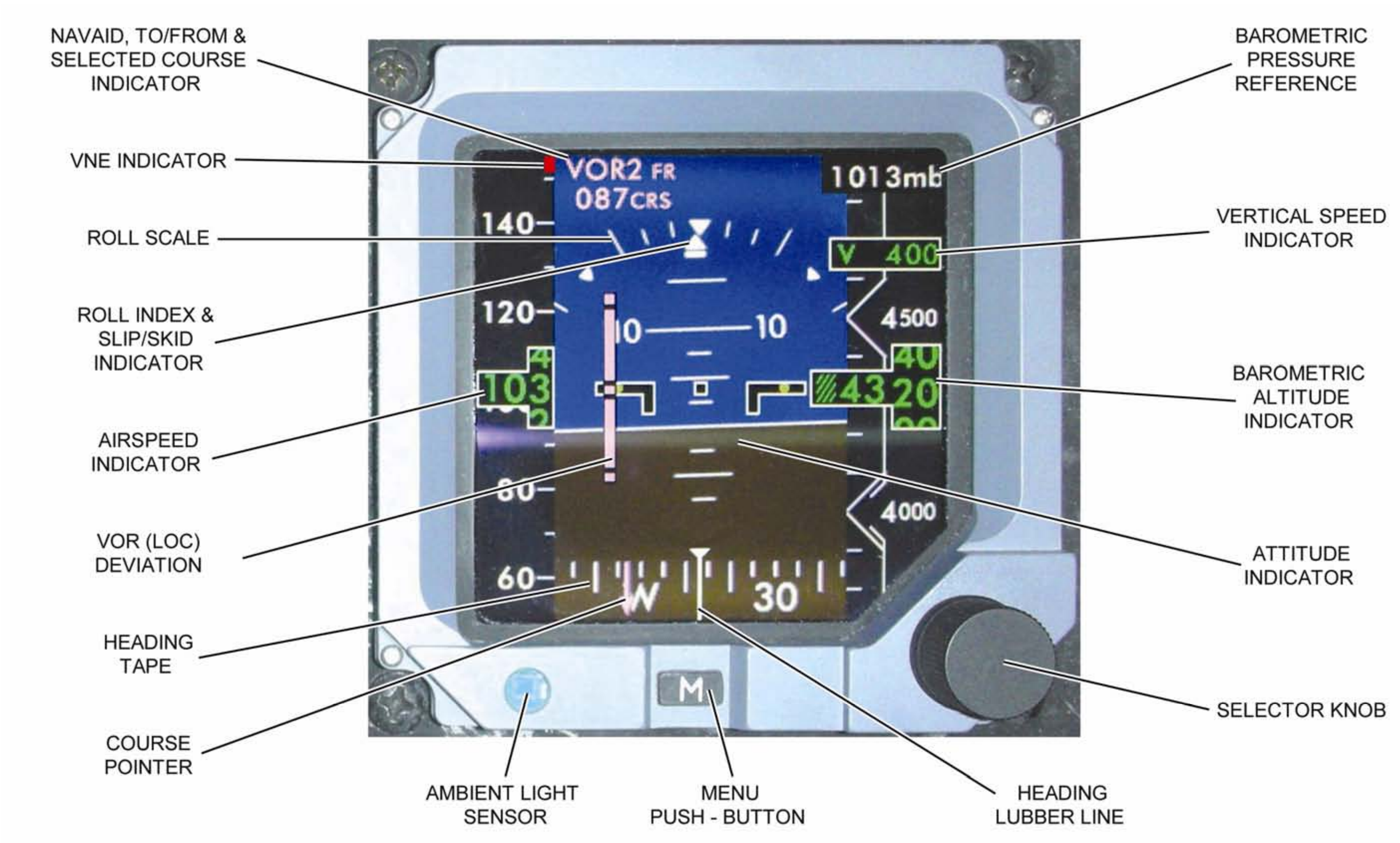
- Magnetic Heading
- VOR/ILS Deviations and TO/FROM
- Marker Beacon

ADI STBY – PRINCIPLE OF OPERATION

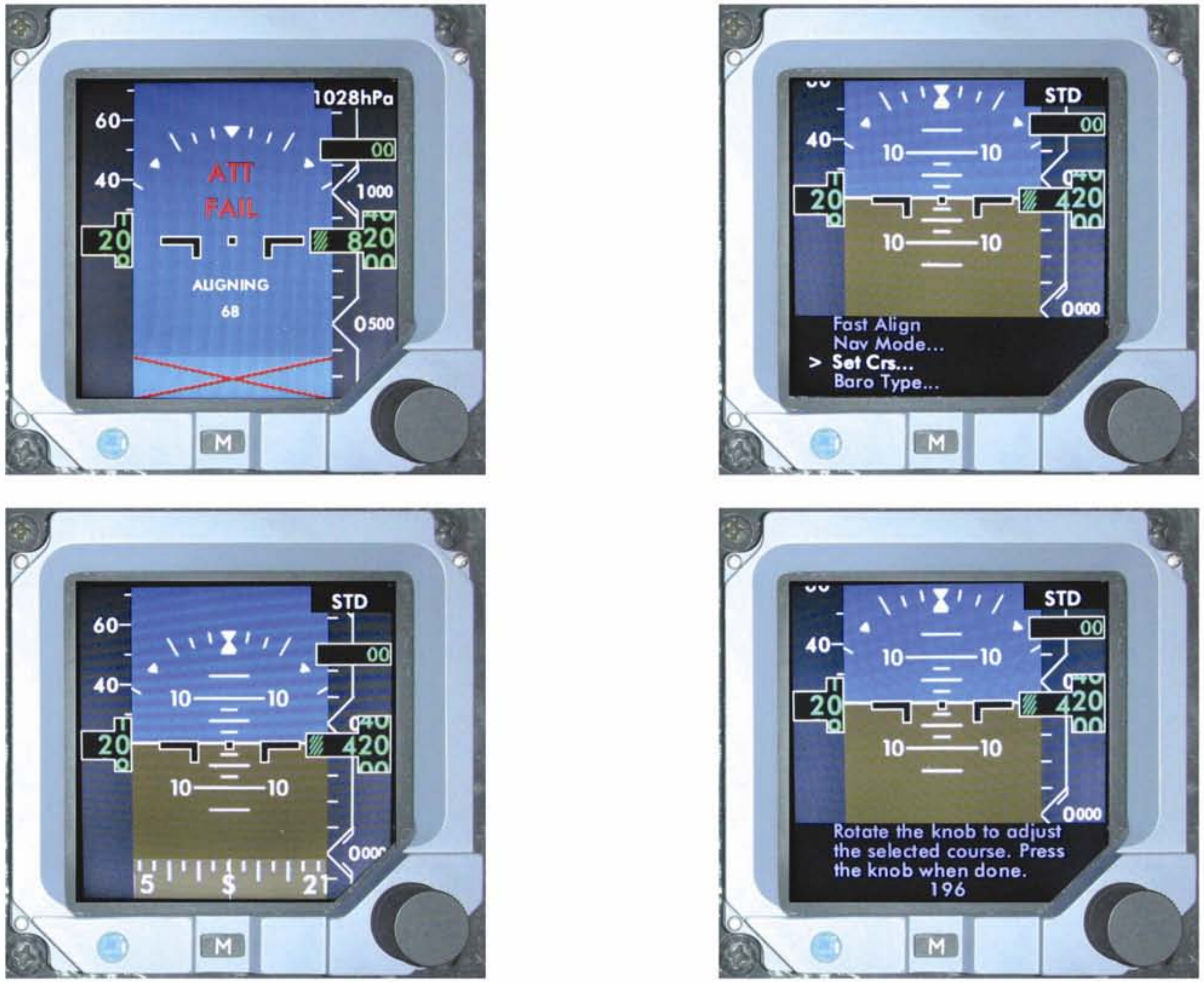
When the STBY ADI is powered, starts an automatic process of self diagnostics prior to normal operations. If no failure is detected, the unit displays the ATT FAIL indication with the message ALIGNING and a completion timer/counter below the aircraft symbol. The sensor alignment reaches the normal operation mode within three minutes of applying power.

During abnormal condition, such as motion during the sensor alignment mode, the indicator will reset and attempt to reach the normal operation mode within 6 minute of applying power. If sensor alignment is unsuccessful, the message will change to ALIGNMENT FAIL and the system will not enter the operational mode.

While the indicator is operating normally, the system continues to perform diagnostic self tests to assure accurate information.



ADI STANDBY INDICATOR (1 of 2)



ADI STANDBY INDICATOR (2 of 2)

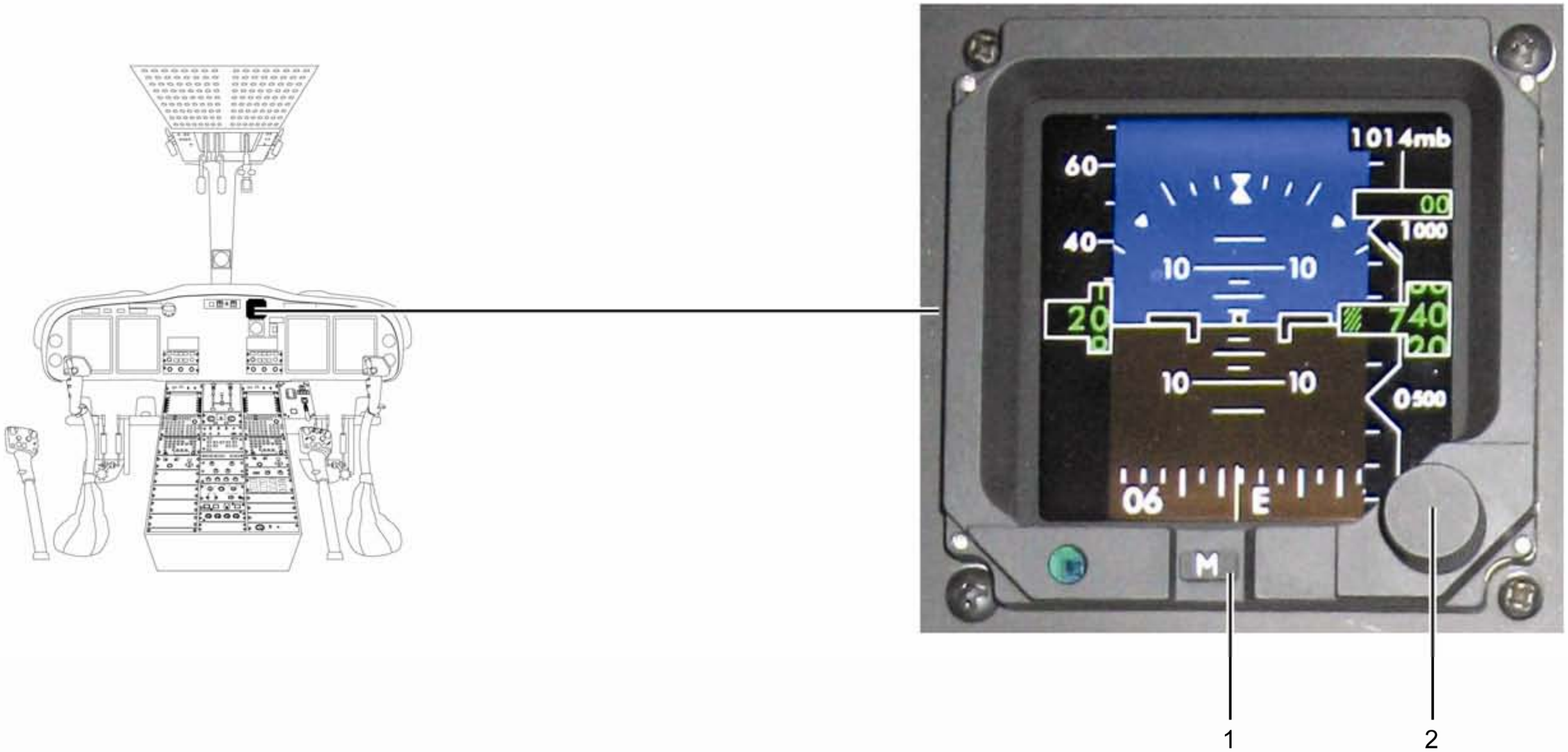
ADI STBY – CONTROLS AND INDICATORS

1. M push button

………………..	when pressed, the menu is available along the bottom portion of the display

2. Adjustment knob

	when rotate, it allows the scroll through the submenu item and highlights the item.
FAST ERECT	press knob to initiate
SET BRIGHTNESS OFFSET	press knob for submenu, rotate knob to adjust, press knob to finish
FAST ALIGN	press knob to initiate
SET HEADING	press knob for sub-menu, rotate knob to set heading, press knob to finish
NAV (ON/OFF)	press knob to toggle for opposite of current condition;
NAV MODE	press knob for sub-menu, rotate knob to select mode, press knob to finish;
SET CRS	press knob for sub-menu, rotate knob to set course, press knob to finish
ILS (BC/NORMAL)	press knob to toggle for opposite of current condition
CRS AUTO CENTER	press to initiate
BARO TYPE	press knob for sub-menu, rotate knob to select type, press knob to finish



ADI STANDBY – CONTROLS AND INDICATORS

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CHAPTER 46

INTEGRATED AVIONICS

SECTION 00 – GENERAL

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INTEGRATED AVIONICS – GENERAL

The aircraft avionics are integrated into the Honeywell PRIMUS EPIC® system whose architecture is based on:

- 2 × Modular Avionics Units (MAU 1 and MAU 2)
- 4 × Display Units (DU 1 thru DU 4)
- 2 × Modular Radio Cabinets (MRC 1 and MRC 2)

The two Modular Avionics Units:

- handle, compute and distribute data of all aircraft systems to the four Display Units
- interface mechanical and virtual controllers to control the displays and the avionics
- integrate the functions of crew alerting, navigation and autoflight systems

The four Display Units, two per pilot acting as PFD (Primary Flight Display) and MFD (Multi-Function Display), provide crew with the information necessary for flight conduct and aircraft system monitoring. (See chapter 31-60)

The two Modular Radio Cabinets:

- integrate all the standard radios
- handle the digital audio

(See chapter 23-00 for detailed description of MRC)

The system includes a Central Maintenance Computer (part of MAU1) to monitor the operation of the helicopter and record events requiring maintenance actions.

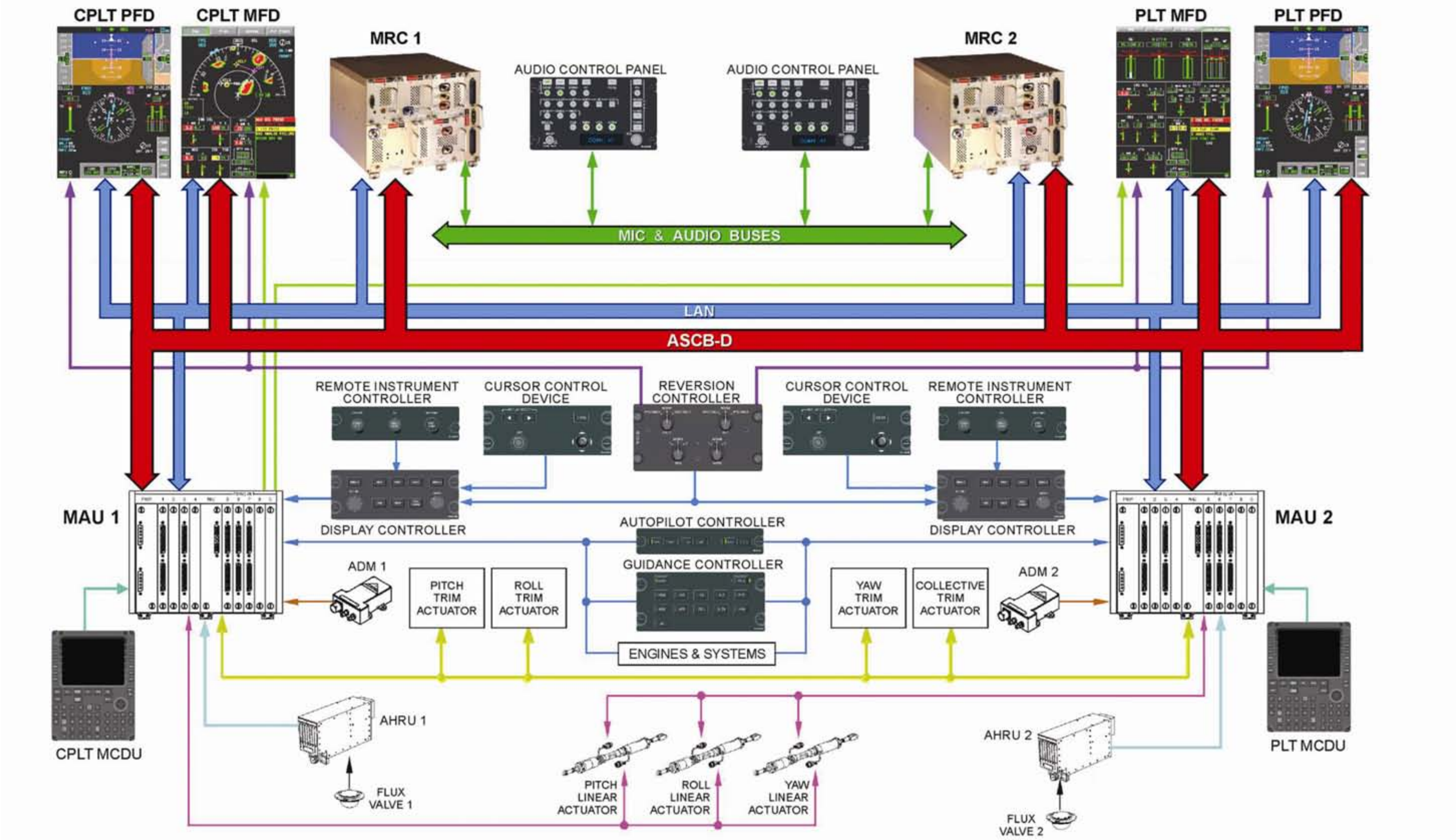
The standard configuration for the PRIMUS EPIC® system includes the following:

- 4 × Display Units (DU)
- 2 × sets of Controllers including 2 × Multifunction Control Display Units (MCDU)
- 1 × Inter-Communications System (ICS)
- 2 × VHF communication radio systems
- 2 × VOR / ILS / MB systems (VHF-NAV)
- 1 × Distance Measuring Equipment (DME)
- 1 × Automatic Direction Finder (ADF)
- 2 × Flight Management Systems (FMS)
- 1 × Global Positioning System (GPS)
- 2 × Air Data Systems (ADS)
- 2 × Attitude and Heading Reference Systems (AHRS)
- 1 × ATC Transponder (XPDR)
- 1 × Radio Altimeter (RAD ALT)
- 2 × Automatic Flight Control Systems (AFCS) including dual Autopilot (AP) and dual Flight Director (FD)

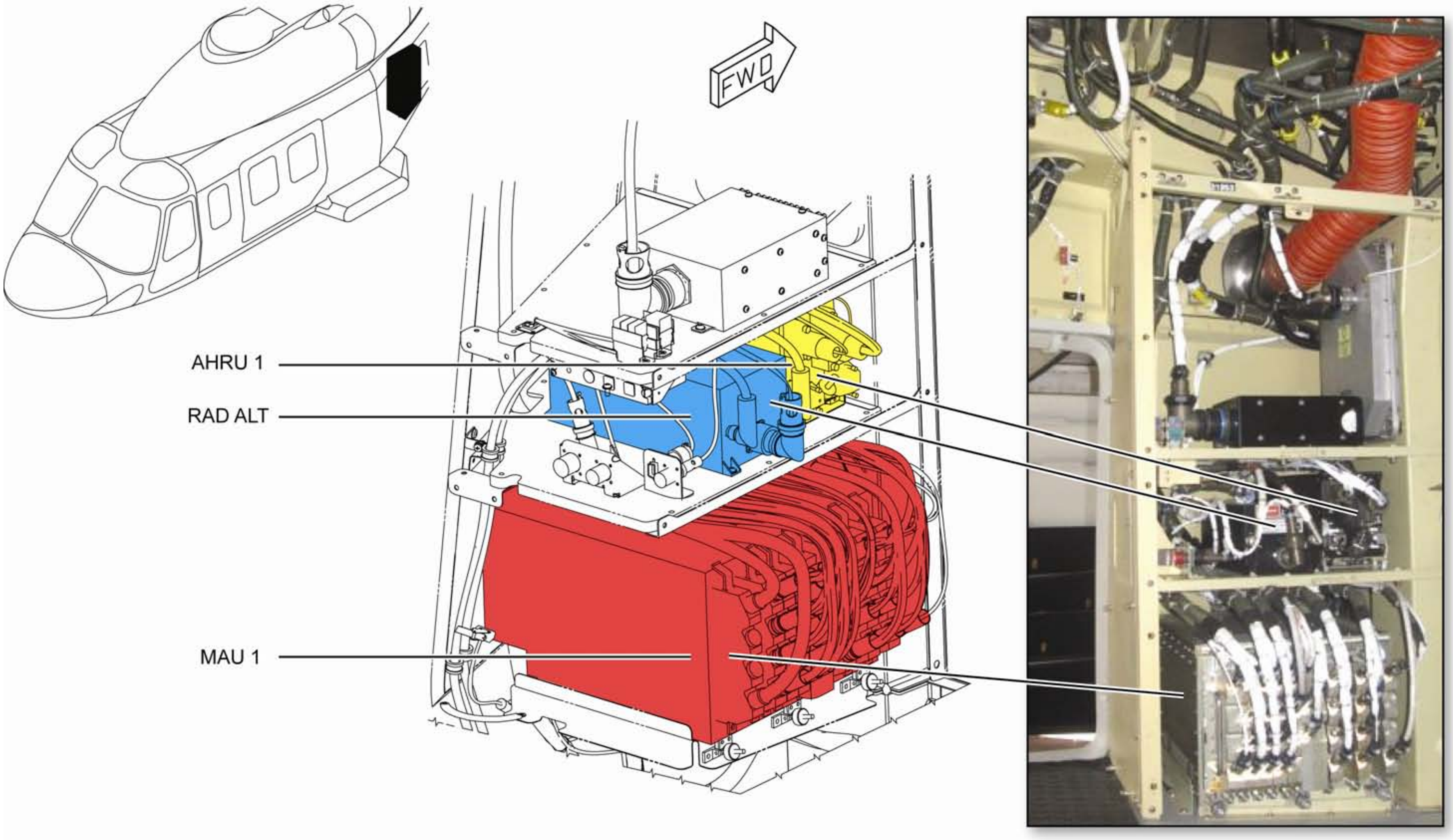
and the following optional systems:

- Weather Radar (WX)
- Lightning Sensor System (LSS)
- High Frequency (HF) communication radio system
- Traffic Alert and Collision Avoidance System (TCAS)
- Enhanced Ground Proximity Warning System (EGPWS)

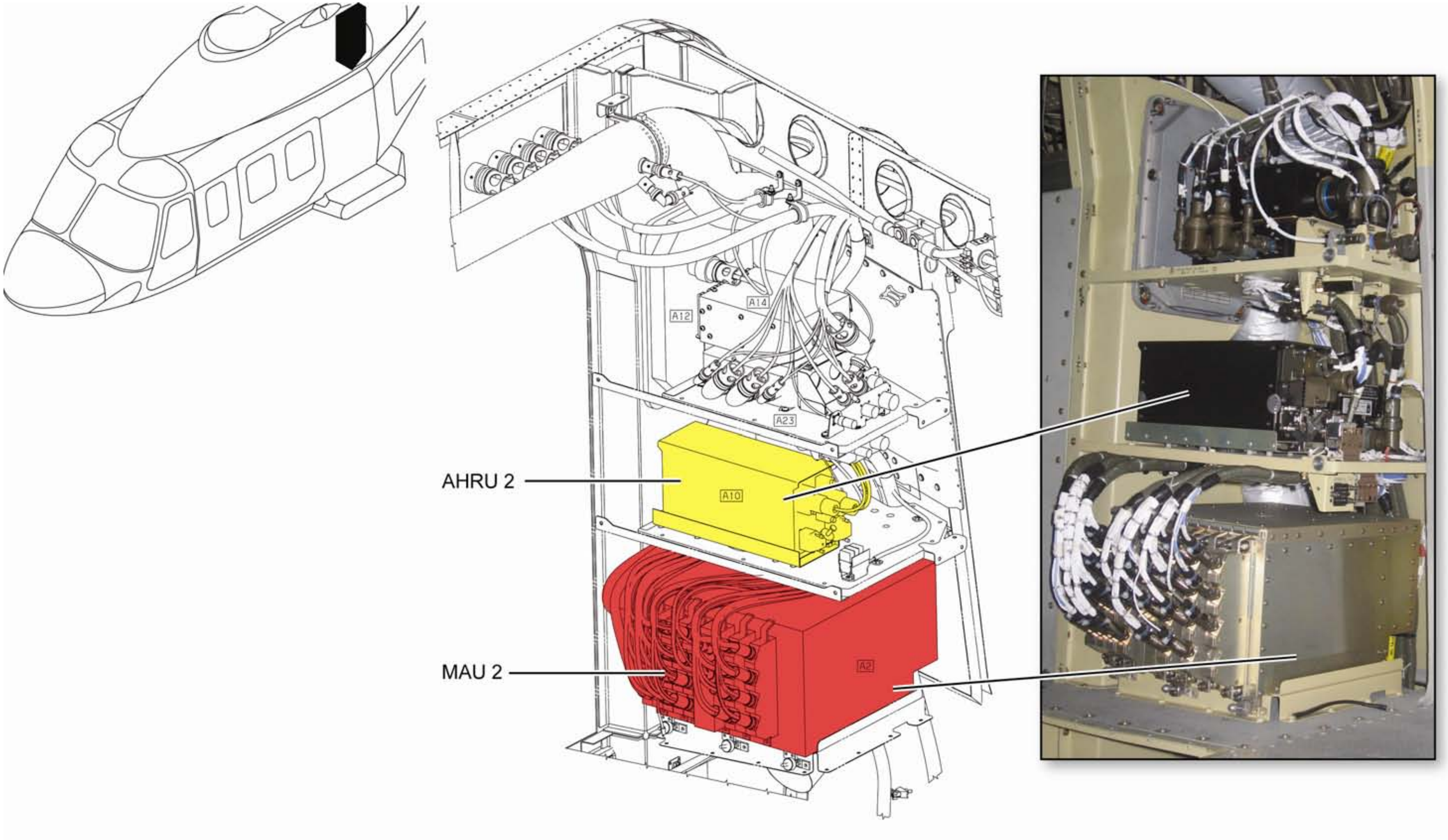
The PRIMUS EPIC® system also supplies interfaces for other equipment installed on the helicopter.



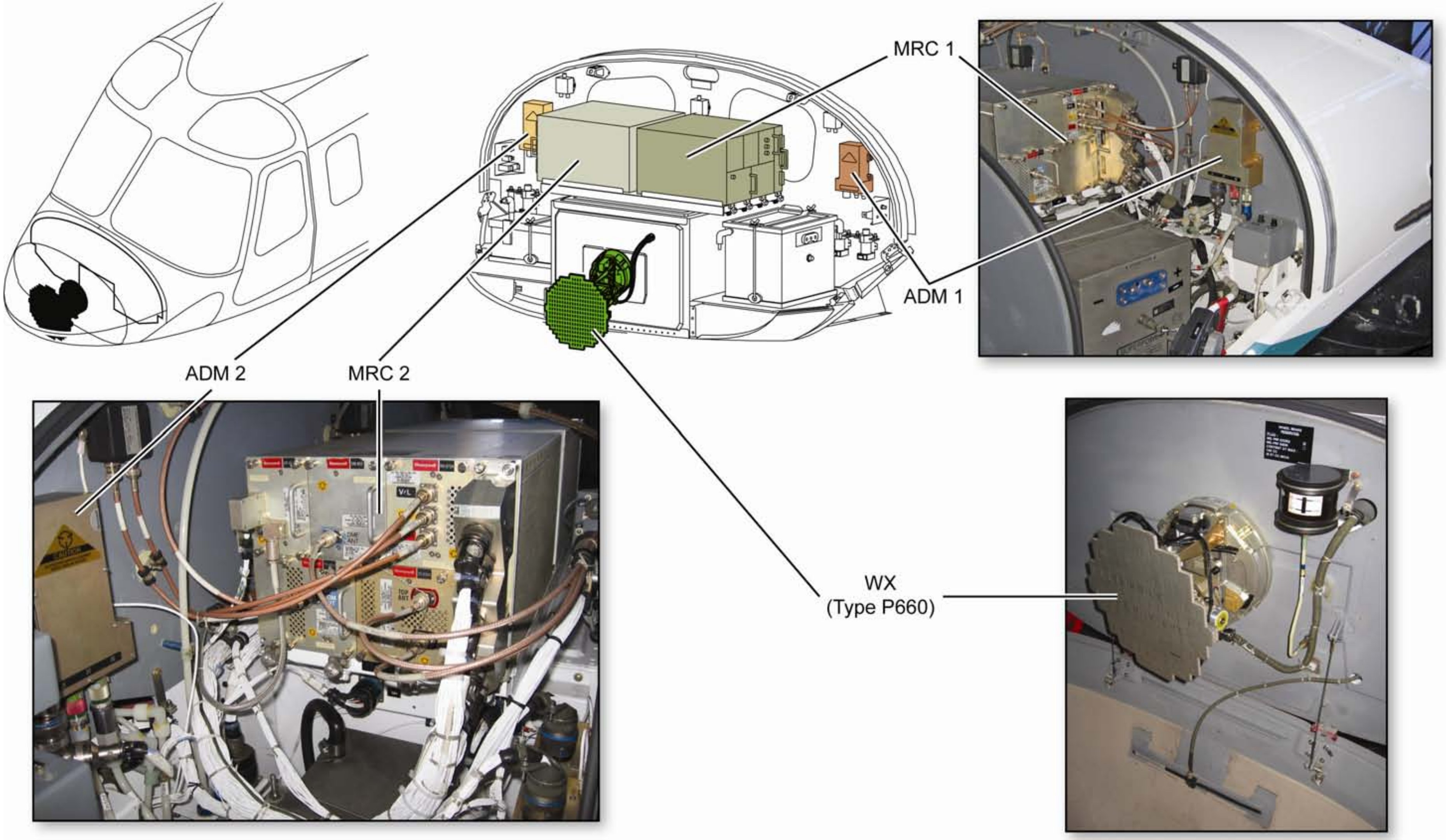
PRIMUS EPIC® - GENERAL ARRANGEMENT



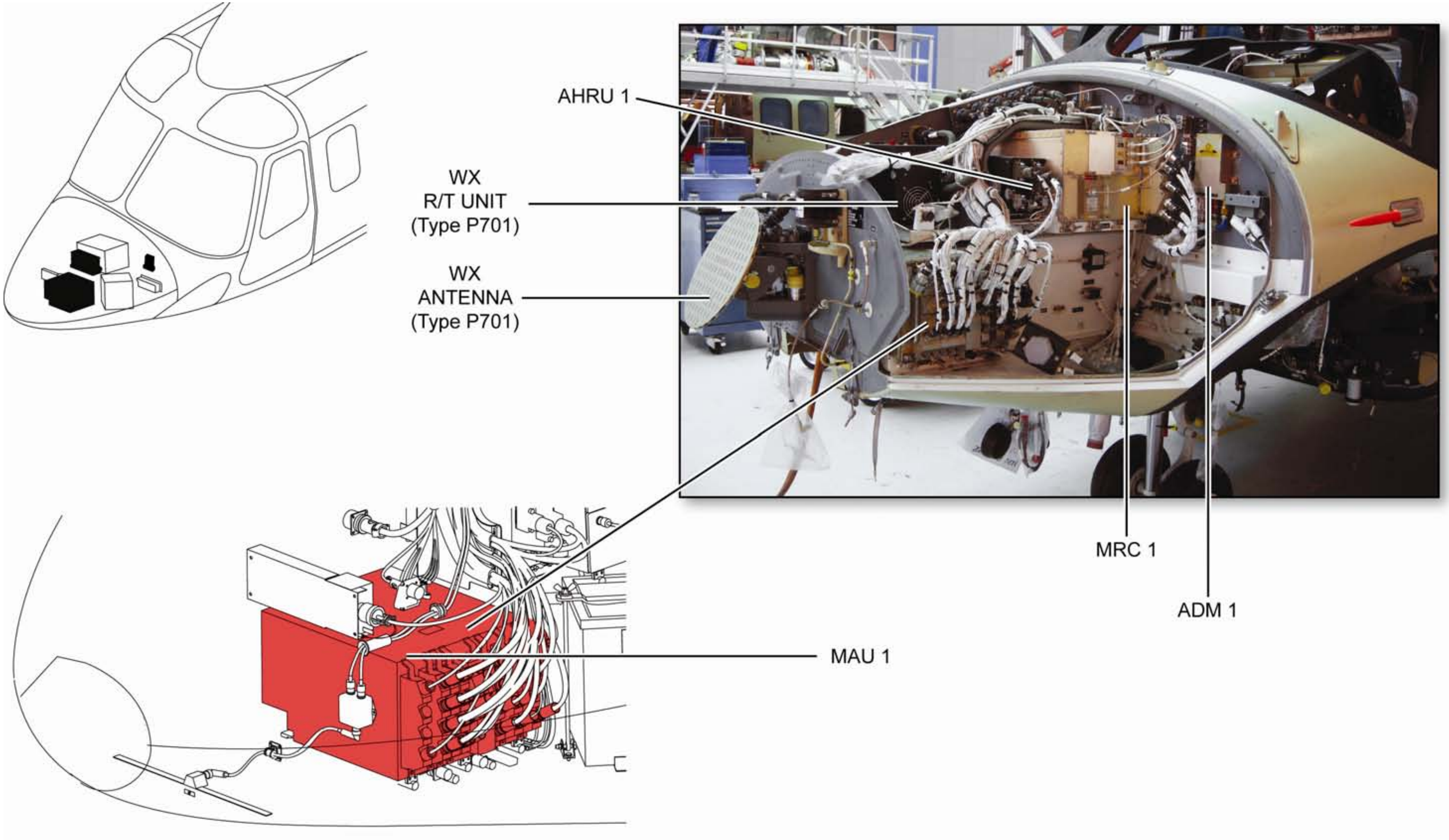
LOCATION OF MAIN COMPONENTS – STANDARD CONFIGURATION (1 OF 3)



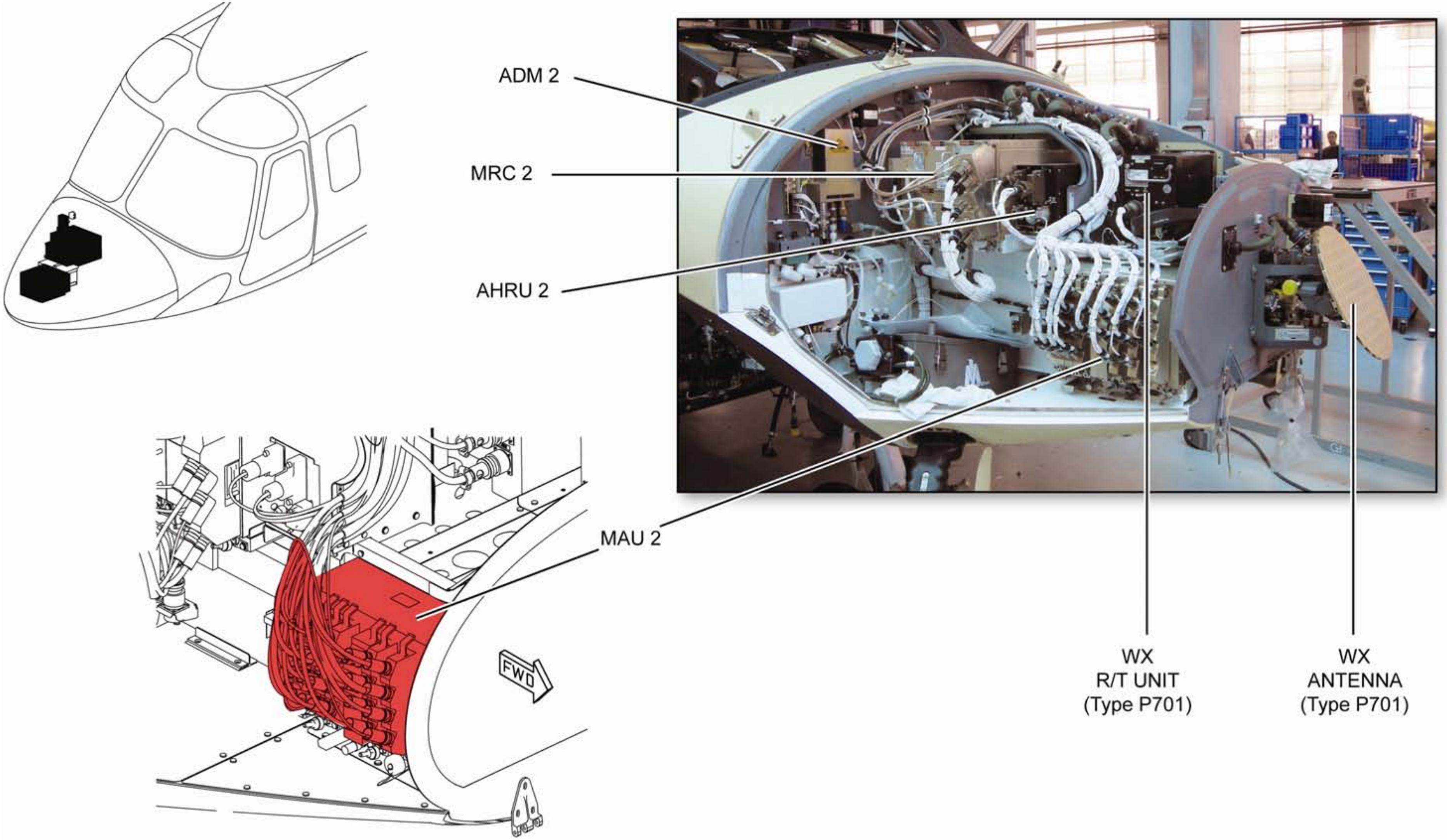
LOCATION OF MAIN COMPONENTS – STANDARD CONFIGURATION (2 OF 3)



LOCATION OF MAIN COMPONENTS – STANDARD CONFIGURATION (3 OF 3)



LOCATION OF MAIN COMPONENTS – LONG NOSE CONFIGURATION (1 OF 2)



LOCATION OF MAIN COMPONENTS – LONG NOSE CONFIGURATION (2 OF 2)

LOCATION OF MAIN COMPONENTS

DATA BUSES (ASCB-D and LAN)

All the units of the PRIMUS EPIC® system use the virtual back plane network to send data between each other. The virtual back plane network contains:

- the Avionics Standard Communication Bus version-D (ASCB-D) which
- the software and the hardware necessary to send and receive data on the ASCB-D

The MAUs, the Display Units (DU) and the Modular Radio Cabinets (MRC) are directly connected to the Avionics Standard Communication Bus version-D (ASCB-D) to distribute data.

The ASCB-D is a high-integrity bi-directional digital data bus and consists of four individual buses, two for LH (copilot side) units and two for RH (pilot side) units, providing dual redundancy in data distribution.

Each unit connected to the ASCB-D contains the software and the hardware necessary to send and receive data on the ASCB-D: this is typically named Network Interface Module (NIM).

The corresponding module in the MAU is instead named Network Interface Controller (NIC) since it governs the whole ASCB-D data flow.

MAU 1, MRC 1 and the copilot's DUs (DU 1 and DU 2) are interconnected by the LH (Main) ASCB-D and by the LH Backup ASCB-D.

MAU 2, MRC 2 and the pilot's DUs (DU 3 and DU 4) are interconnected by the RH (Main) ASCB-D and by the RH Backup ASCB-D.

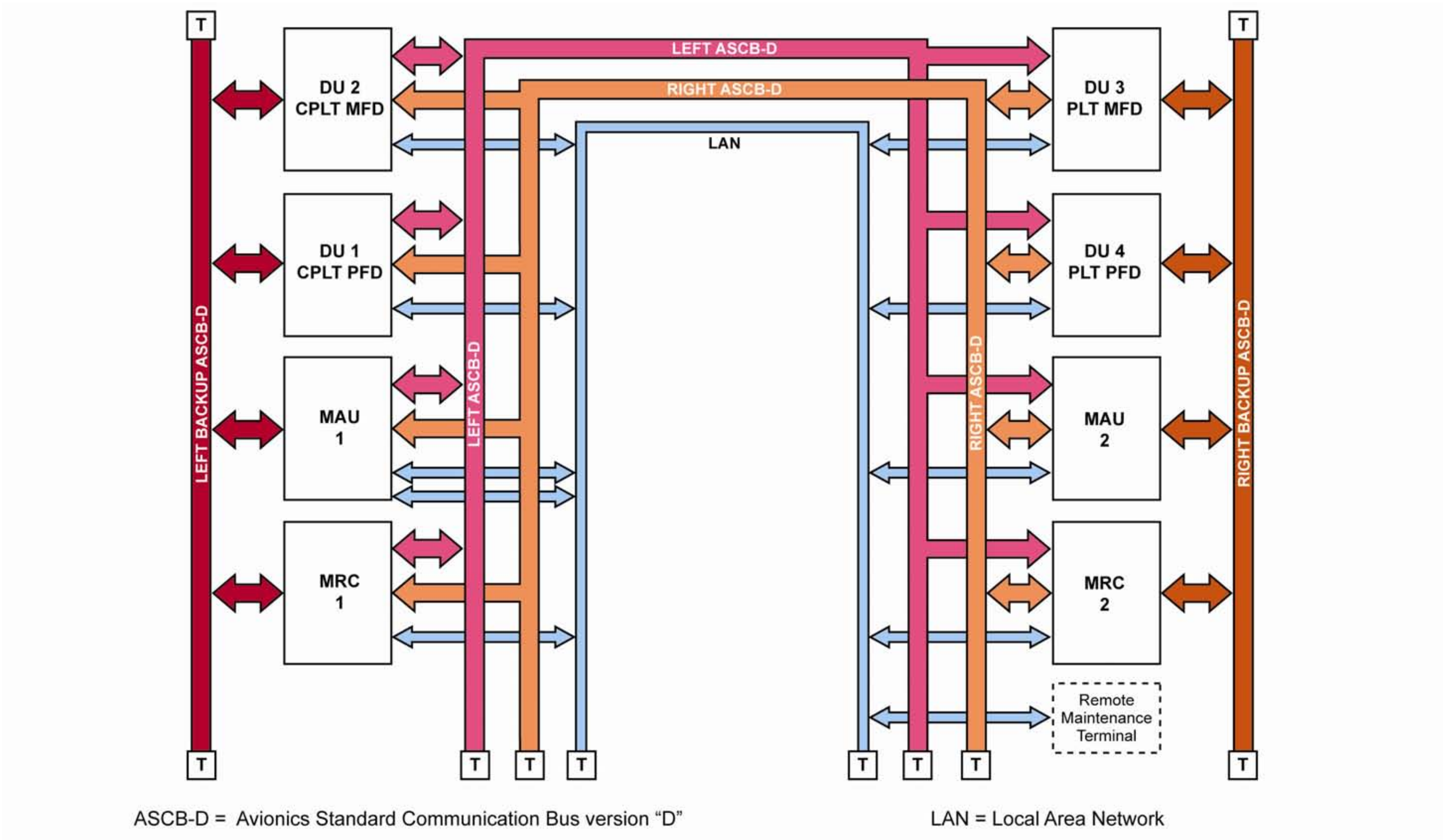
The two LH and RH (Main) ASCB-D buses cross-feed data to the opposite side units.

A Local Area Network (LAN) is installed for data loading, maintenance and test functions.

The LAN interfaces the same modules as the ASCB-D and the CMC (Central Maintenance Computer) module of MAU 1.

Maintenance is carried out on the ground by connecting a computer to the LAN. This allows:

- to upload database files and/or software to the system
- to download data or fault history files



ASCB-D AND LAN CONFIGURATION

MAU ARCHITECTURE

The Modular Avionics Unit is a cabinet that contains line replaceable modules that integrate the different MAU functions.

The most important modules are duplicated in MAU 1 and MAU 2 for redundancy, while other modules are single.

The MAU uses a Digital Engine Operating System (DEOS) and data is made available through a bus (called Virtual Back Plane Peripheral Component Interconnect (VbPCI) bus) that connects the NIC to all the modules in the MAU and to the ASCB-D.

The Power Supply module in each MAU is supplied with two 28 VDC inputs for redundancy: one power input only is used at a time.

Two circuit breakers are therefore provided for each MAU:

- MAU 1 PRI PWR connected to MAIN 1 bus
- MAU 1 AUX PWR connected to MAIN 2 bus
- MAU 2 PRI PWR connected to ESS 1 bus
- MAU 2 AUX PWR connected to ESS 2 bus

NOTE: MAU 1 power inputs are connected to ESS 1 and ESS 2 buses when the aircraft is on the ground or when starting an engine.

Two electrical fans are installed in each MAU cabinet to keep the electronic modules cool; they are automatically operated and monitored by the CMC only, i.e. no indication is provided during flight in case of failure.

A temperature switch triggers the 1(2) MAU OVHT caution message in the CAS if the MAU temperature exceeds a set value following dual fan failure in an MAU.

The most important module in the MAU is the NIC/PROC (Network Interface Controller and Processor) module which:

- Handles, processes and distributes all the data via the ASCB-D
- Provides the Monitoring Warning Function
- Houses the Flight Management System (FMS) software

Failure of the NIC/PROC module causes the total failure of the MAU that is annunciated by the 1(2) MAU caution message in the bottom left corner of both PFDs.

Interface with aircraft systems, controllers and sensors is provided by two modules in each MAU:

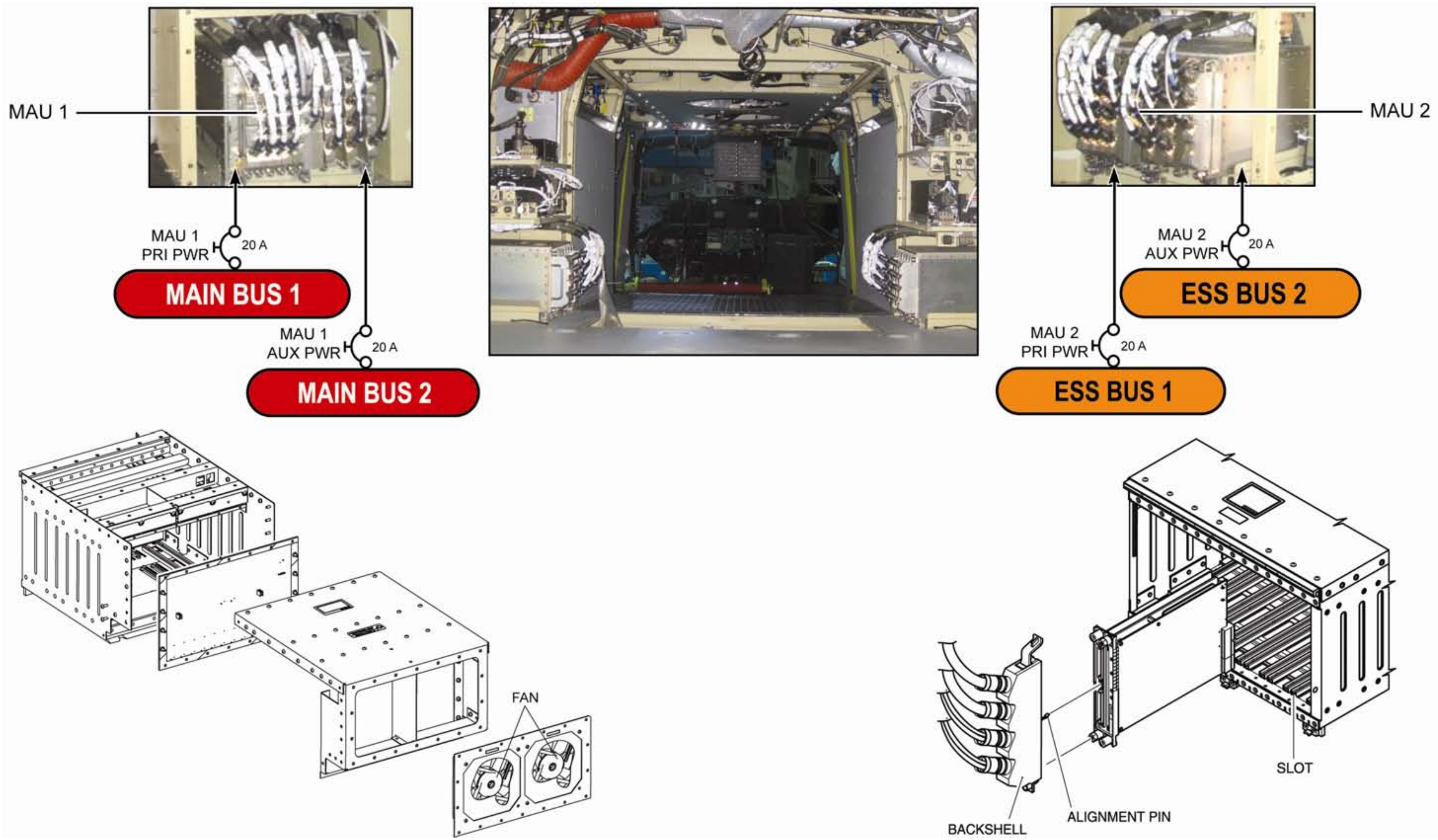
- CIO (Control Input/Output) module
- CSIO (Custom Input/Output) module

Only the CSIO module in MAU 2 provides Aural Warning Generator function (see chapter 31-50).

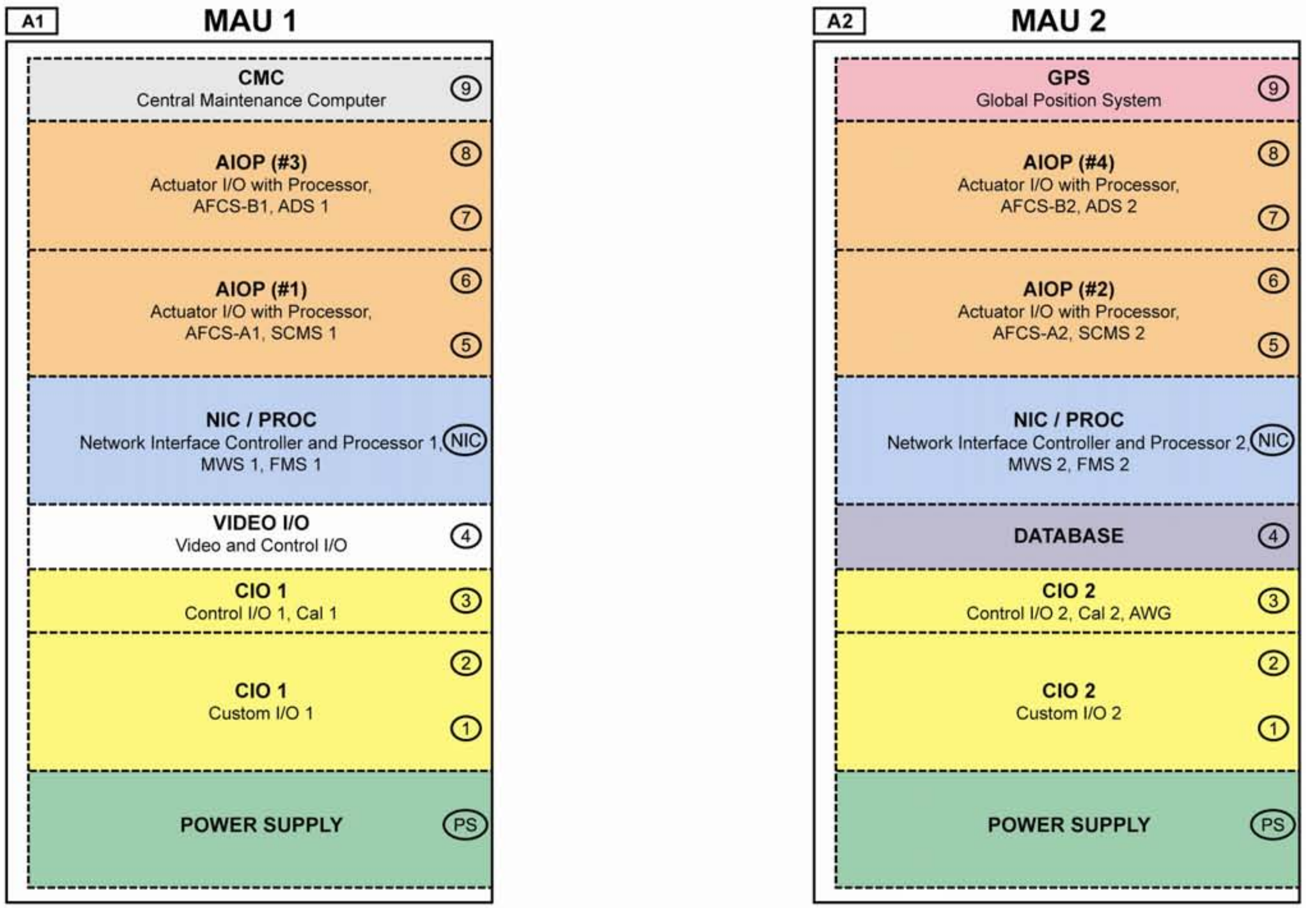
Each MAU also contains two AIOP (Actuator Input/Output with Processor) modules: they provide AFCS functions (see chapter 22-00).

The GPS module and the DATABASE module are only provided in MAU 2 (see chapter 34-00).

The CMC (Central Maintenance Computer) module and the optional VIDEO I/O module are only provided in MAU 1.



MAU 1 AND MAU 2 LOCATION



ADS = Air Data System

AFCS = Automatic Flight Control System

SCMS = Software Configuration Management System

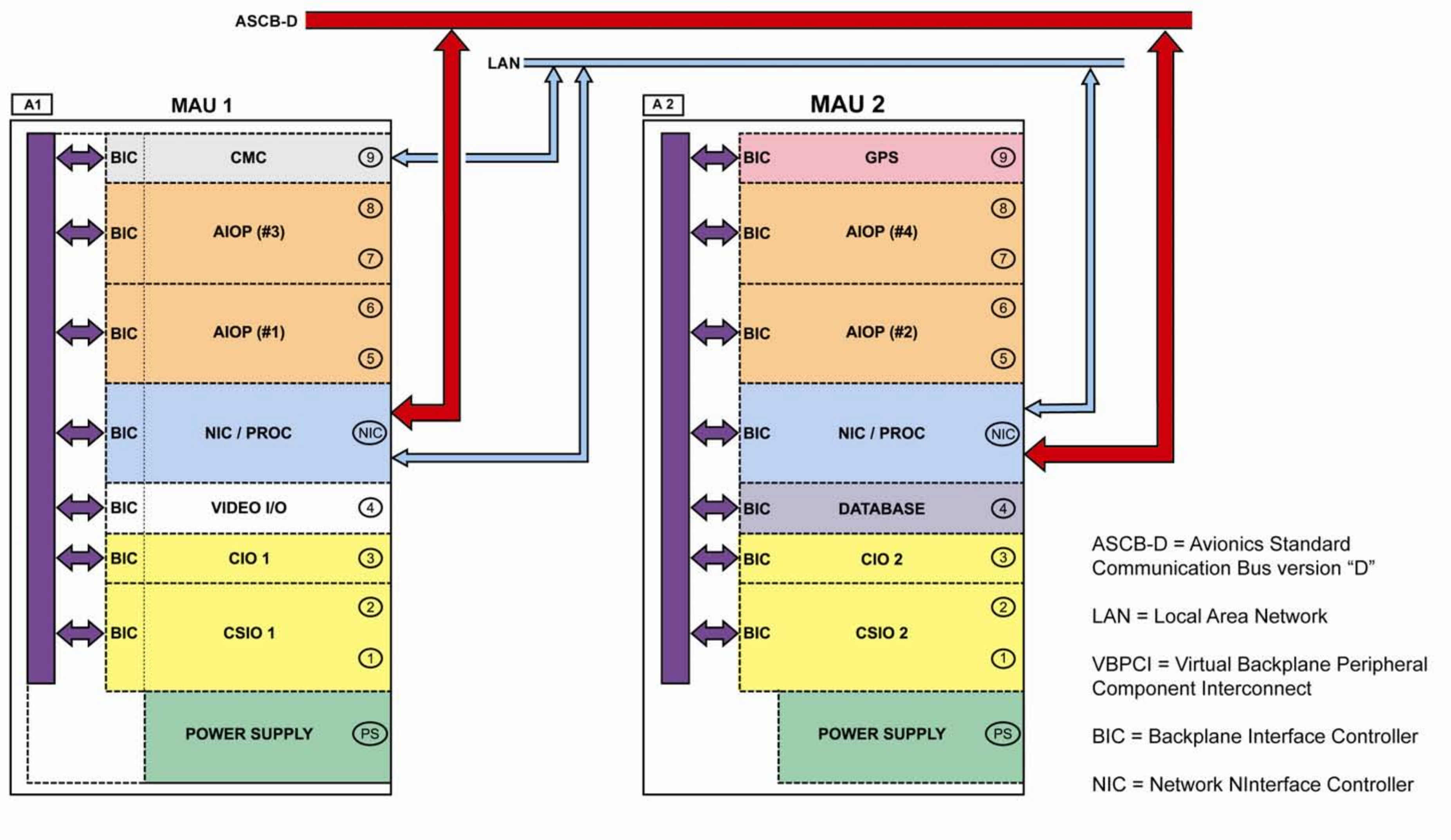
AWG = Aural Warning Generator

FMS = Flight Management System

MWS = Monitor Warning System

NIC = Network Interface Controller

MAU 1 AND MAU 2 CONFIGURATION



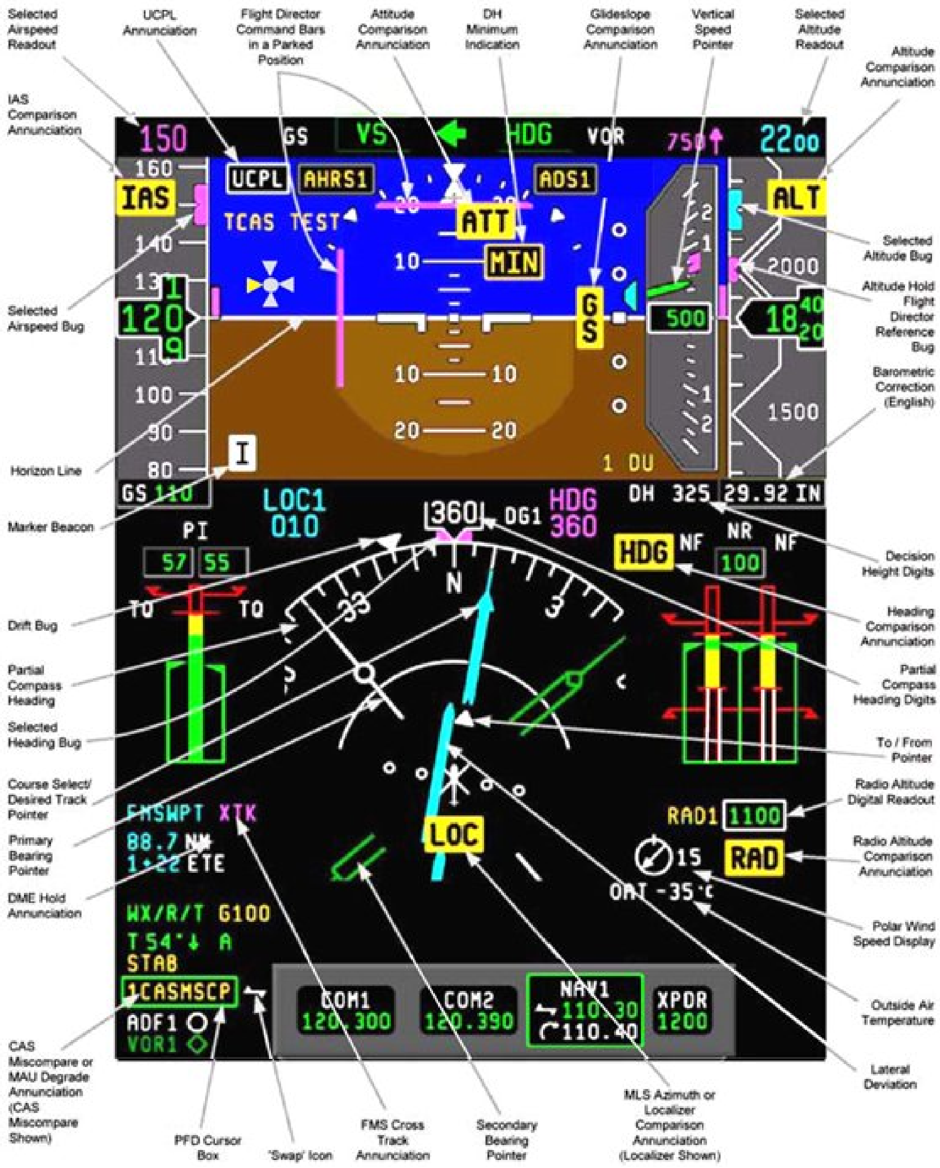
MAU 1 AND MAU 2 ARCHITECTURE

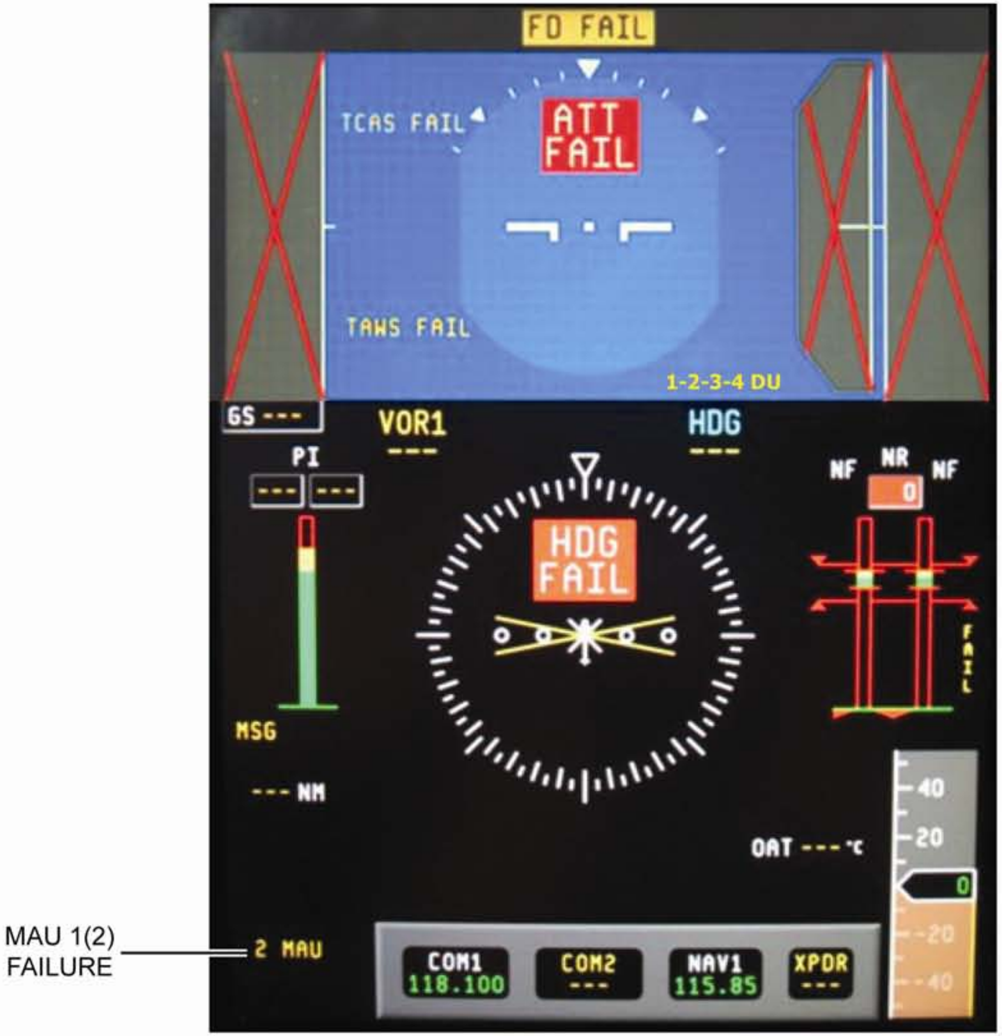
INTEGRATED AVIONICS - ANNUNCIATORS

A summary of main messages and failure indications are displayed in the figure aside.

This figure is available in the Rotorcraft Flight Manual (AW139–RFM–4D, end of Section 2).

Refer to the Honeywell PRIMUS EPIC® Pilot's Guide for a complete description of the symbols.





MAU FAILURE INDICATIONS

CAS CAUTION MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
1(2) MAU OVHT	Associated MAU overheat	MODULAR AVIONICS UNIT OVERHEAT / FAIL	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES
SYS CONFIG FAIL	Hardware or software configuration problem (displayed on ground only)	SYSTEM CONFIGURATION FAILURE	
SYS CONFIG FAIL	Hardware or software changed, configuration validation operation required (displayed on ground only)	VALIDATE CONFIGURATION	

PFD CAUTION MESSAGES

PFD MESSAGE	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
1(2) MAU	Associated MAU failure	MODULAR AVIONICS UNIT OVERHEAT / FAIL	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES

CHAPTER
51
STRUCTURE

SECTION 00 – GENERAL

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STRUCTURE – GENERAL

This chapter describes the main features of the following components:

- Fuselage;
- Doors;
- Windows.

FUSELAGE - GENERAL

The fuselage comprises:

- the forward fuselage;
- the center fuselage;
- the rear fuselage;
- the tail section.

FORWARD FUSELAGE

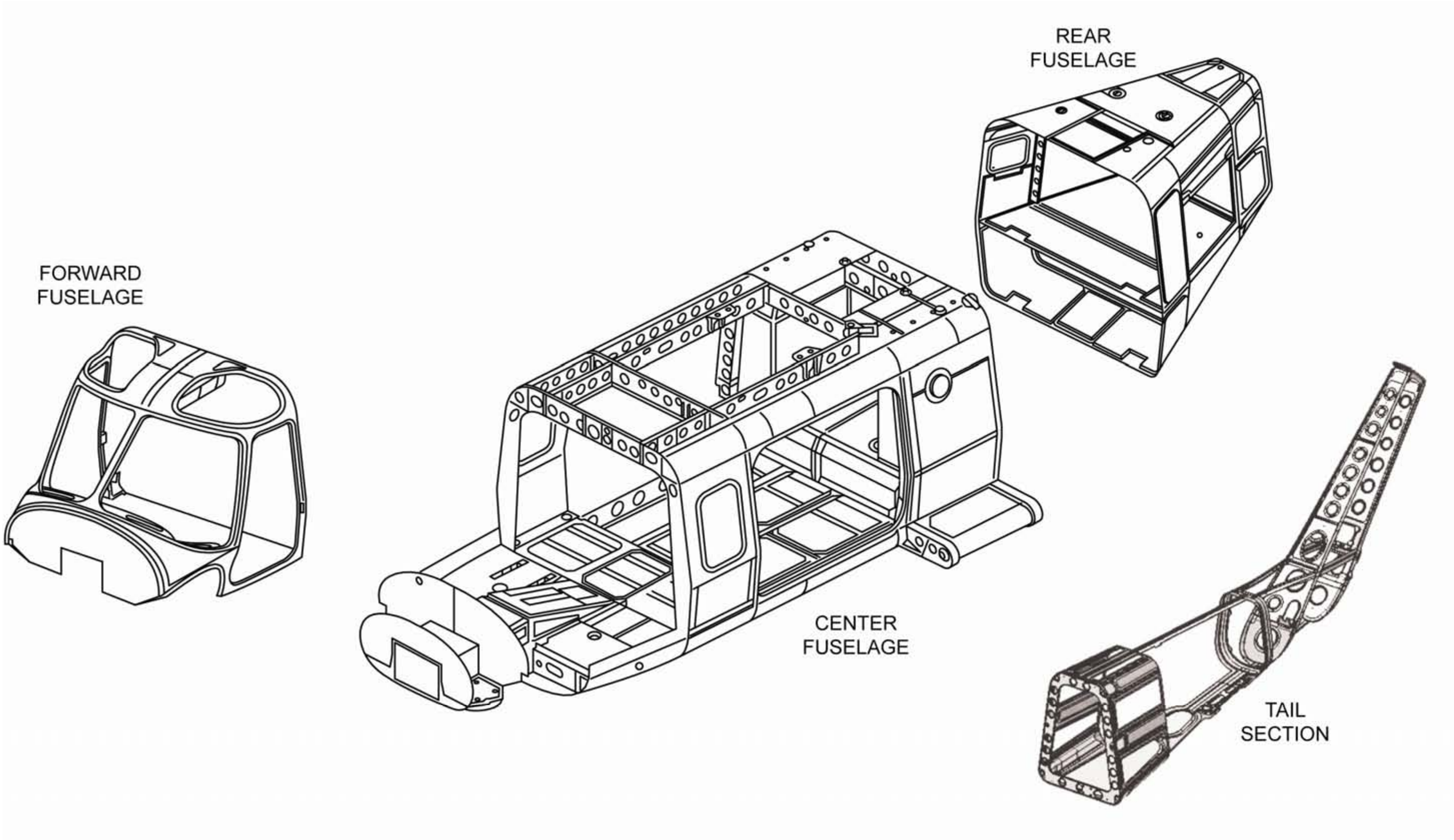
The forward fuselage is a metallic construction made with longerons, frames and sandwich panels; it mainly provides the pilots cabin floor; interface with the cockpit, houses the nose landing gear: the upper part of the forward fuselage is designed as canopy.

CENTER FUSELAGE

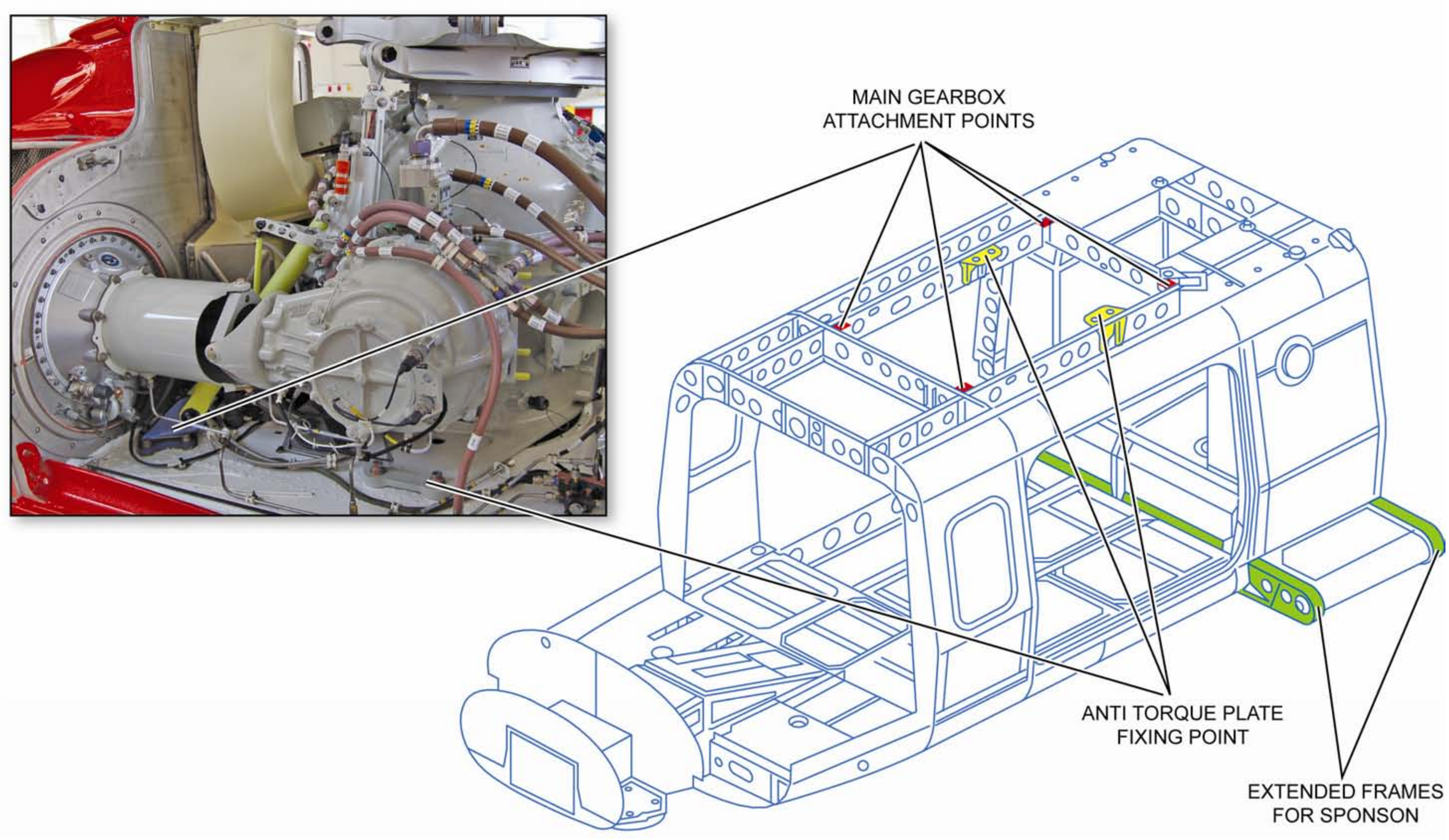
The center fuselage is a metallic semi-monocoque construction made with sheet, beams, frames and sandwich panels; it mainly provides the passengers cabin fuel, hydraulics, dynamic components, landing gear, flight controls and engine installation. The top side of the centre fuselage is completed with the upper deck/engine fairings and cowlings. The upper deck fairings are manufactured in composite materials. The sponsons are located at the end of the center fuselage.

REAR FUSELAGE

The rear fuselage is a metallic semi-monocoque construction made with sheet, beams, frames and sandwich panels; it mainly provides the baggage compartment, the avionics installation and the engine installation. The top side of the rear fuselage is completed with the engine fairings and cowlings.


REAR
FUSELAGE
FORWARD
FUSELAGE
CENTER
FUSELAGE
TAIL
SECTION

FUSELAGE



CENTER FUSELAGE - DETAILS

TAIL SECTION

The tail section is a metallic semi-monocoque construction made with sheet, beams and frames; it provides attachments for the tail rotor drive, the flight controls, the tail rotor and the stabilizers (horizontal and vertical).

The tail section consists of a cone and a fin; subdivision is intended for convention only. The tail section included the tail rotor drive fairings and the leading and trailing edge of the vertical fin.

The structure of the fuselage is maintained ON CONDITION and re-establishment is required only in case of damage. No overhaul is required before a service life of 30 years or 25,000 flight hours.

Longerons, formers, bulkheads are made of aluminium alloy; the lateral skin is made of aluminium honeycomb panels while the upper deck is made of titanium.

The most important fuselage safety requirements include:

- in case of crash the deformations are controlled without jeopardizing occupants escape
- natural frequencies are different from multiple frequency values of the main rotor and tail rotor to minimize risk of vibrations
- the structure sustains lighting effects
- the fuselage is fireproof in fire critical areas, like center fuselage and engine areas
- ground resonance effects are to be considered void

The structure of the tail section is maintained ON CONDITION and re-establishment is required only in case of damage.

TAIL SECTION

MATERIALS

The most important characteristics of the materials used to build the fuselage are the following:

FIBERGLASS

- high ratio of surface area to weight
- in contrast to carbon fiber, the glass inside the fiber can undergo more elongation or stretch ratio
- moisture is easily absorbed. This process can rapidly deteriorate microscopic cracks (if existing) and increase surface defects that can reduce the tenacity of the fiberglass

KEVLAR

- light, strong synthetic fiber related to other aramids fibre such as NOMEX and TECHNORA
- five times stronger than steel on an equal weight basis
- the ultraviolet (UV) component of the light degrades and decomposes Kevlar

CARBON FIBER

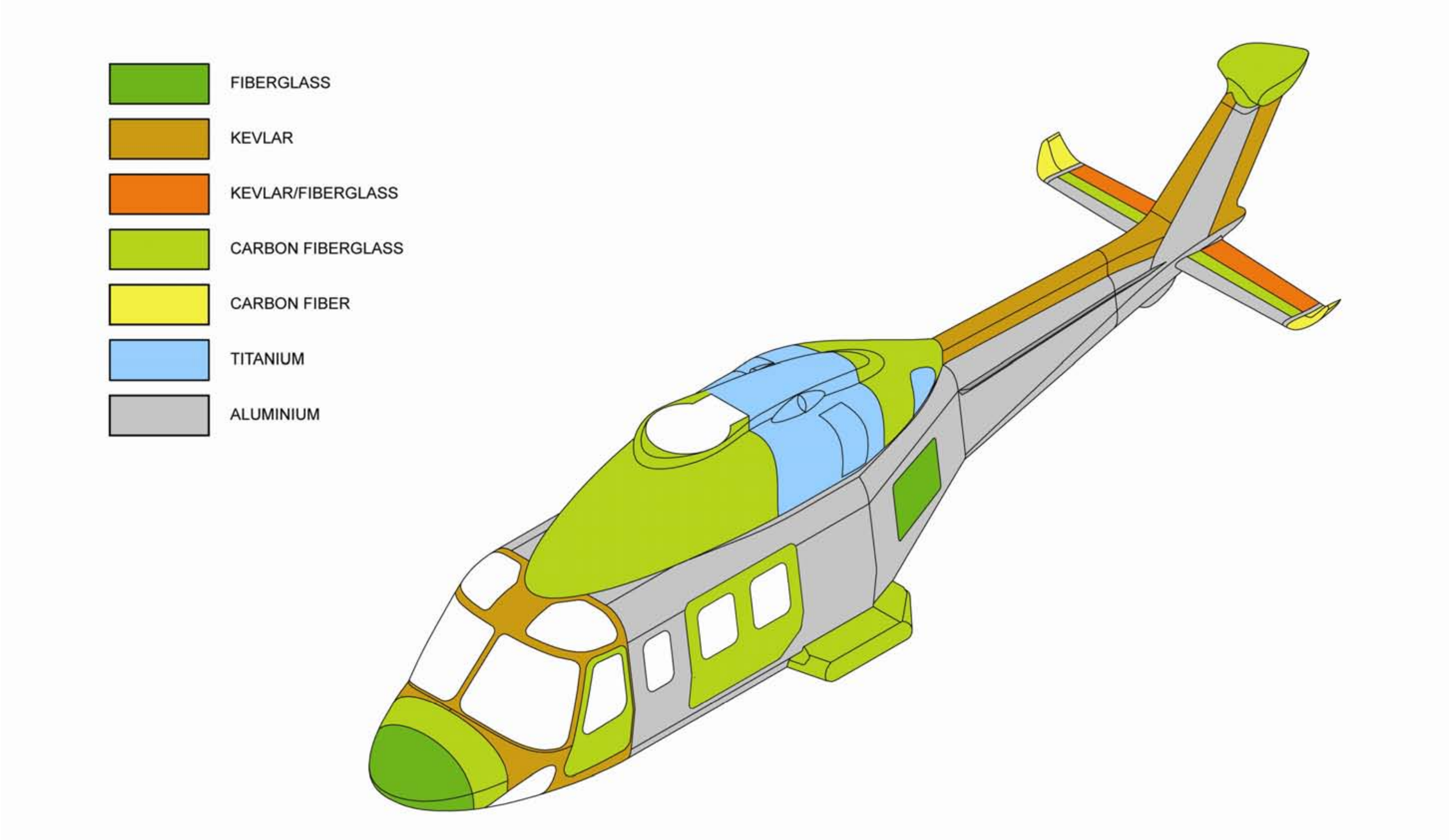
- highest compressive strength of all the reinforcing materials
- high strength to weight ratio and low coefficient of thermal expansion

TITANIUM

- high tensile strength to density ratio
- high corrosion resistance
- high temperatures resistance without creeping

ALUMINIUM

- good strength and durability
- aluminium alloys have no well-defined fatigue limit (meaning that fatigue failure will eventually occur under even very small cyclic loadings
- aluminium alloys will melt without first glowing red
- sensitivity to heat


FIBERGLASS
KEVLAR
KEVLAR/FIBERGLASS
CARBON FIBERGLASS
CARBON FIBER
TITANIUM
ALUMINIUM

MATERIALS

STABILIZERS – GENERAL

The stabilizers include the vertical fin and the horizontal stabilizer.

VERTICAL FIN

The vertical fin provides static stability of the aircraft with the following characteristics:

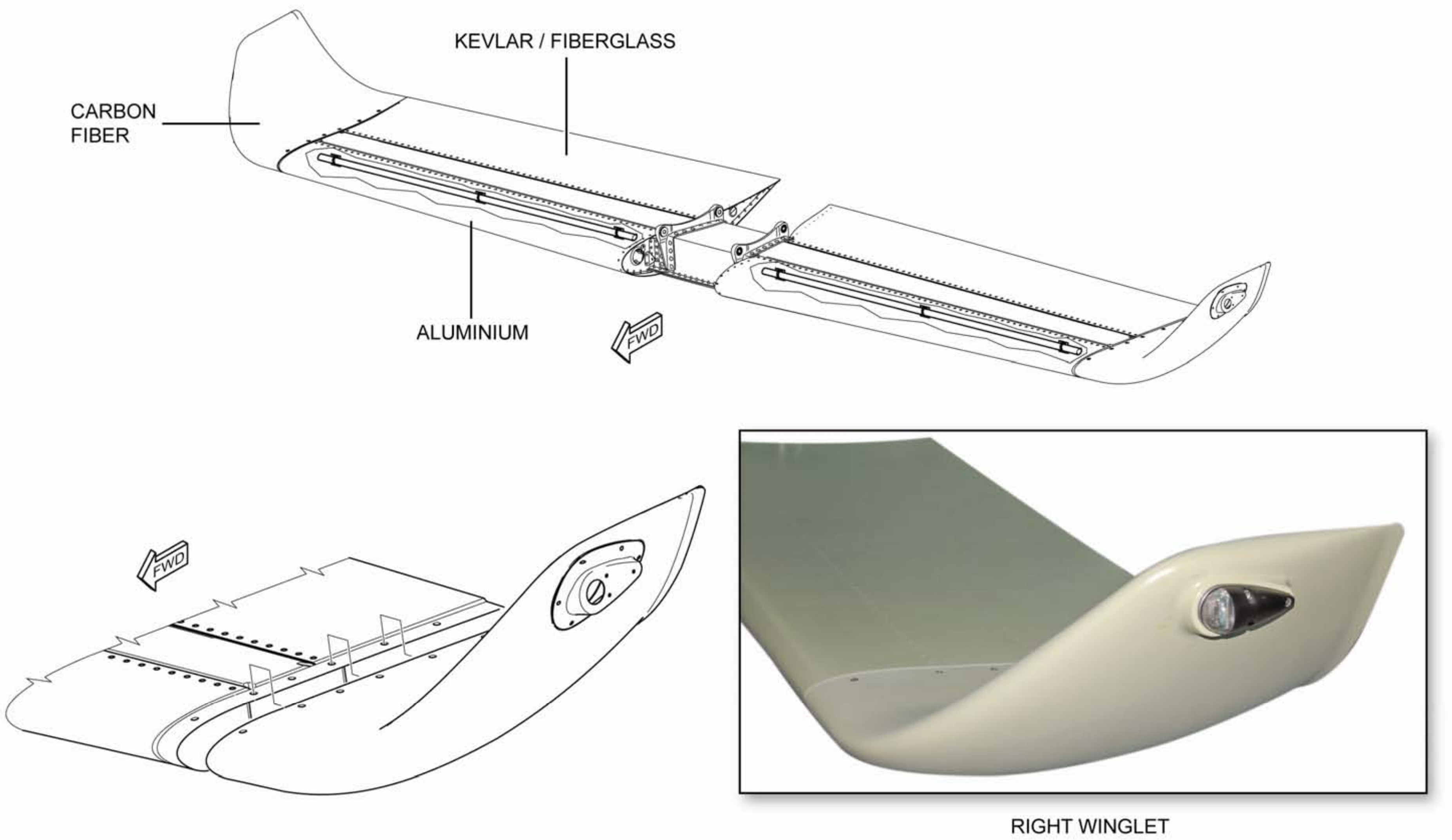
- in forward flight replaces the tail rotor for anti-torque forces
- the side force on the vertical fin must be great enough to lift off the tail rotor but, at the same time, sufficiently limited to permit autorotation (if this condition occurs)
- prevents dynamic instability as much as it has a large surface

HORIZONTAL STABILIZER

The horizontal stabilizer is made by a fixed structure installed on the tail section. The left and the right sides of the horizontal stabilizer are two wings with a different angle of incidence. This structural difference is due to aerodynamic effects of the main rotor on the horizontal stabilizer.

STRAKE

The strake is an aluminium device mounted on the upper left side of the tail section. This device produces an aerodynamic force that improves the helicopter performance in hover and lateral flight conditions.



HORIZONTAL STABILIZER AND WINGLETS

DOORS – GENERAL

Doors allow the crew and the passengers to enter or exit the aircraft. From the pilot point of view, the most important doors are:

- two cockpit doors (left and right)
- two cabin doors (left and right)
- one baggage door (located on the left side; the right baggage door is an optional)
- one nose door
- one external power door

The pilot can verify if the doors are fully closed and locked:

- by a direct visual inspection of the locking mechanism
- by monitoring the absence of caution messages on the CAS window of the MFD. The micro-switches are installed on each door allowing to check the position of the locking mechanism alerting the crew in case that the door are not closed.

COCKPIT DOORS

The cockpit doors provide access to both sides of the cockpit. The front side of the doors is hinged to the fuselage to allow them to be open freely. The other side of the doors is equipped with a handle.

Both cockpit doors can be opened from outside by turning the handle counter-clockwise and pulling the door; vice versa pilot and co-pilot doors can be opened from inside by pressing and turning the handle counter-clockwise and pushing the door.

An actuator installed in the upper part of the fuselage maintains the door opened.

The cockpit doors are provided with microswitches that generate a message of caution in case one door (left or right) is not closed.

CABIN DOORS

The cabin doors are located on the left and right sides of the center fuselage. Two rails with four rollers located two on the top and two on the bottom of the door allowing to slide the door backwards and to access to the cabin.
Cabin doors open from outside by pulling and rotating the lever and sliding the door backwards; vice versa to open from inside need to be pressed the safety pin, near the handle and sliding the door backwards.

The cabin doors are provided with microswitches that generate a message of caution in case one door (left or right) is not closed.

BAGGAGE DOOR

The baggage door is located on the left side of the rear fuselage and is hinged to the structure on the upper side. A

handle located at the bottom of the door allows to open the door. The right baggage door is furnished as an option.
The left baggage door is provided with a microswitch that generates a caution message in case the door is not properly closed. (Also on the right baggage door is installed a microswitch when the right baggage door is installed).

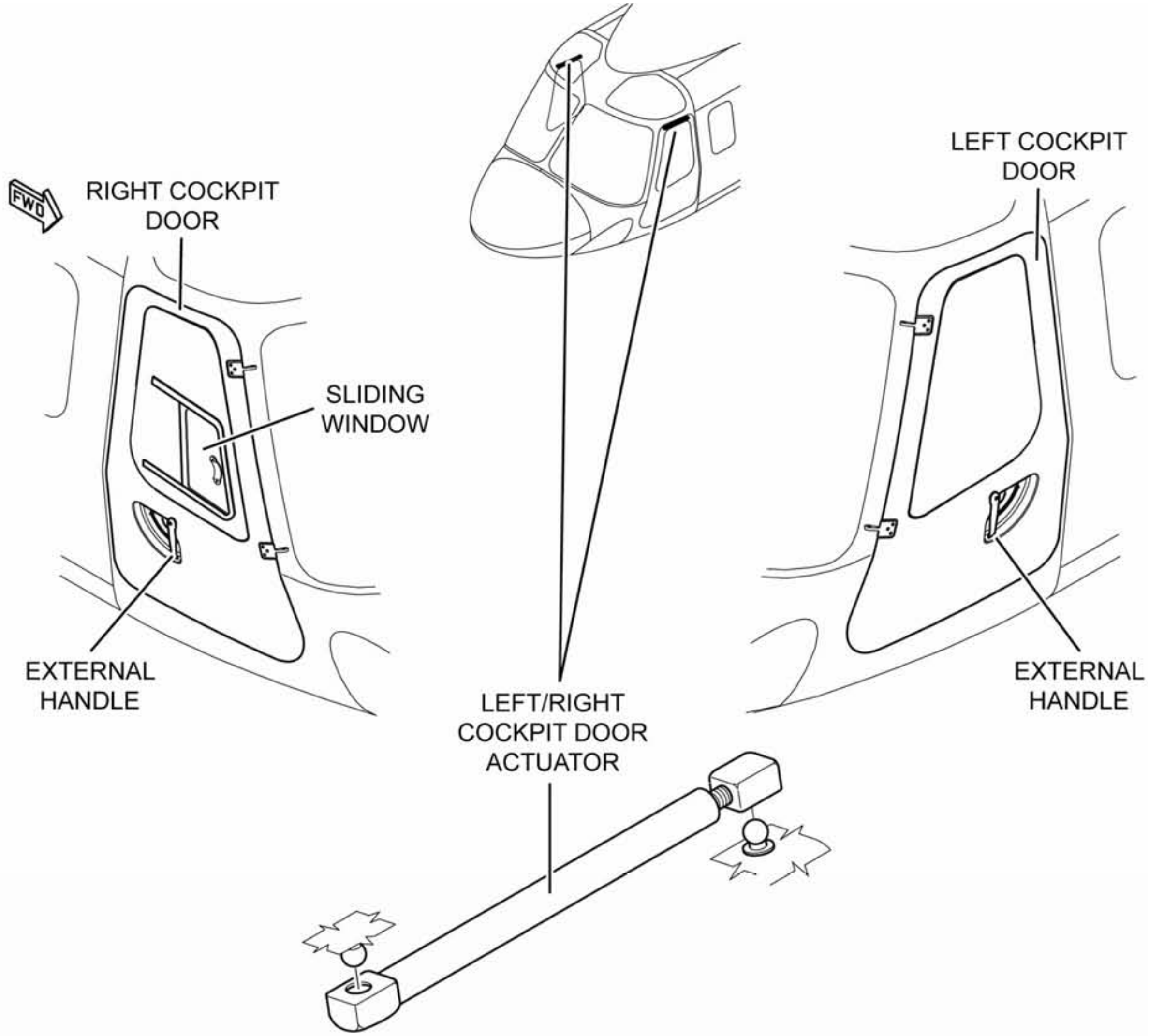
NOSE DOOR

The nose door provides access to the relevant compartment. The door is hinged to the fuselage on the bottom side. Centering pins installed on the upper side, secure the door when closed.
On the nose door no microswitches are installed.

EXTERNAL POWER DOOR

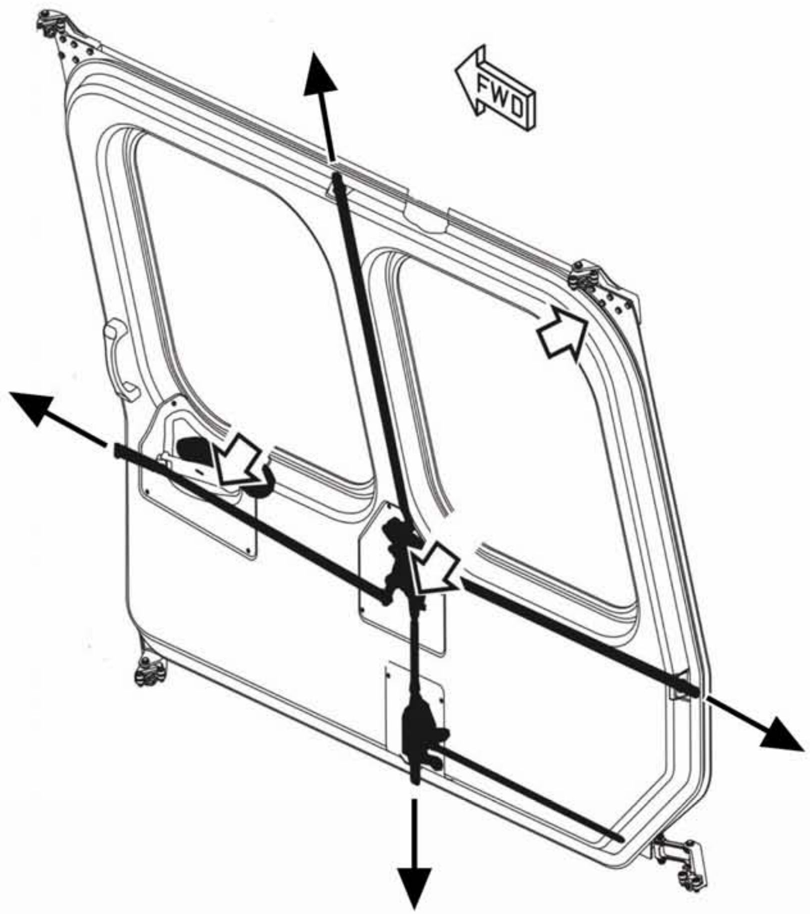
The external power receptacle is provided with a door that closes the access to the receptacle.

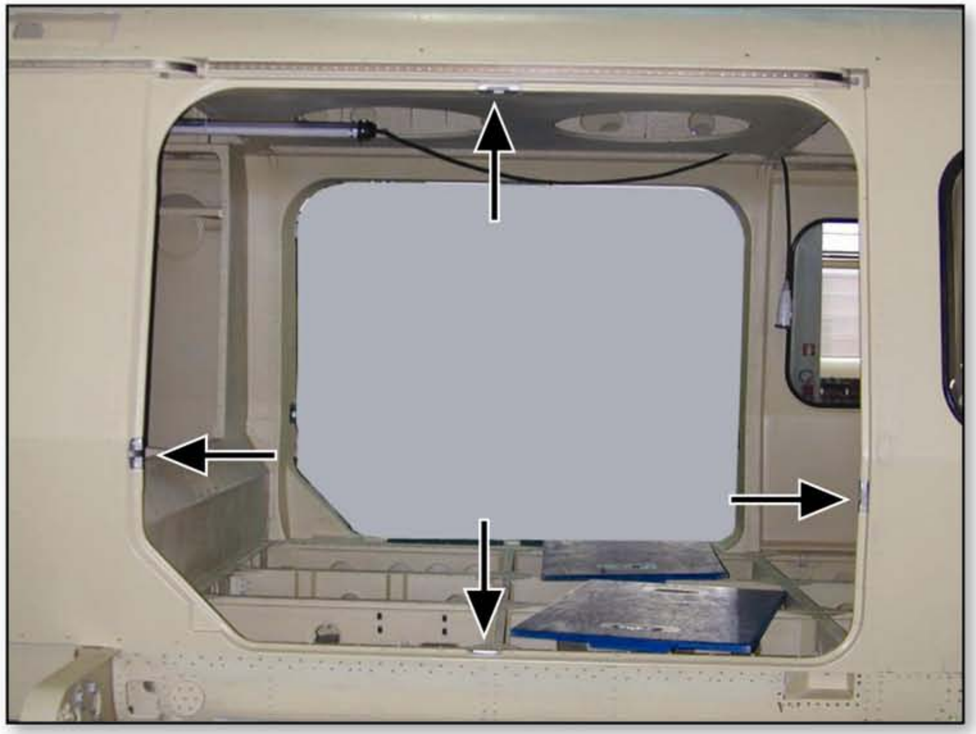
The door is provided with a microswitch that generates a caution message in case the door is not properly closed.



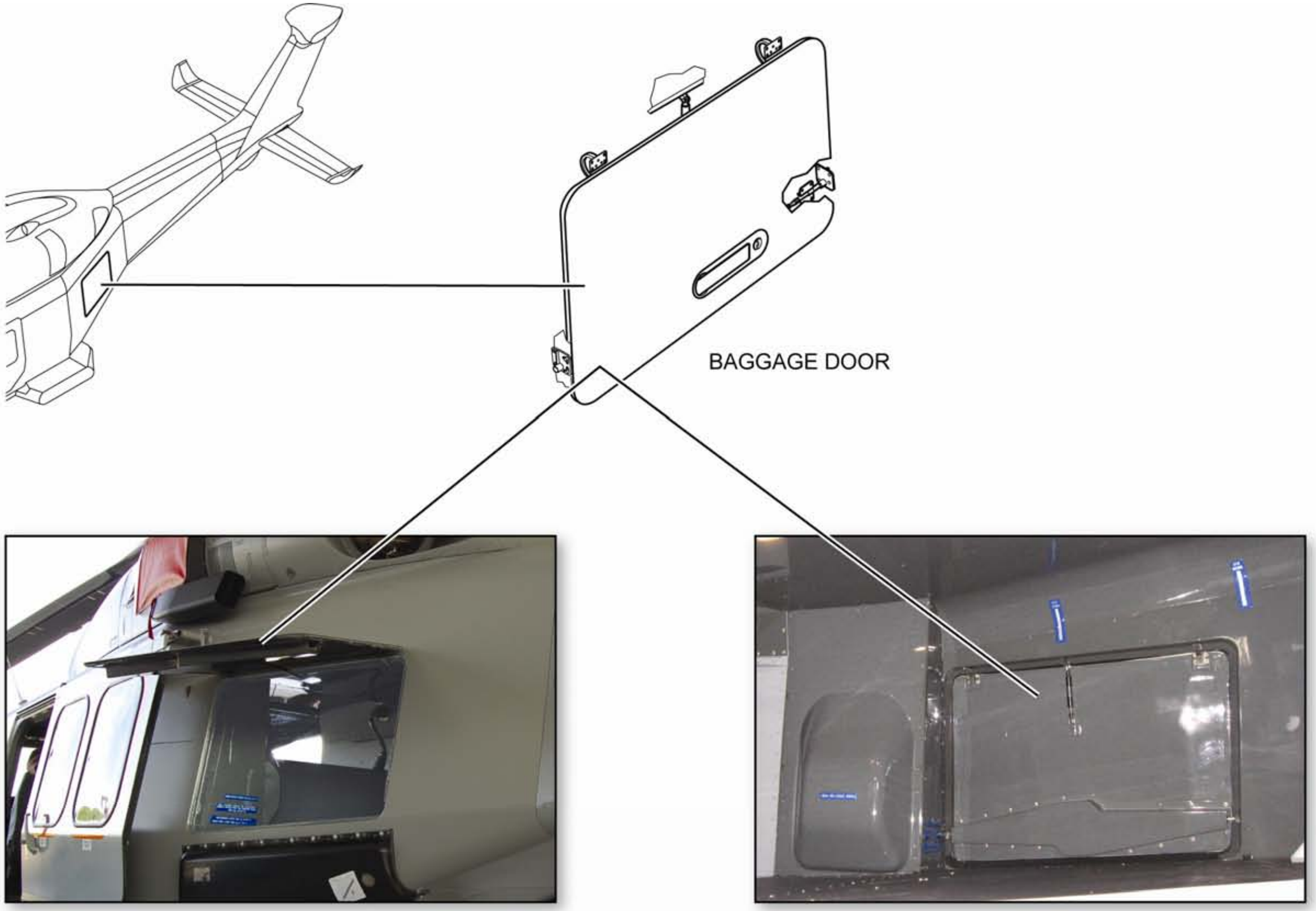
COCKPIT DOORS

CABIN DOORS

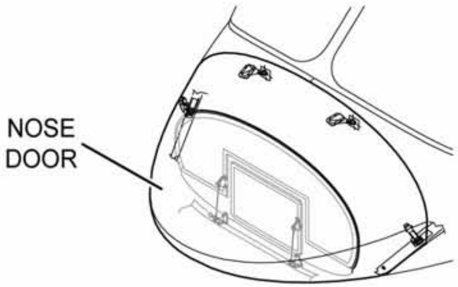




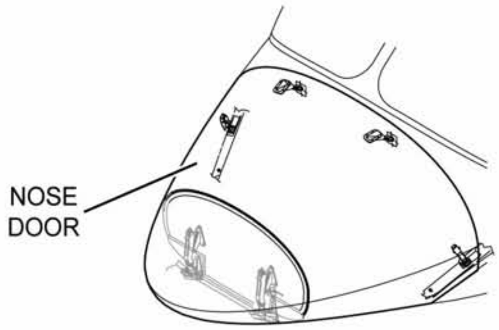
CABIN DOOR LOCKING MECHANISM



BAGGAGE DOOR

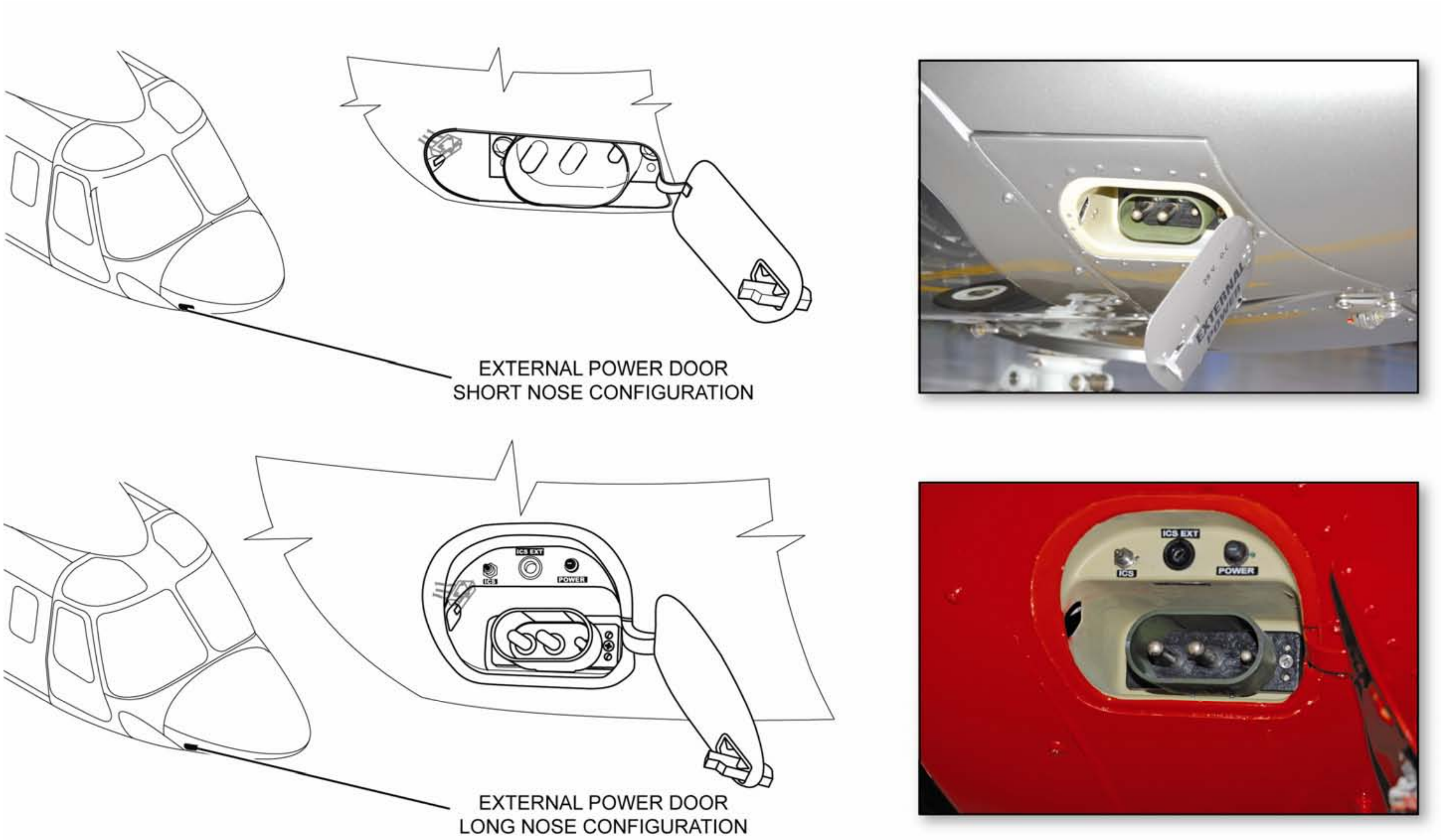


SHORT NOSE CONFIGURATION

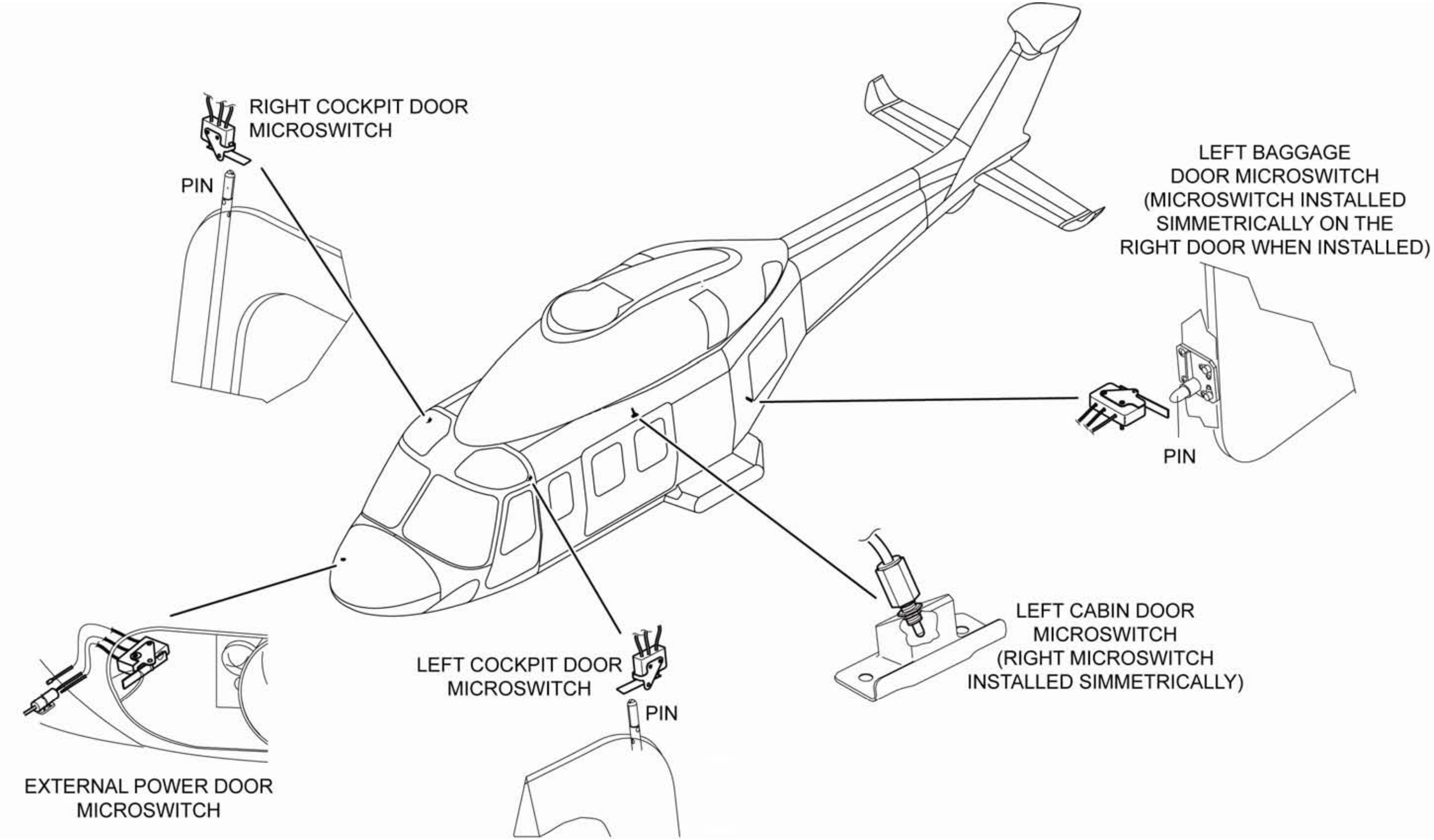


LONG NOSE CONFIGURATION

NOSE DOOR



EXTERNAL POWER DOOR



MICROSWITCHES LOCATION

DOORS - CAS CAUTION MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
COCKPIT DOOR	A cockpit door not closed	COCKPIT DOOR OPEN	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES MISCELLANEOUS
CABIN DOOR	A cabin door not closed	CABIN DOOR OPEN	
BAG DOOR	Baggage door not closed	BAGGAGE BAY DOOR OPEN	
EXT PWR DOOR	External power socket door not closed	EXTERNAL POWER SOCKET DOOR OPEN	

DOORS - LIMITATIONS

Refer to AW139-RFM-4D Sect.1.

WINDOWS – GENERAL

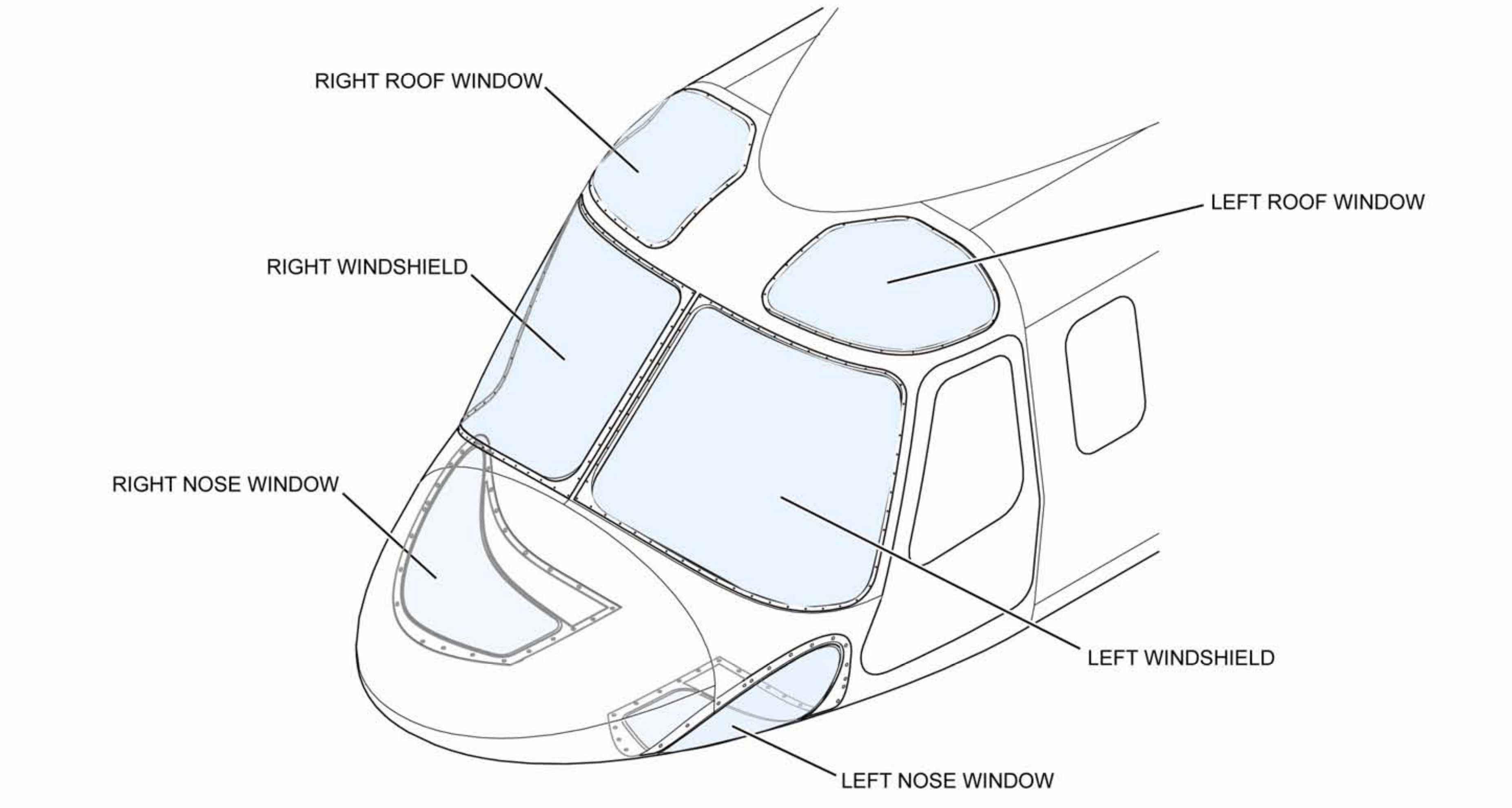
The helicopter is provided with windows in order to assure good visibility for crew and passengers. All windows are made of acrylic transparent material. Windows include:

- flight compartment windows
- fuselage compartment windows
- doors windows

FLIGHT COMPARTMENT WINDOWS

The windows of the flight compartment comprise:

- the left and the right windshield. The windshields are installed on the left (right) side of the cockpit and let the forward visibility to the crew.
- the left and the right roof window. The roof windows are installed on the top side of the cockpit and let a good visibility through the top side of the helicopter.
- the left and the right nose window. The nose windows are installed on the bottom side of the nose structure and let a good visibility through the bottom side of the helicopter.



FLIGHT COMPARTMENT WINDOWS

FUSELAGE COMPARTMENT WINDOWS

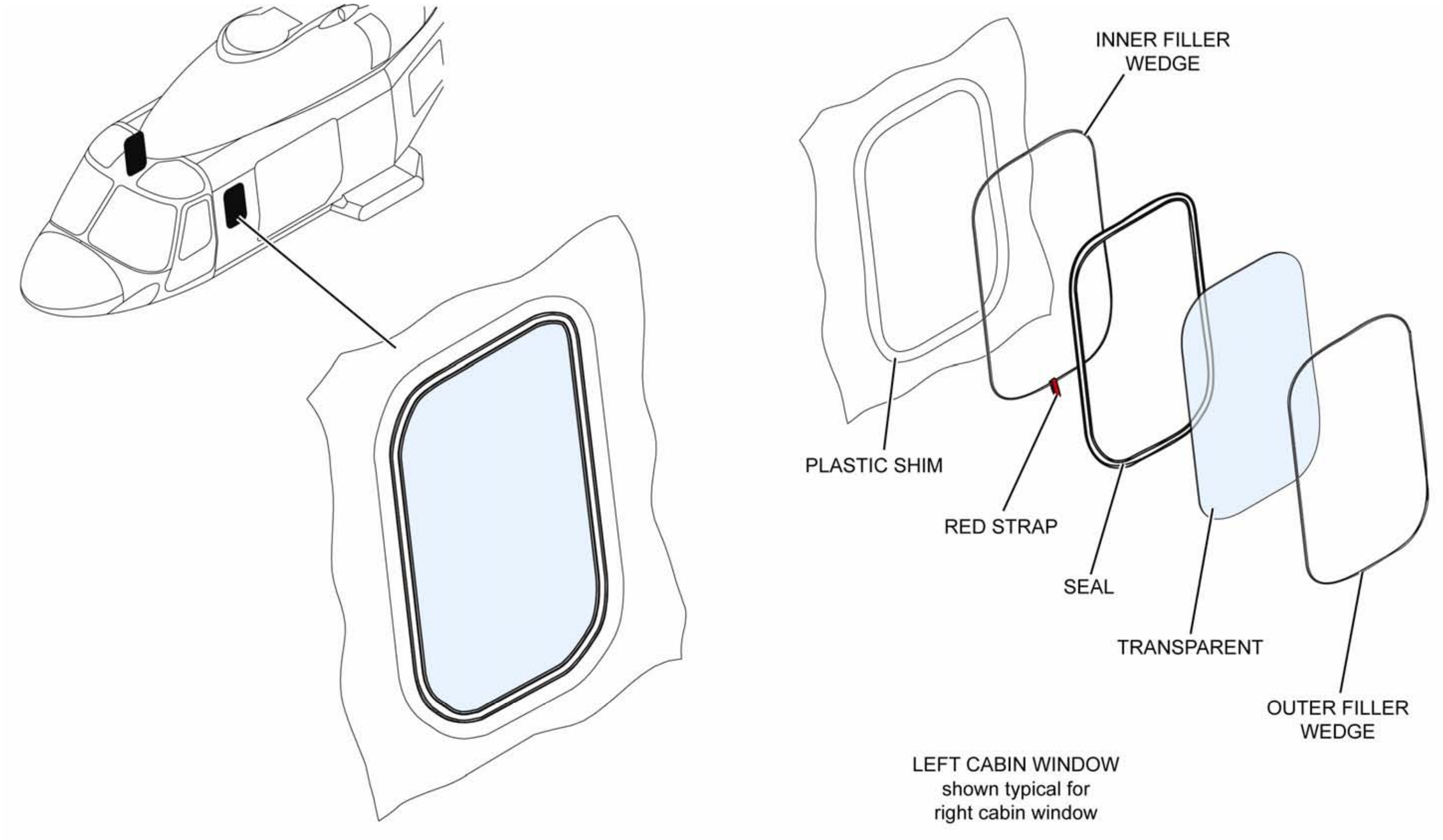
The windows of the fuselage compartment include:

- the left and the right cabin windows.

The cabin windows are installed on both sides of the cabin and are made of transparent material that lets a good visibility to passengers inside the aircraft.

A plastic shim, two filler wedges and a seal hold the transparent in its position. A red strap is installed on the internal side of the inner filler wedge and is attached to a clinch stud.

The cabin windows can be removed in an emergency. To do this, it is necessary to remove the red strap from the clinch stud and pull it. This causes the removal of the inner filler wedge and the seal: consequently the window disengages from the frame and falls out when pushed out.



CABIN WINDOWS

DOORS WINDOWS

The door windows comprise:

- the left and the right cockpit doors windows
- the left cabin doors forward and aft windows
- the right cabin door forward and aft windows

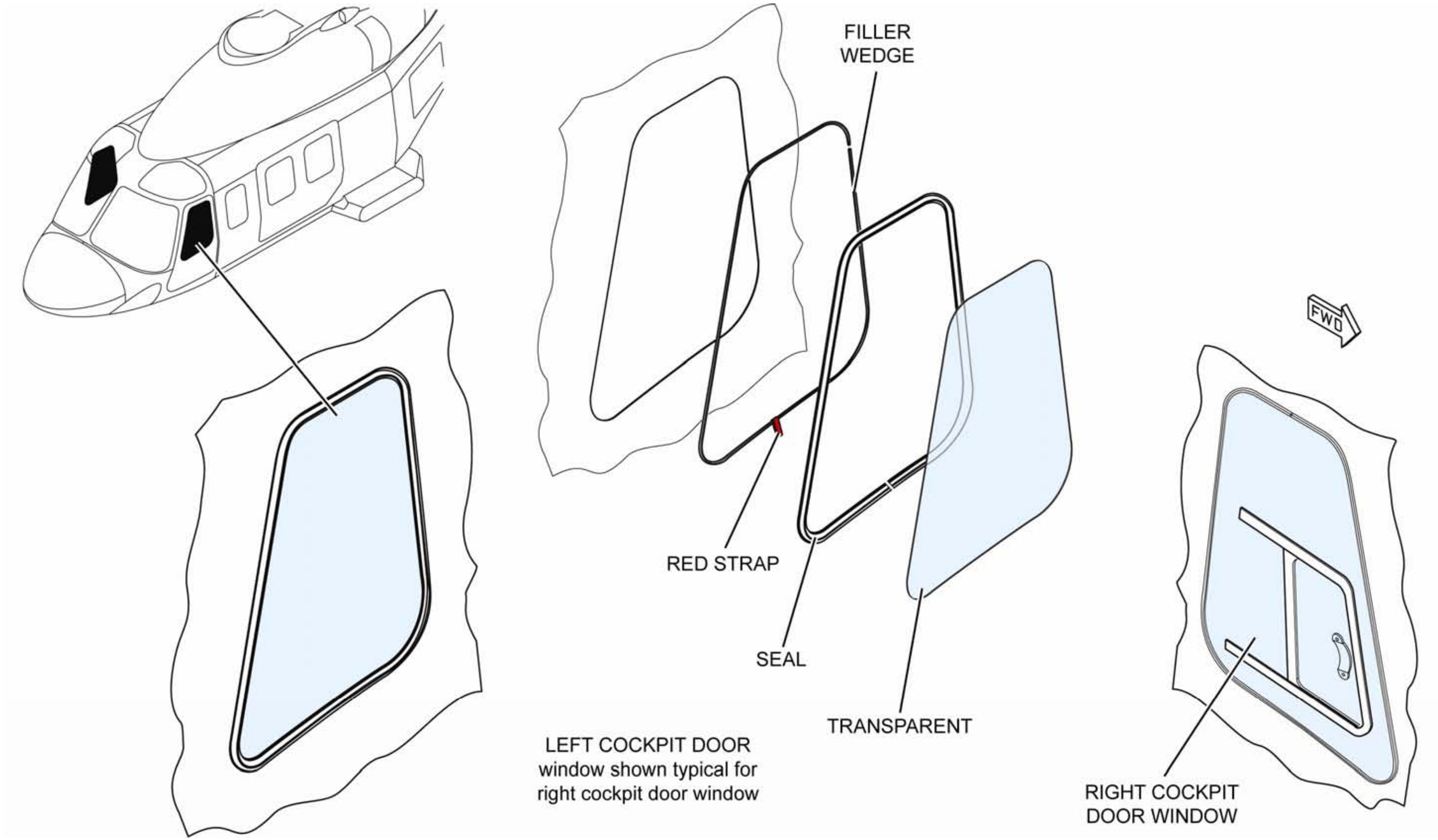
The left cockpit door window is installed on the left cockpit door. The window gives the visibility to the copilot and can be furnished with an additional sliding window (optional).

The left cockpit door window can be removed in an emergency. To do this it is necessary to remove the red strap and pull it: the window falls out when pushed.

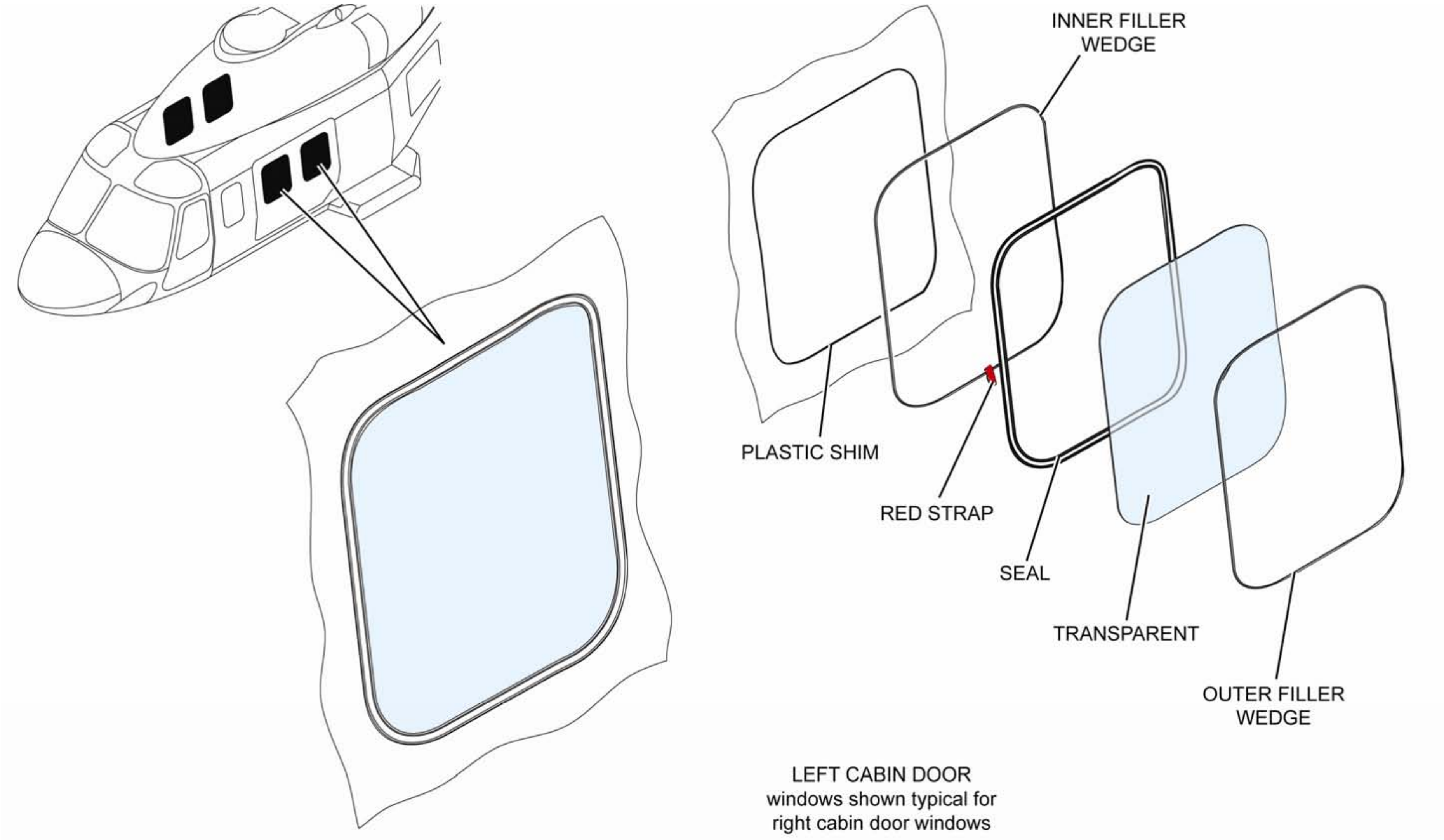
The same installation is made for the right cockpit door window.

The left cabin door has two windows (one forward and one at the rear) installed on the left cabin door. The windows are made of transparent material that lets a good visibility to the passengers on the left side of the helicopter. As for the cockpit doors windows, the two cabin windows can be removed in an emergency in the way already described.

The same installation is made for the two right cabin door windows.



COCKPIT DOORS WINDOWS



CABIN DOORS WINDOWS

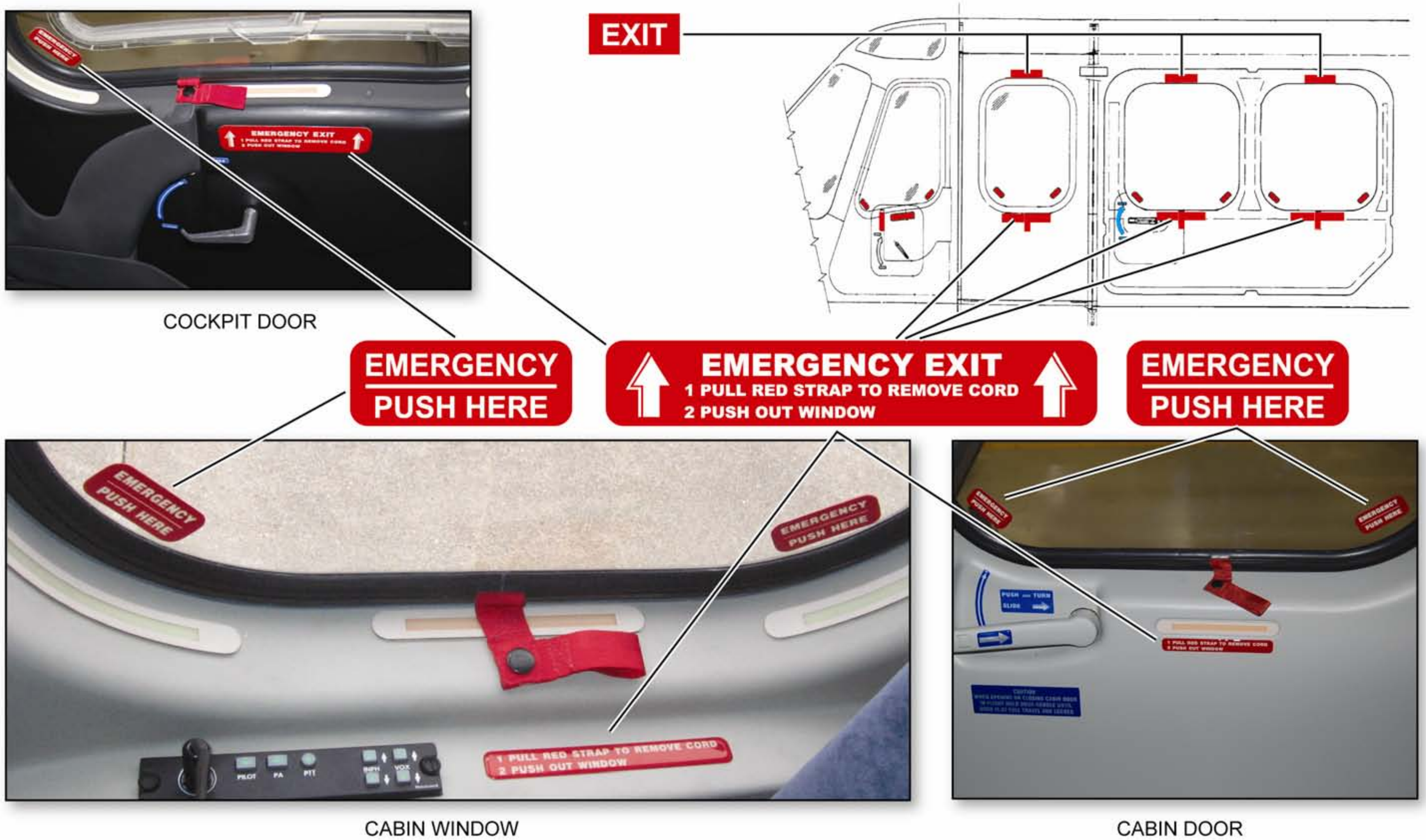
EMERGENCY EXIT

Emergency exits are located on cockpit and cabin windows.

The emergency exits of pilots and passengers are provided by the acrylic plastic windows. There are two windows in each large sliding door. The windows are jettisonable from the inside to the outside of the helicopter.

Each transparent is held in place with a silicone rubber seal. Applying hand pressure is enough to jettison the window.

EMERGENCY EXITS (1 OF 2)



EMERGENCY EXITS (2 OF 2)

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CHAPTER
62
MAIN ROTOR

SECTION 00 – GENERAL

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MAIN ROTOR – GENERAL

The rotors provide for lift and thrust necessary for flight and include:

- the Main Rotor (MR);
- the Tail Rotor (TR).

The Main Rotor (MR) provides for aircraft lift and forward flight. The MR comprises:

- the main rotor head;
- the blades;
- the rotating controls and swashplate assembly.

The main rotor is a fully articulated type rotor with 5 blades.

MAIN ROTOR - MAIN COMPONENTS

MAIN ROTOR HEAD

The main rotor head is driven by the main rotor shaft and includes the hub, the tension links, the elastomeric bearings, the dampers, the control levers and the beanie.

HUB

The hub connects the blades to the MR shaft. It is made by titanium and splined internally to fit the MR shaft. The hub transmits the rotation to the swashplate via the scissors links.

TENSION LINKS

The tension links connect the blades to the hub and transmit centrifugal forces from the blades to the hub through elastomeric bearings.

ELASTOMERIC BEARINGS

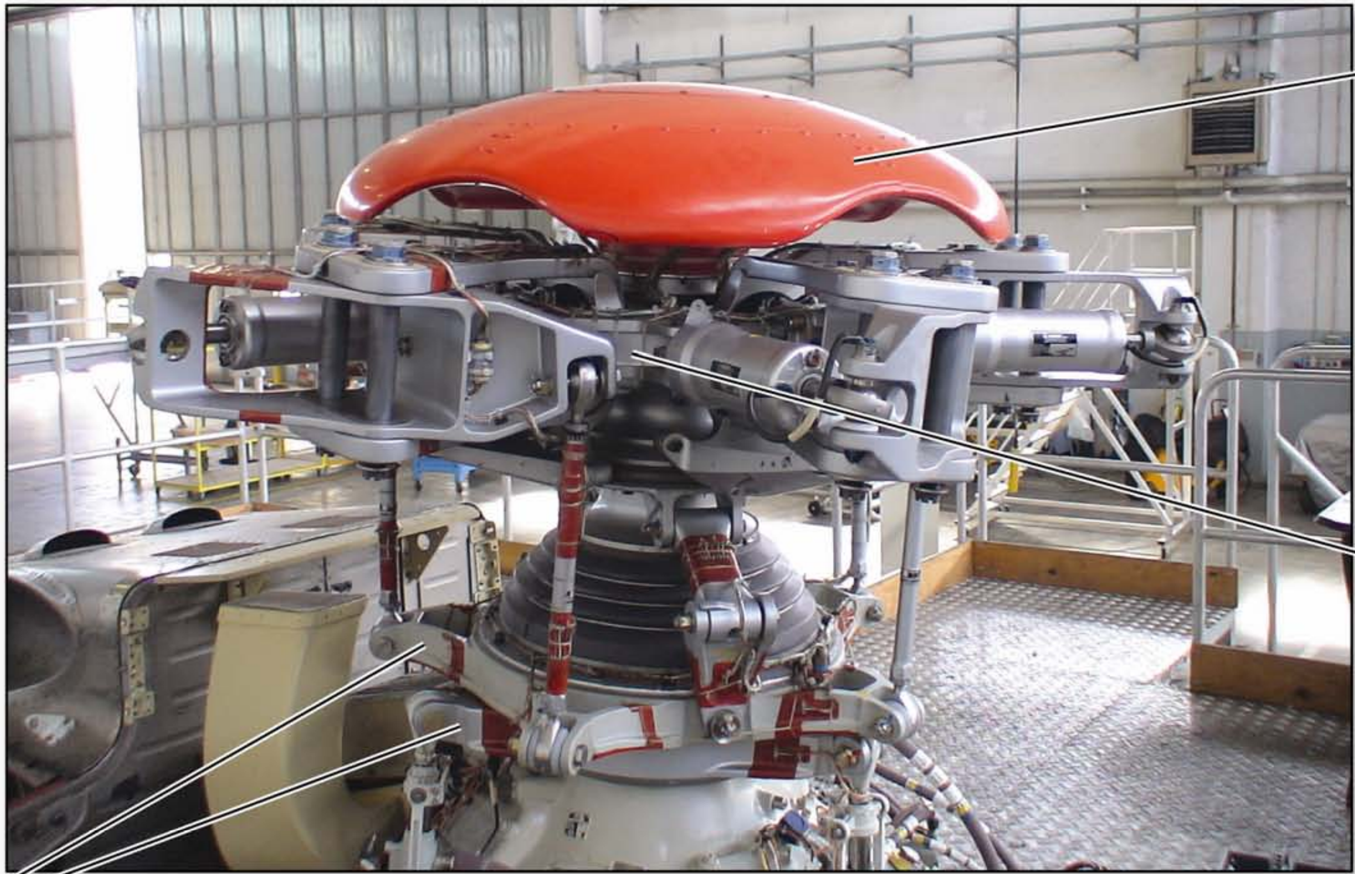
The elastomeric bearings allow a fully articulation of the blades around the flap, pitch and lag axis. They are made by rubber and metallic disks.

DAMPERS

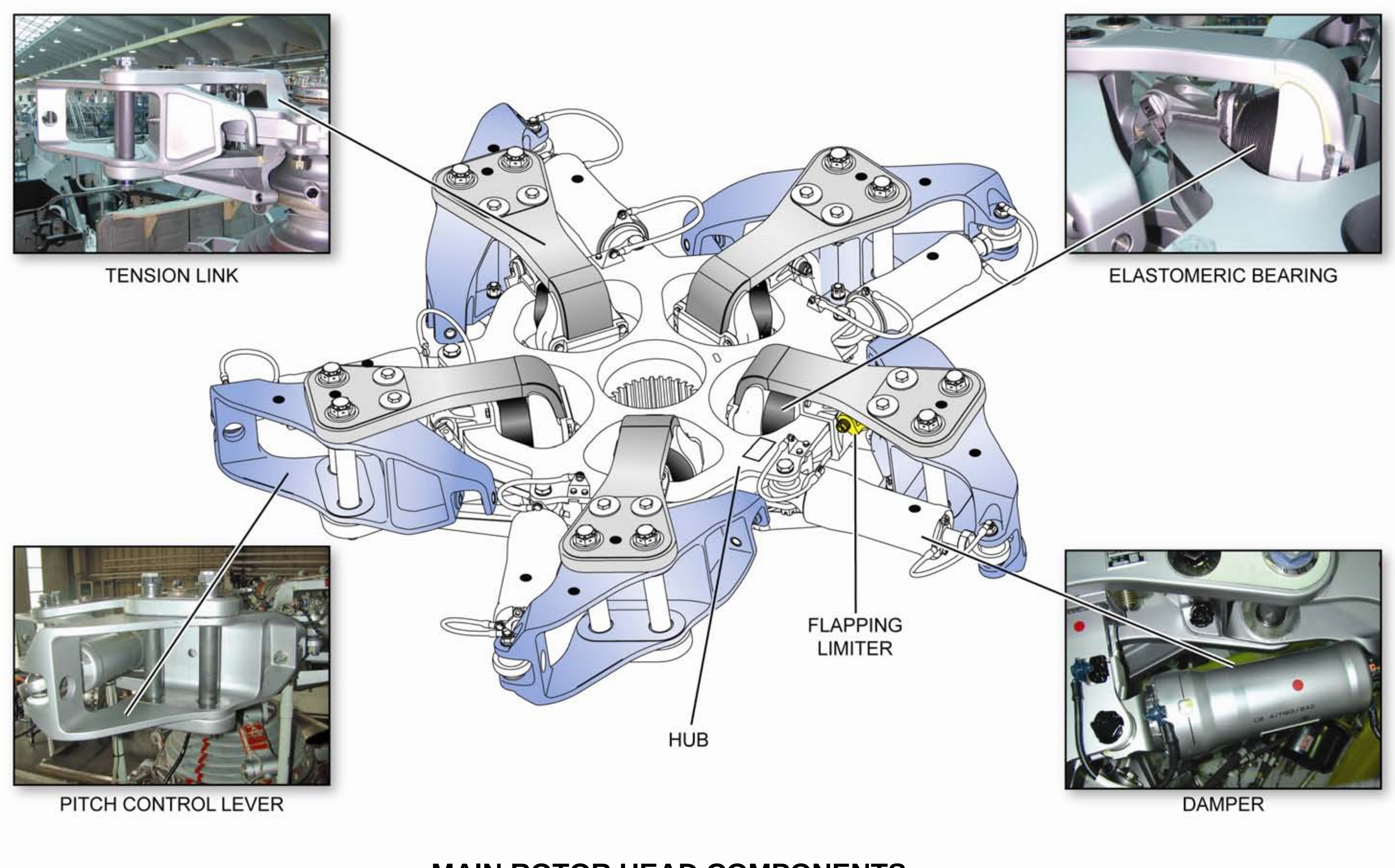
The hydraulic dampers react to lead lag motion and provide also the stops. A separated stop system is provided for flapping: it includes the upper stop and the lower stop. The first is provided with a support (for flight) and a limiter (for ground); the second is made by a bracket with a sliding ring at one end.
The pitch control levers provide the connection points between the MR head and the swashplate assembly. They transmit flight control input to the blades.

BEANIE

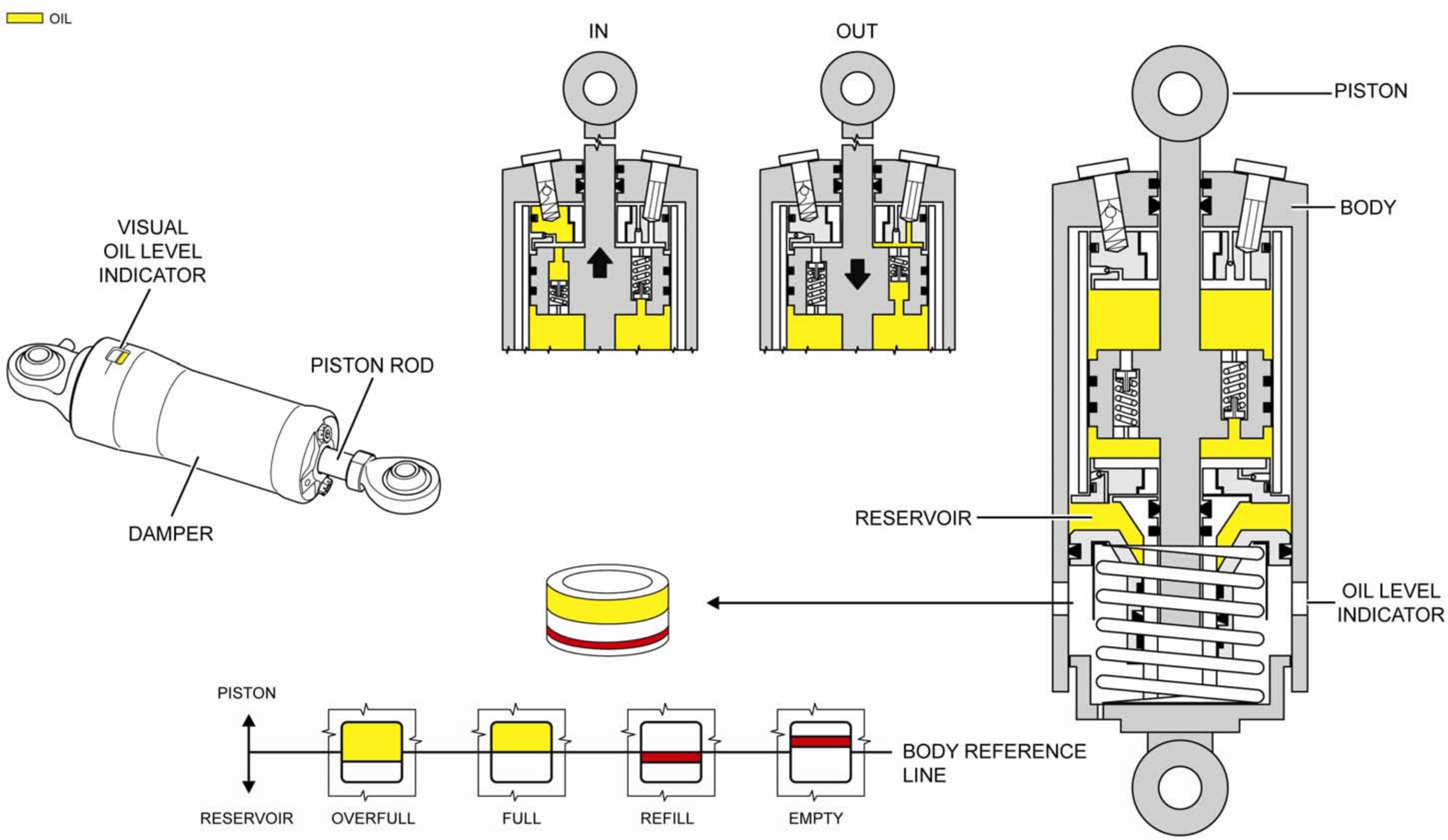
An aerodynamic beanie is bolded to the top of the MR head.



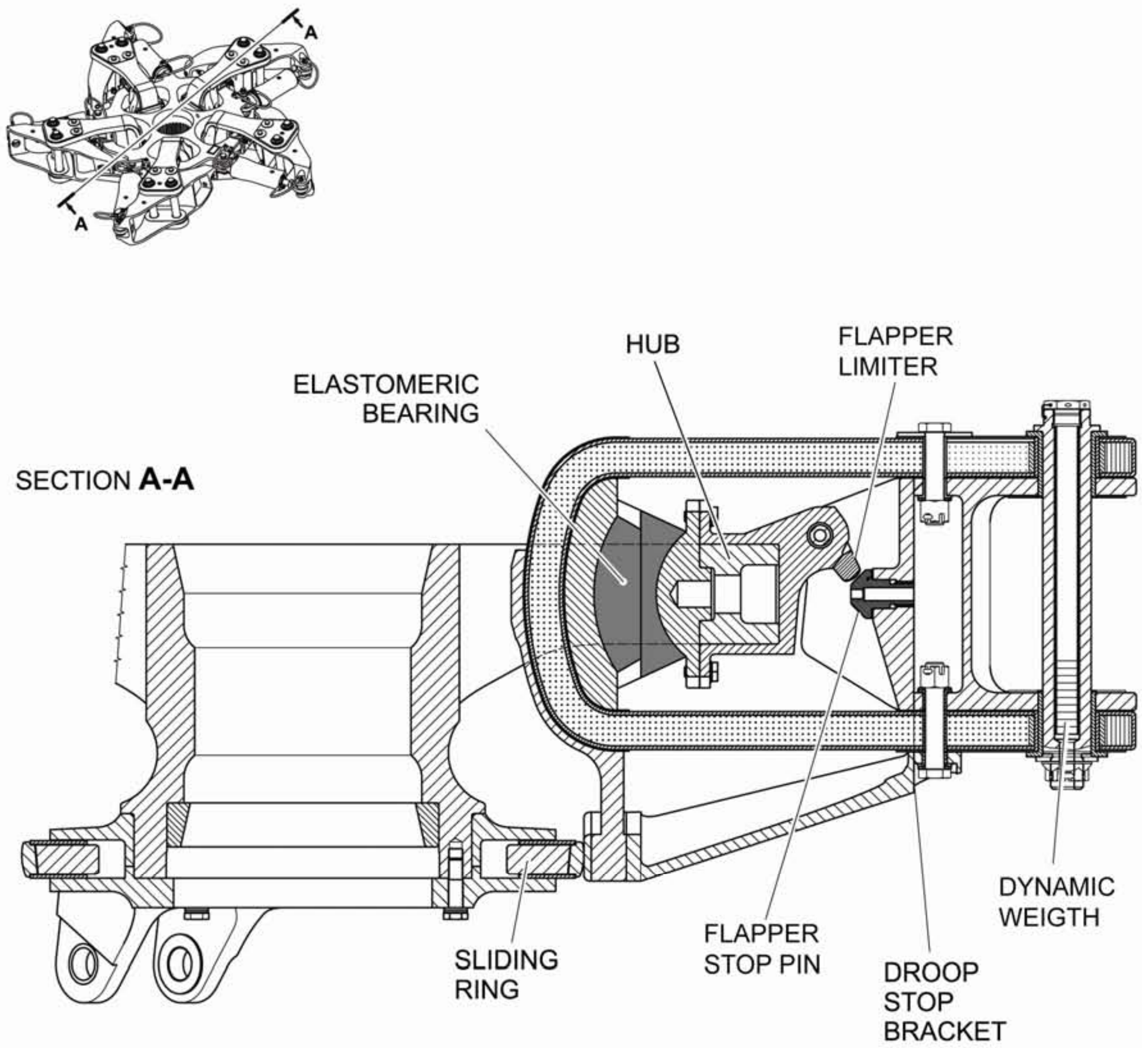
MAIN ROTOR



MAIN ROTOR HEAD COMPONENTS



MAIN ROTOR DAMPER



FLAPPER LIMITER ENGAGED

FLAPPER LIMITER DISENGAGED

MAIN ROTOR SECTION

BLADES

Each rotor blade generates an aerodynamic lift which depends on the angle of attach of the blade.

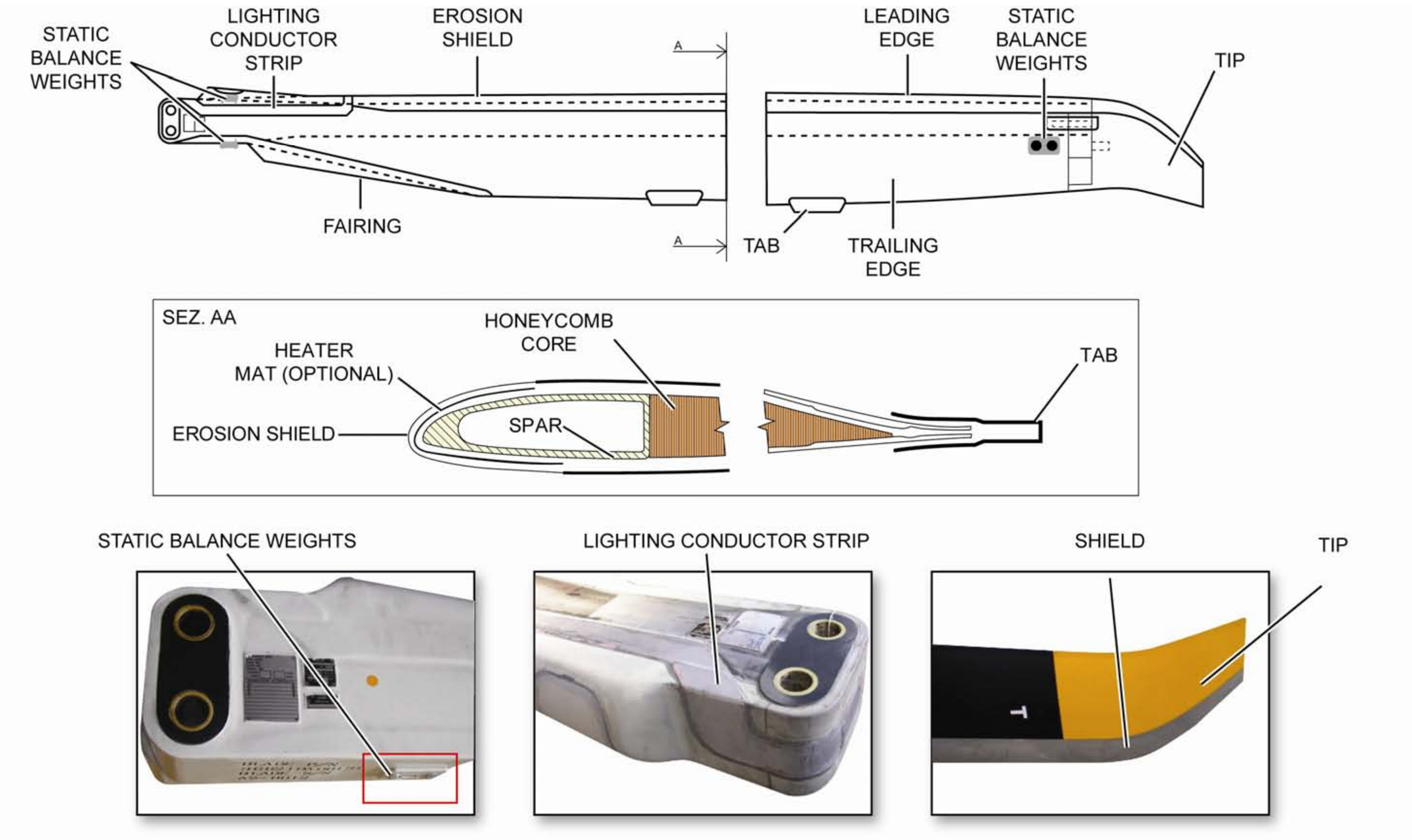
The blades are made of composite material. A stainless steel erosion shield is attached to the leading edge of the blade. The leading edge and the trailing edge of the blade root have fairings.

Mass-balance weights are used to balance the blade statically and are inserted in two holes: one is located near the blade root and one is located near the blade tip cap.

Two metallic trim tabs are attached to the trailing edge and can be bent as necessary to do the tracking of the main rotor and to balance the blade dynamically.

A lightning conductor strip is attached to the top surface of each blade root.

The blade can be heated by means of an optional heater mat installed behind the leading edge of the blade.


STATIC
BALANCE
WEIGHTS
LIGHTING
CONDUCTOR
STRIP
EROSION
SHIELD
A
LEADING
EDGE
STATIC
BALANCE
WEIGHTS
TIP
FAIRING
A
TAB
TRAILING
EDGE
SEZ. AA
HEATER
MAT (OPTIONAL)
HONEYCOMB
CORE
TAB
EROSION SHIELD
SPAR
STATIC BALANCE WEIGHTS
LIGHTING CONDUCTOR STRIP
SHIELD
TIP

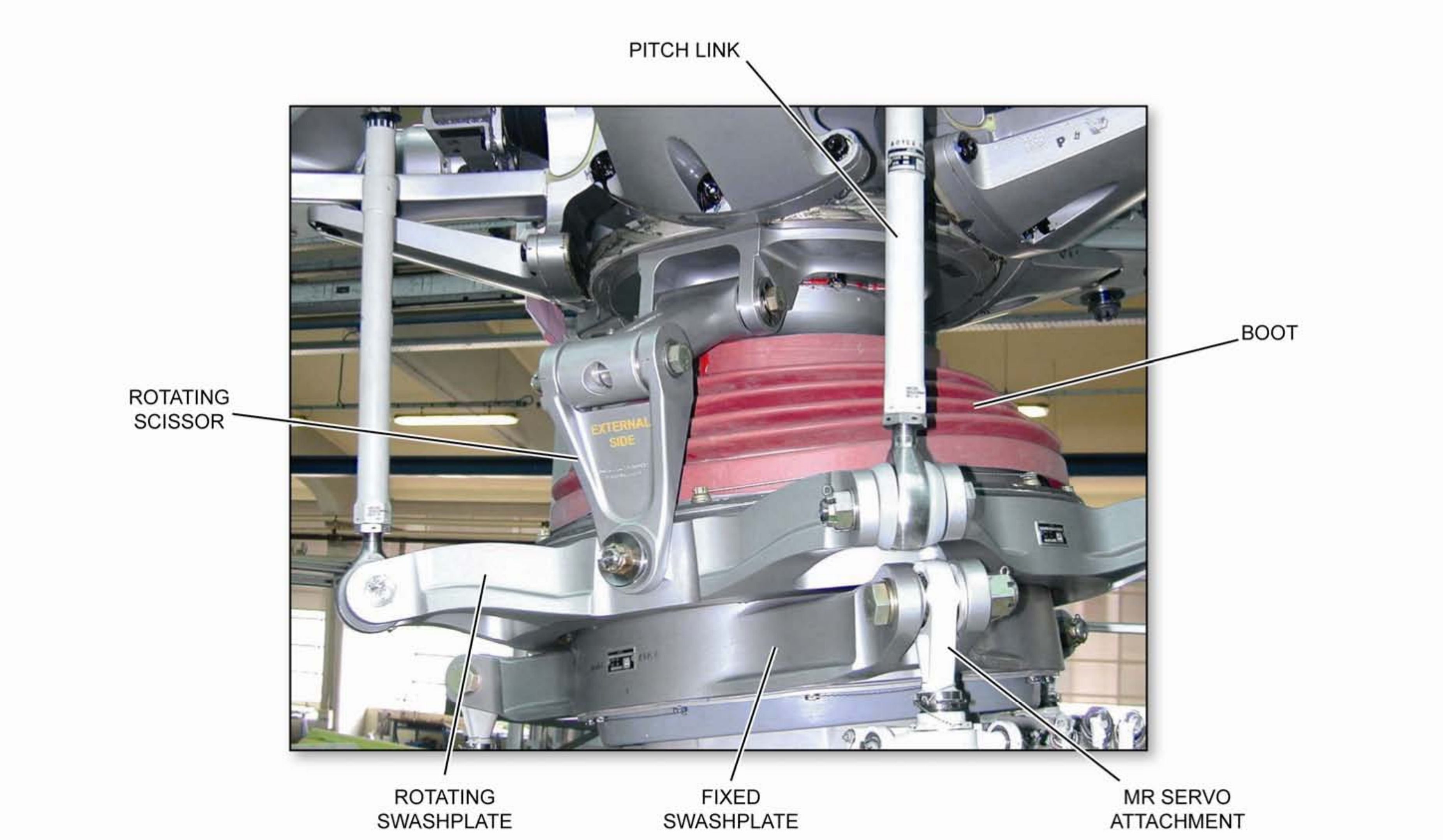
BLADES

ROTATING CONTROLS AND SWASHPLATE ASSEMBLY

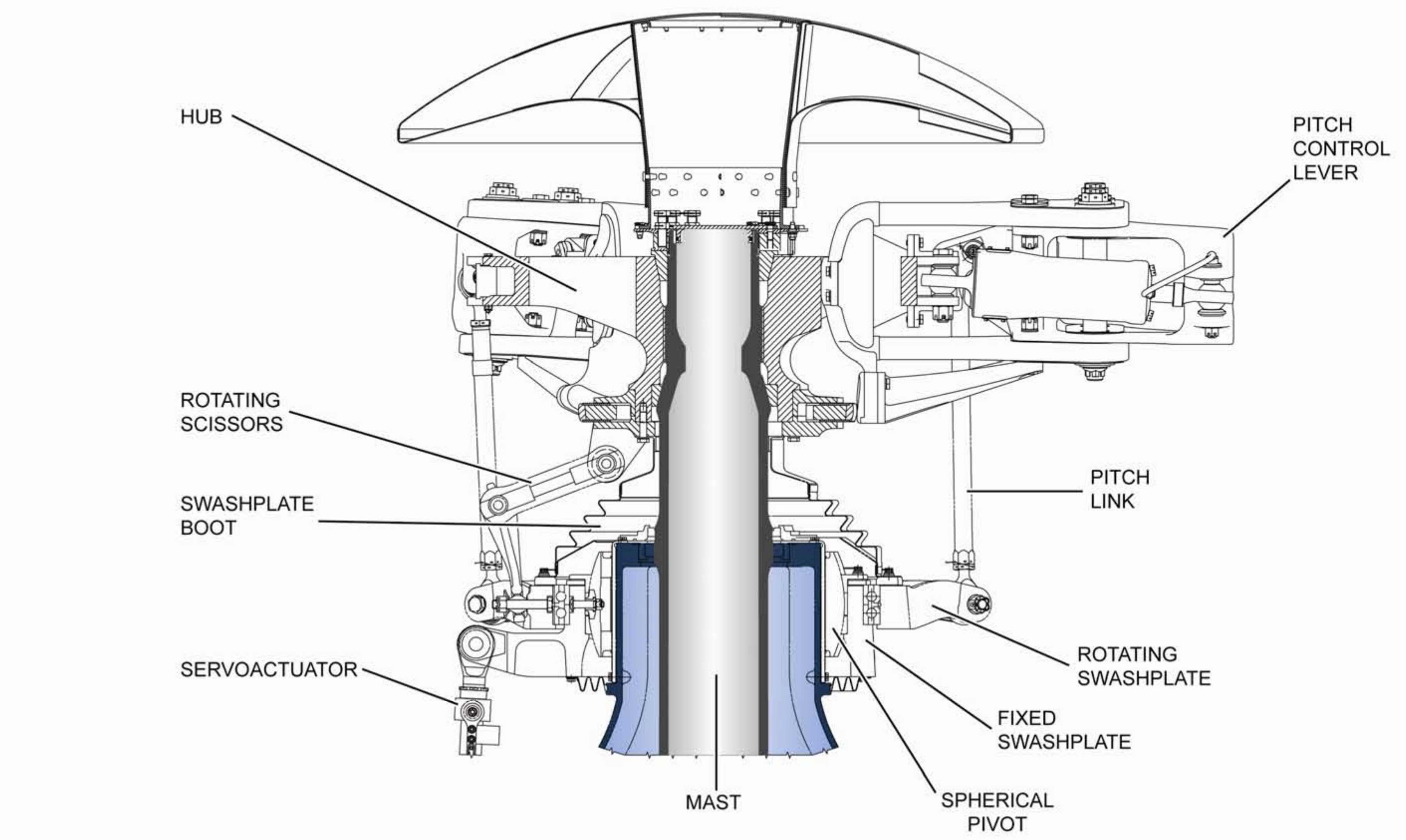
The main rotor interfaces the flight control system via the rotating controls and swashplate assembly. The rotating controls and swashplate assembly provides the attachment points for the three MR servo actuators and transmits the flight controls output coming to the blades.

The rotating controls include:

- five pitch links, one for each blade, connecting the pitch control levers of the MR blades to the swashplate;
- two rotating scissors assemblies installed between the MR hub and the rotating swashplate connecting the hub (scissors attachment flange) to the rotating swashplate and keeping it rotating. Each assembly is composed of an upper lever and a lower lever hinged to the swashplate through a ball bearing;
- two adapters installed between the MR head and the swashplate. A swashplate boot is attached to the top and bottom adapters with metallic clamps;
- the swashplate assembly installed on the top of the MGB between the MR head and the three MR servo-actuators and composed by rotating and stationary stars. The relative rotation is allowed by a double raw of ball bearing. The stationary star receives the control input from the MR servo-actuators (fixed controls) and transmits it to the rotating star. To do this the swashplate is tilted with respect to the axis of a central pivot element.



ROTATING CONTROLS AND SWASHPLATE ASSEMBLY (1 OF 2)



ROTATING CONTROLS AND SWASHPLATE ASSEMBLY (2 OF 2)

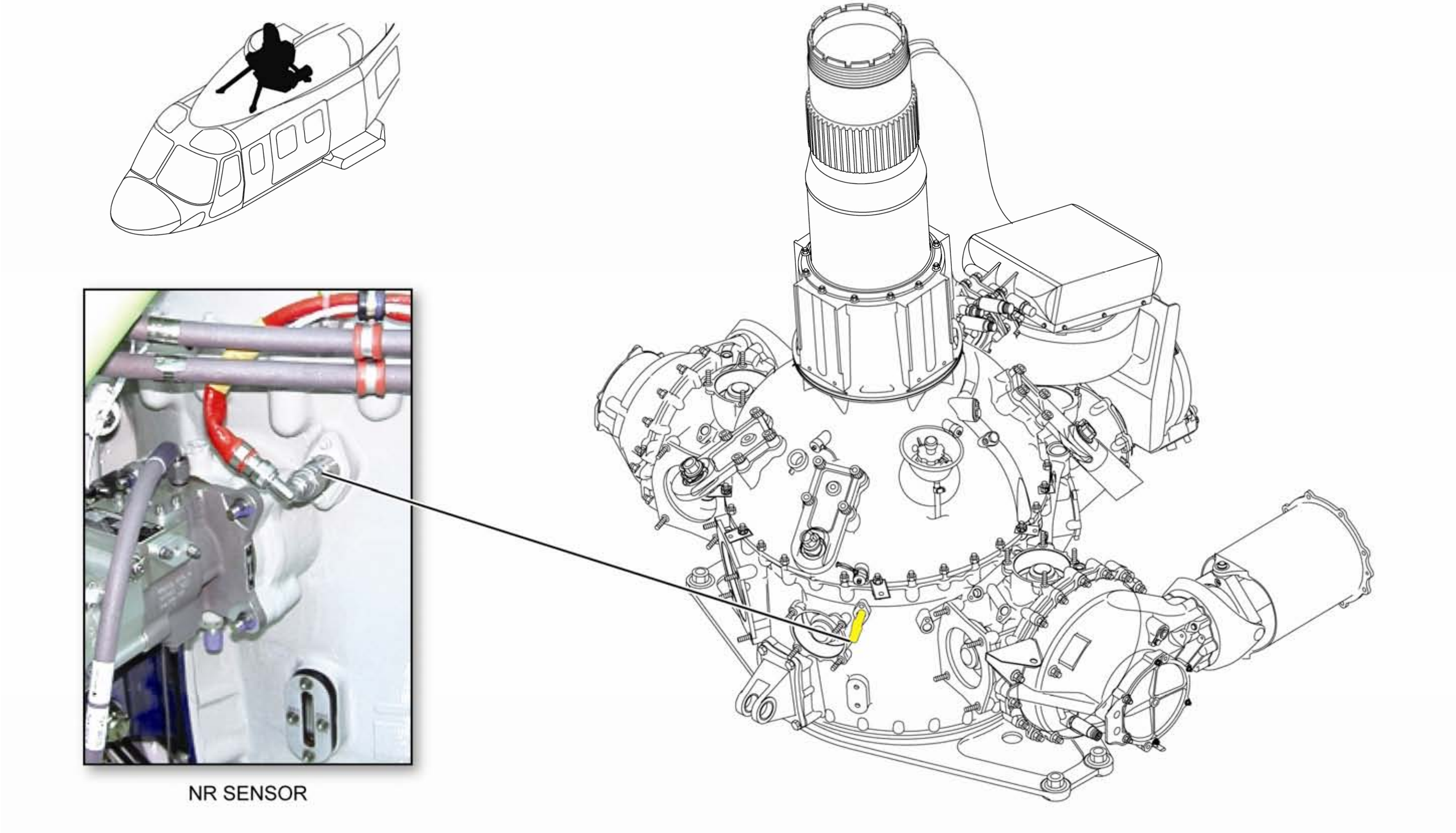
NR SENSOR

A rotor speed probe is used to provide an output for the indicating of the main rotor speed. This probe provides three independent analogue outputs (three independent coil windings) directly related to the speed of the main rotor. Each signal is used by the EEC1 (Electronic Engine Control), EEC2 and MAU1 (Modular Avionics Unit) and MAU2.

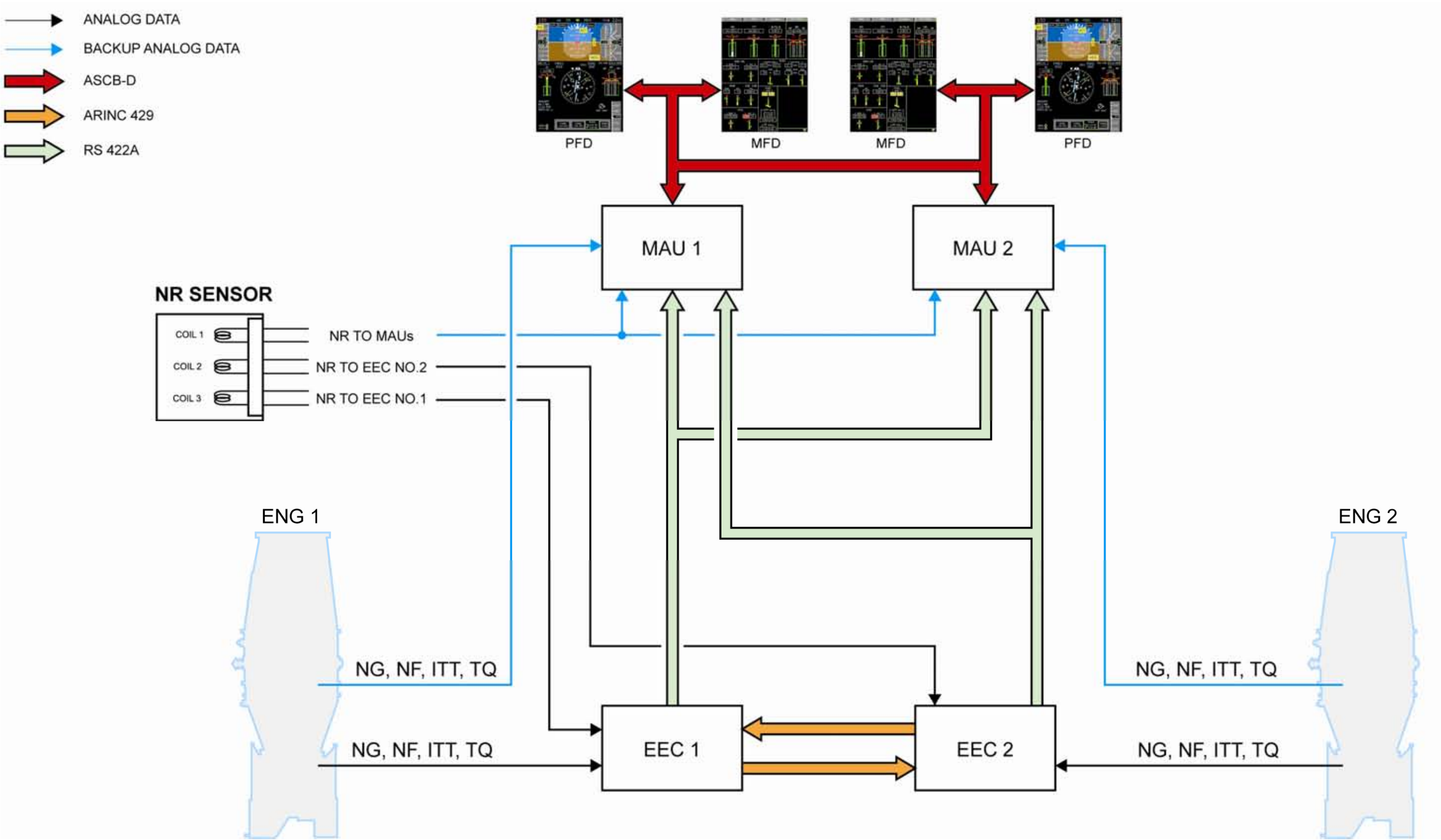
The analogue NR information received by the sensor are digitalized inside the EEC and addressed to the MAUs via the ARINC 429.

A dedicated function of the MAUs permits to perform a continually compare between the digital signal received from the EECs and the dedicated signal received from the NR sensor.

In case of a miscompare between analog data and digital data, the caution message NR MISCOMPARE will be generated in the CAS window.



NR SENSOR



NR SENSOR OPERATIONS

MAIN ROTOR - INDICATIONS

The PFD default page and the MFD MAIN page display the values of NF (engine free turbine speed) and NR grouped in the TRIPLE TACHOMETER scale.

1. NF/NR (Power ON)

   The values of NF in the engine no.1 and no.2 are represented by digital readouts under the two NF labels on the left and on the right respectively. Graphically these values are represented on a vertical scale by means of two pointers (triangles) that match the color of the area on the scale.

   The digital readouts and the pointers are displayed red when the pointer is in the red zone (warning), amber when the pointer is in the amber zone (caution) and green in normal conditions.

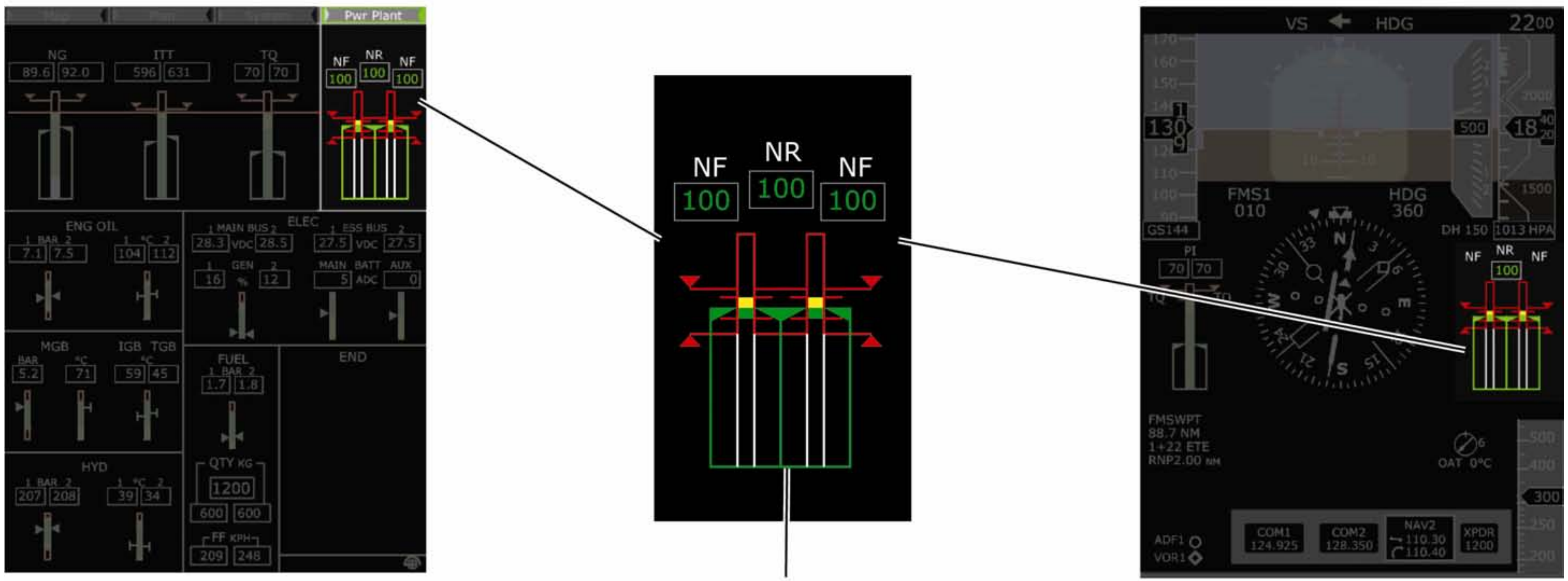
   A red horizontal line is displayed corresponding to the MIN TRANSIENT NR 95% and the MAX TRANSIENT NR/NF 106%.

When the EECs and the Modular Avionic Units (MAUs) detect an invalid NR/NF input signal, the associated pointer is removed from the display and amber dashes replace the numerical readouts.

If the NF sensor fails, the relative pointer is removed and replaced by the amber legend FAIL in reverse video.

The baseline of the triple tachometer is green at all times, even when the pointers are amber or red.

When a parameter being monitored exceeds the normal range of operation (green band), the colour of associated pointer matches the colour of applicable range marking in order to highlight the particular critical condition.



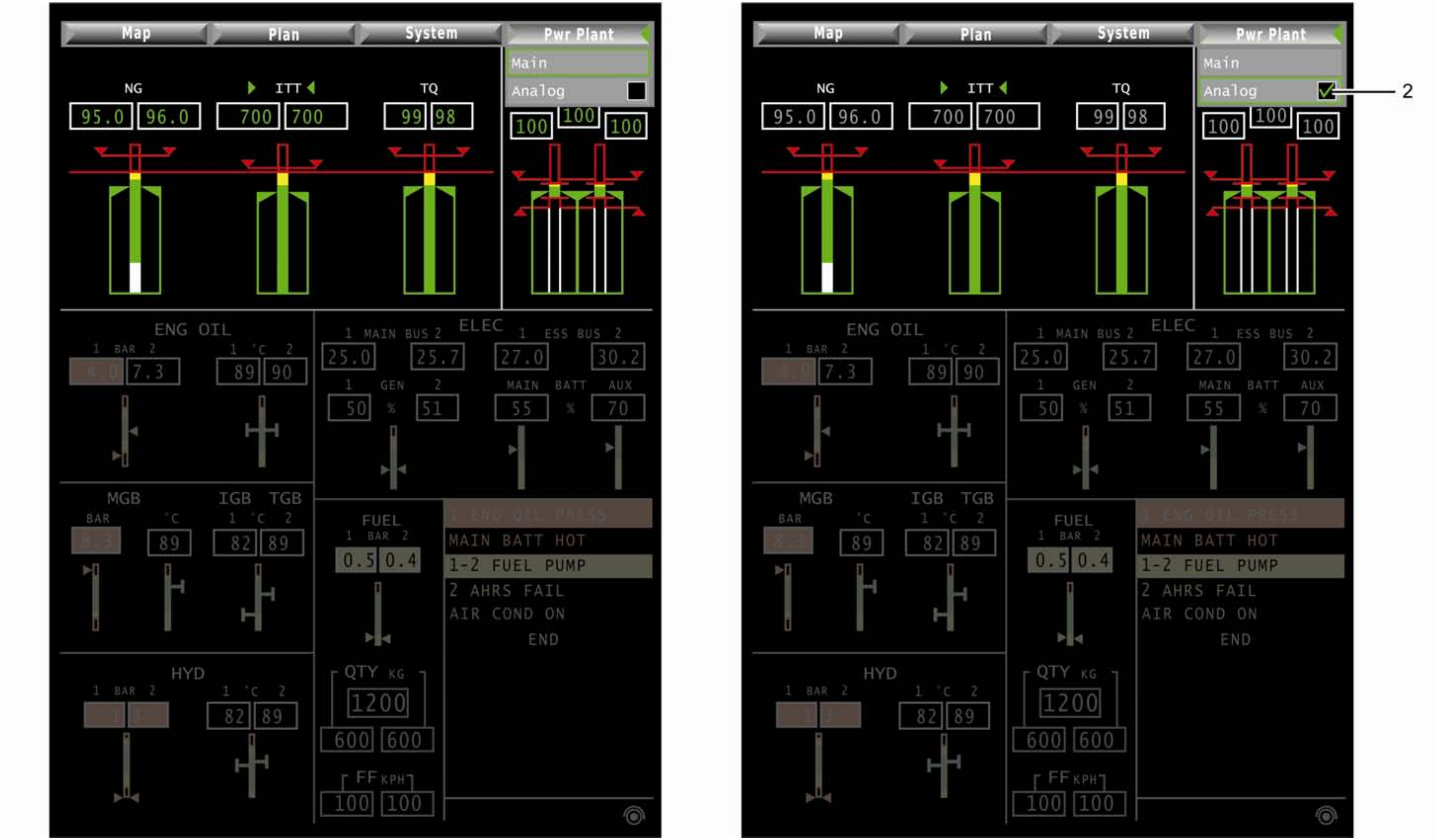
MFD MAIN PAGE 1 PFD DEFAULT PAGE

MAIN ROTOR INDICATIONS

PRIMARY DATA / BACKUP DATA INDICATIONS

Primary data of NF and NR come from the relative Electronic Engine Control units through the MAUs and are displayed in green.

When ANALOG is selected to be the NR data source, the analogue read-out numbers change the colour in white and the displayed data (backup data) comes from the relative sensors through the MAU.



PRIMARY DATA / BACKUP DATA INDICATIONS

MAIN ROTOR – CONTROLS AND INDICATORS

1. 100% / 102% RPM selector switch on the pilot collective grip

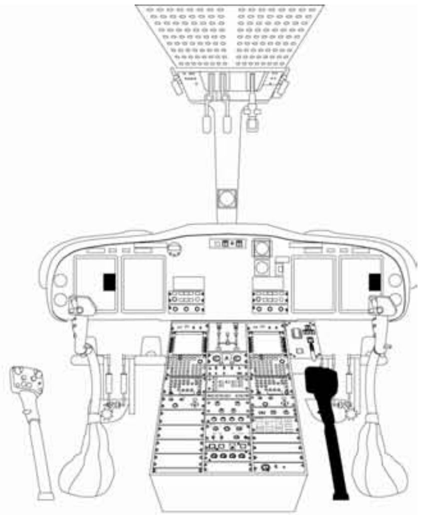
   100% …………… set the NR / NF to 100%

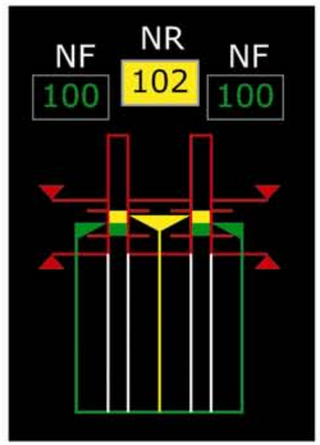
   102% …………… set the NR / NF to 102%

   NOTE. With RPM selected at 102%

   - VNE is 90 kts
   - NR reach yellow range
   - There are NO CAS messages during this type of selection

PILOT COLLETIVE ONLY





MAIN ROTOR– CONTROLS AND INDICATORS

CAS WARNING MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
ROTOR LOW	Rotor RPM below limit • when the NR < 98 % in power-on condition • when the NR < 95 % in power-off condition • when the NR < 90 % in power-on condition (OEI) 3 tones + voice warning: ROTOR LOW-ROTOR LOW This sequence is continuously repeated until the failure condition is corrected or the reset input activates	ROTOR UNDER-SPEED	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES ROTOR XMSN
ROTOR HIGH	Rotor RPM above limit • when the NR > 104 % in power-on condition • when the NR > 110 % in power-off condition • when the NR > 104 % in power-on condition (OEI) 3 tones + voice warning: ROTOR HIGH-ROTOR HIGH This sequence is continuously repeated until the failure condition is corrected or the reset input activates	ROTOR-OVERSPEED	

CAS CAUTION MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
NR MISCOMPARE	Discrepancy between EEC and analogue value of NR NOTE The caution is generated when comparison with backup parameter exceeds NR 3% This caution shall be displayed only if NR_{EEC} > 20% or NR_{BACKUP} > 20%	ENGINE AND ROTOR PARAMETERS MISCOMPARE	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES ENGINE
ENG ANALOG FAILURE	Failure of an analogue parameter	ENGINE ANALOGUE SENSOR FAILURE	
RPM SELECT	RPM selector switch failed	ROTOR SPEED SELECTOR	

ROTOR SPEED - LIMITATIONS

Refer to the AW139-RFM-4D Section 1.

NOTE.

Each rotor starting and stopping in wind speeds above 27 kts must be recorded in the helicopter log book.

CHAPTER
63
MAIN ROTOR DRIVE

SECTION 00 – GENERAL

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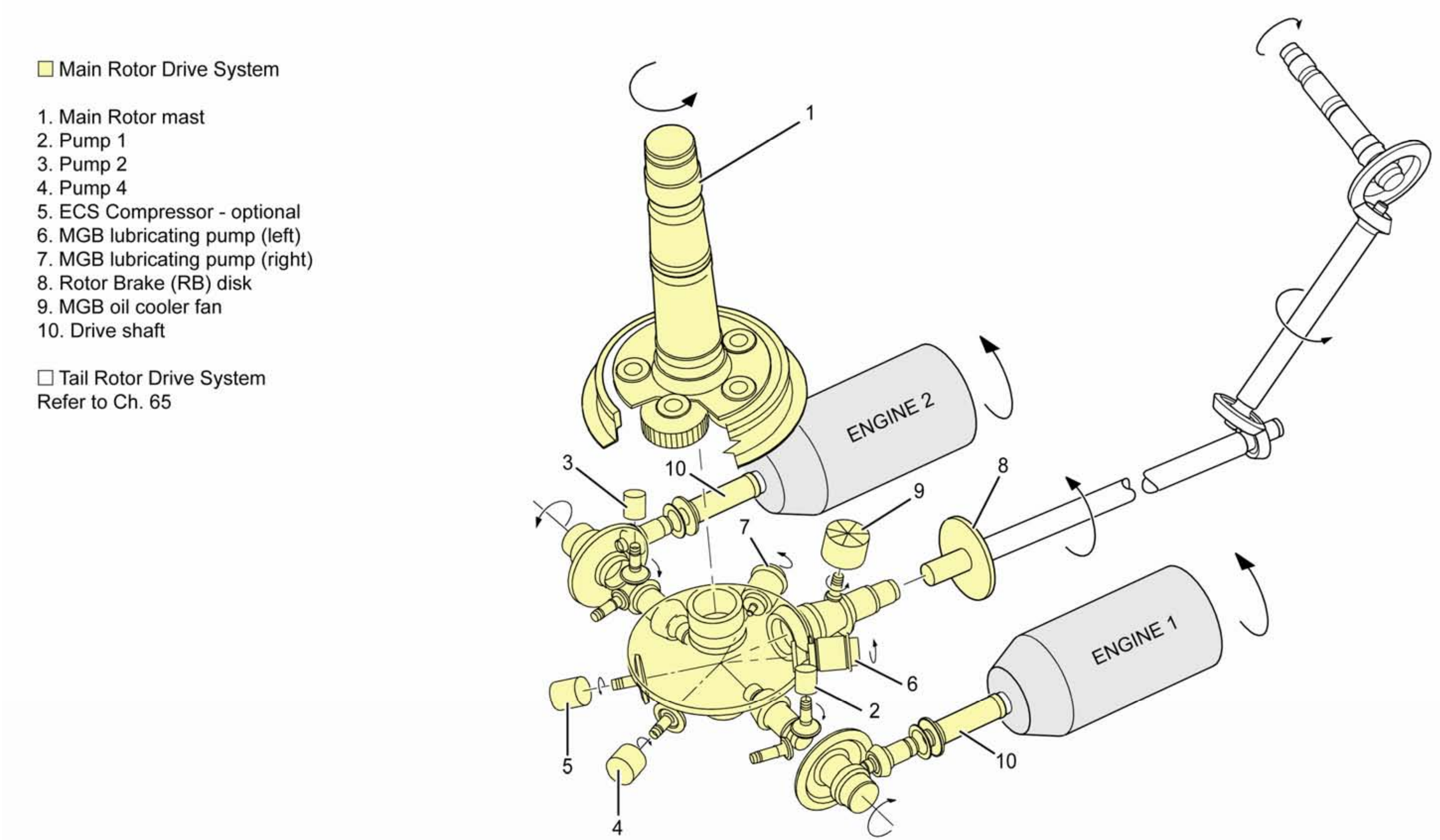
TRANSMISSION - GENERAL

The transmission supplies the drive from the engines to the Main Rotor (MR), the Tail Rotor (TR) and the accessories (pumps, ECS compressor, and alternators - if installed).

The transmission is composed by:

- the Main Rotor Drive System (MRDS)
- the Tail Rotor Drive System (TRDS)

as visualized in the schematic that follows. This chapter describes the Main Rotor Drive System.



TRANSMISSION - GENERAL

MAIN ROTOR DRIVE SYSTEM – GENERAL

The Main Rotor Drive System transmits and decreases the rotational drive from the engines to the Main Rotor. The MRDS comprises:

- the Main Gear Box (MGB);
- two engine gearbox couplings;
- two input modules;
- the engine couplings;
- the freewheels.

The MGB reduce the rotational speed between the input and output drive. The MGB has a self-contained splash lubrication and condition-monitoring oil system.

MRDS – MAIN COMPONENTS

MAIN GEAR BOX

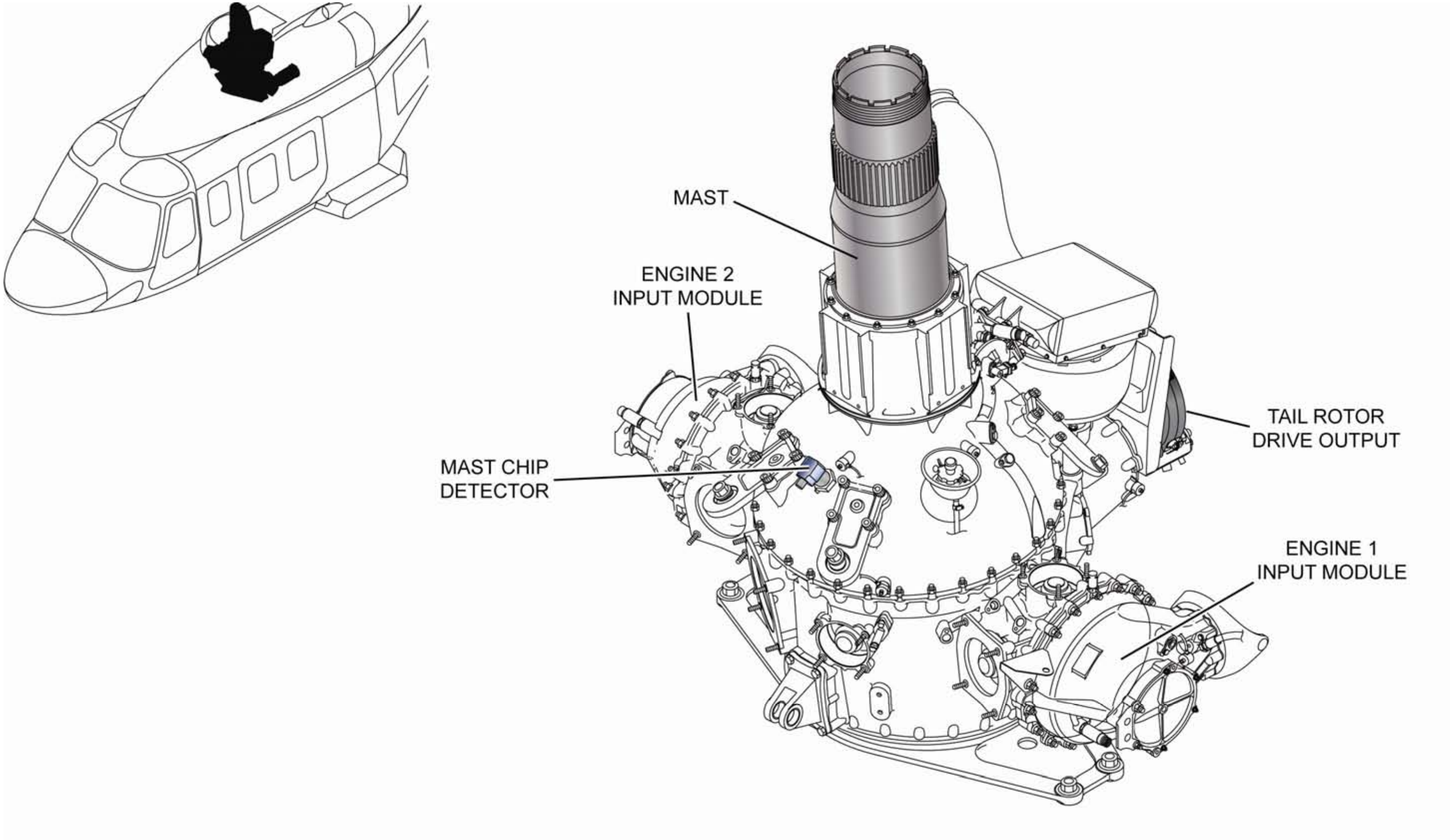
The MGB is the most important component of the MRDS. The MGB is made of aluminium alloy and comprises two (engine) input shaft modules cases, the main case, the upper module case and the bottom case.

The MGB changes the horizontal drive from the engines to a vertical drive of the main rotor (MR) shaft. The MR shaft is secured to the MGB and drives the MR head on which are attached the blades.

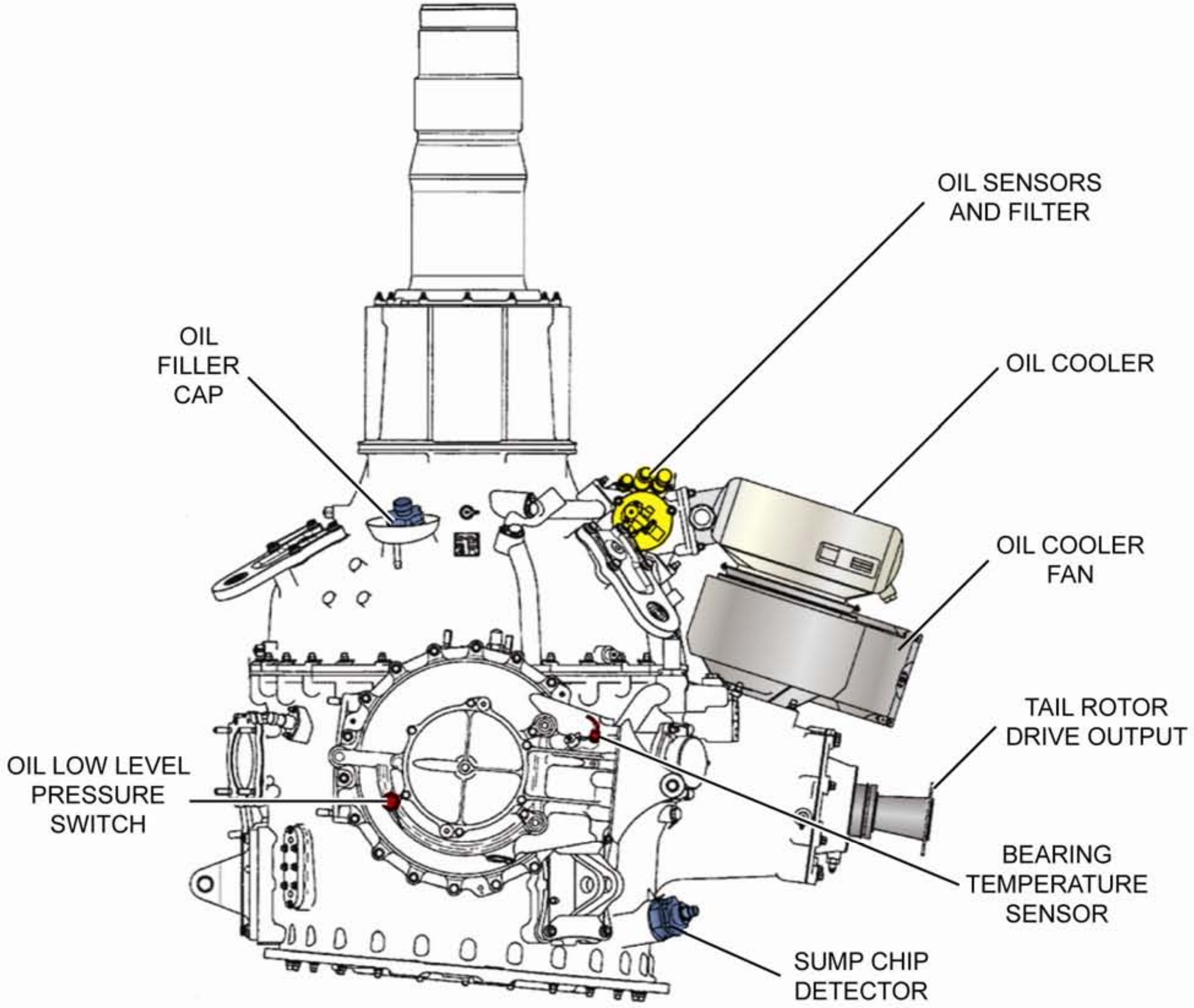
The MGB has three reduction stages that reduce the 21000 rpm of the engines input speed to the 296 rpm of the MR speed (at 100% Nr).

The MGB supplies drive through the Tail Rotor Drive System (TRDS) to the Tail Rotor (TR) and supplies drive at different rotational speeds to the following accessories:

- one MGB oil cooler fan;
- two MGB lubrication pumps;
- three hydraulic power pumps (pump 1, pump 2, pump 4);
- one ECS compressor (optional).



MAIN GEAR BOX



MAIN GEAR BOX – LH SIDE VIEW

MOUNTS AND ATTACHMENTS

Mounts and attachments support and restrain the MGB to the helicopter fuselage. The MGB holds the attachments of

- three MR servo actuators (FW, LH, RH)
- four rods (two FWD and two AFT)
- one anti-torque beam bolted to the bottom of the MGB
- the rotor brake system

The four rods and the anti-torque beam connect the MGB to the helicopter upper deck and allow to transfer torque and dynamical loads from the main rotor mast to the fuselage.

The MGB is tilted 5° forward respect to the helicopter upper deck.

MGB FRONT

MGB REAR

MAIN GEAR BOX MOUNTS AND ATTACHMENTS

ENGINE GEARBOX COUPLINGS

Each one of the two engine gearbox couplings connect the relevant engine to the MGB.
The installation includes these components:

- Torque tube;
- Drive shaft;
- Crosshead;
- Flexible coupling.

The drive shaft is metal tube that transmits the torque of the engine to the MGB. At one end the drive shaft is splined; at the opposite end the drive shaft has a flexible coupling.

The torque tube protects the drive shaft and, together with the crosshead and the flexible coupling, adjusts the incorrect misalignment during engine operations.

INPUT MODULES (LEFT AND RIGHT)

Each one of the two input modules (left and right) realize the connection between the engine shaft and the MGB.

The input module changes the direction of the axis of rotation and acts as first reduction stage for the MGB. Each input module includes a centrifugal freewheeling unit that allows to override the engine in case of failure or engine shutdown so that the rotor can be driven by the remaining operative engine.

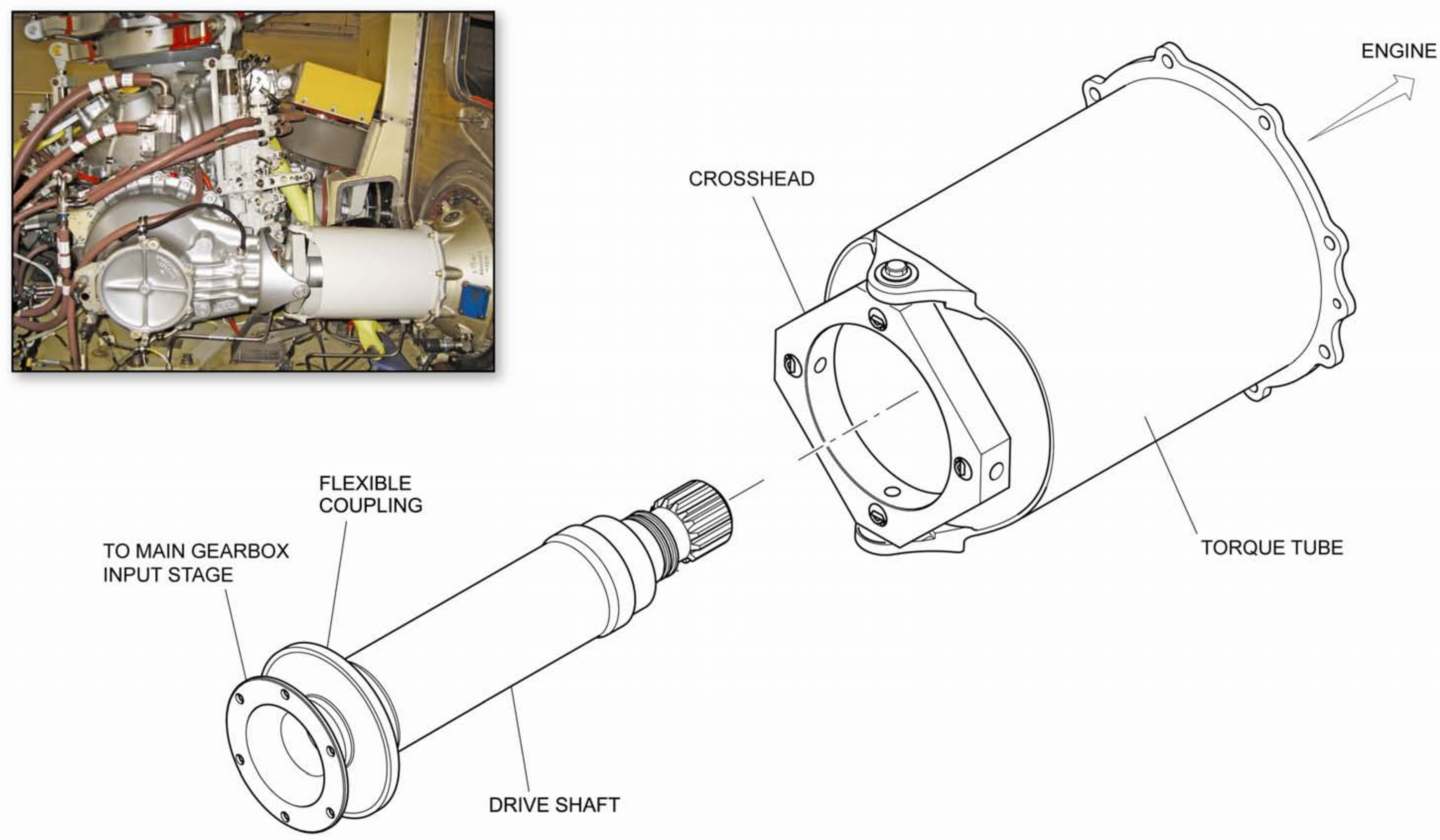
FREEWHEELS

Freewheels enable the movement to be transmitted only in the engine to rotor direction. There are two freewheels, one for each input module. There are two possible conditions:

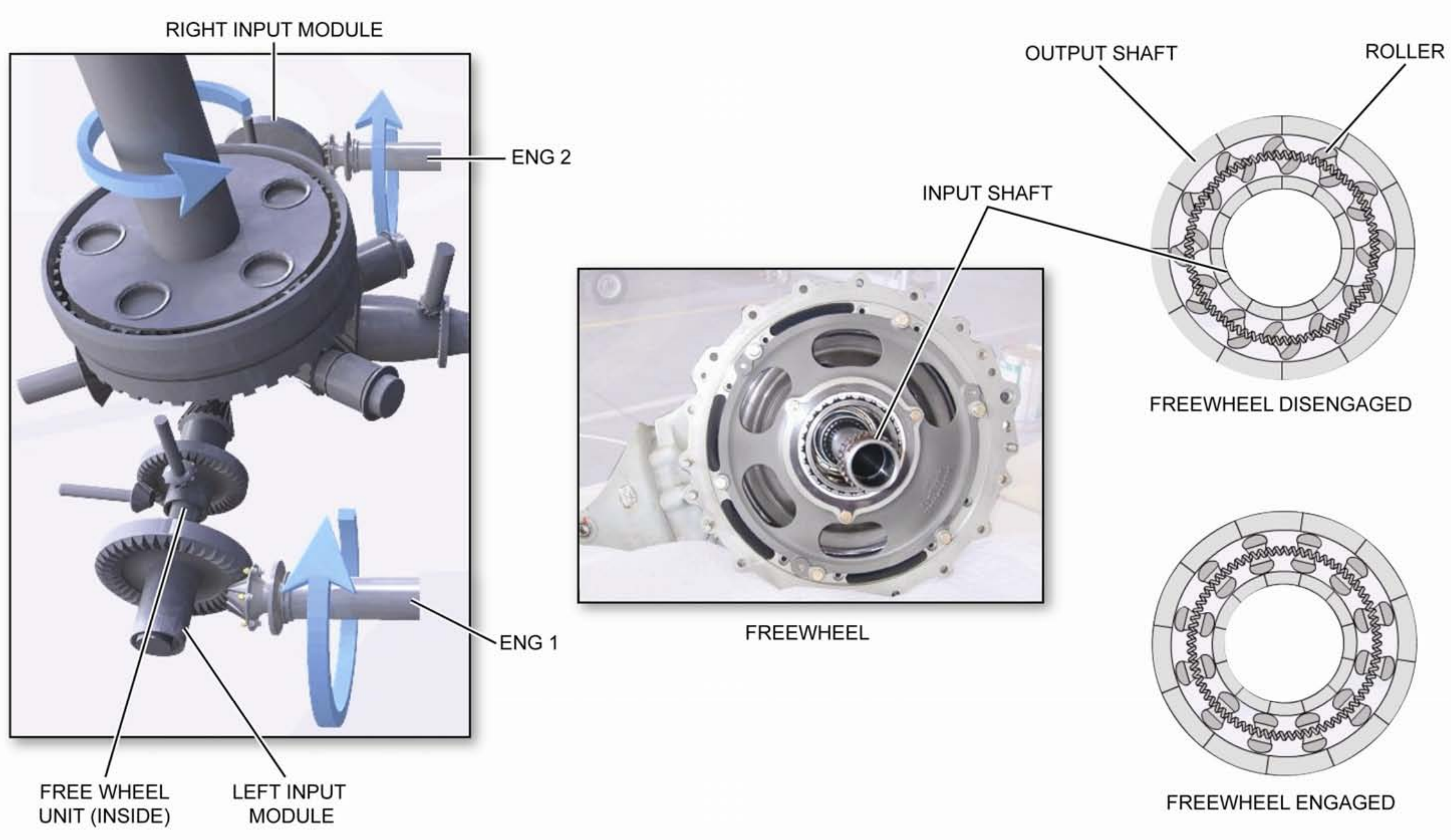
- Freewheel engaged: rollers are caught between the input shaft (engine shaft) and the MGB output shaft that remains driven;
- Freewheel disengaged (autorotation or one engine failed): the output shaft which is driven by the rotor (or by the second engine) frees the rollers: the freewheel acts as a bearing and the (failed) engine is not driven.

When both engines are operating, the LH and the RH freewheels are engaged and the main rotor and tail rotor (and the accessories) are driven.

In autorotation both freewheel are disengaged: the tail rotor and the accessories are driven by the main rotor.



ENGINE GEARBOX COUPLING


RIGHT INPUT MODULE
ENG 2
ENG 1
FREE WHEEL
UNIT (INSIDE)
LEFT INPUT
MODULE
INPUT SHAFT
FREEWHEEL
OUTPUT SHAFT
ROLLER
FREEWHEEL DISENGAGED
FREEWHEEL ENGAGED

FREEWHEELS

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MAIN ROTOR DRIVE – INDICATIONS

The PWR PLANT PAGE displays the values of pressure and temperature in the MGB area.

1. PRESSURE

   The values of pressure in the MGB are represented by digital readouts under the label BAR. Graphically these values are represented on a vertical scale by means of a pointer (triangle) that matches the color of the area on the scale.

   The digital readouts and the pointers are displayed red when the pointer is in the red zone (warning), amber when the pointer is in the amber zone (caution) and green in normal conditions.

2. TEMPERATURE

   The oil temperature values in the MGB are represented by digital readouts under the label °C. Graphically the values are represented on a vertical scale by means of a pointer (T symbol) that matches the color of the area on the scale.

   Green band of the analogue vertical scale represents a normal condition for the hydraulic oil temperature and so the associated digital readout values in Celsius degrees.

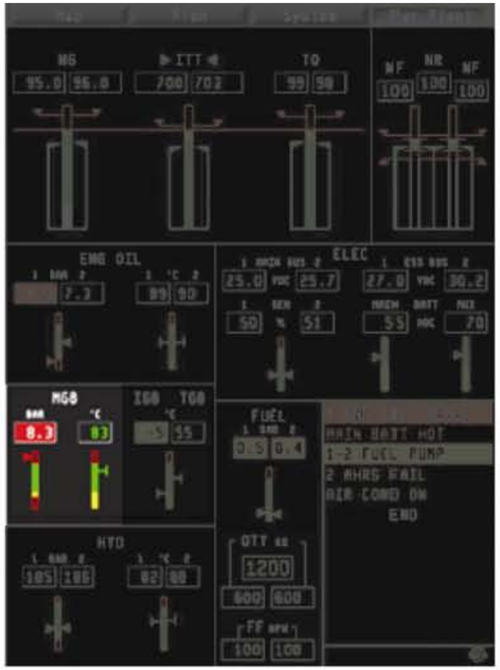
   The amber band represents a caution condition while the red band is associated to a warning condition.

The COMPOSITE FORMAT displays the values of pressure.

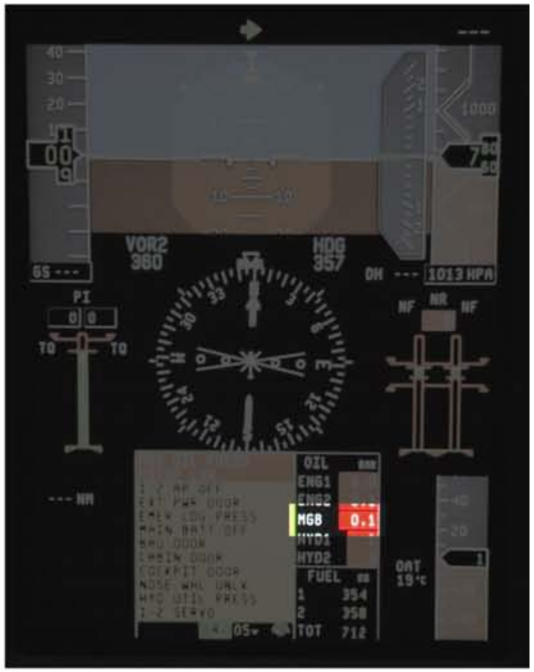
3. PRESSURE

   The values of pressure in the MGB are represented by digital readouts aside the label MGB.

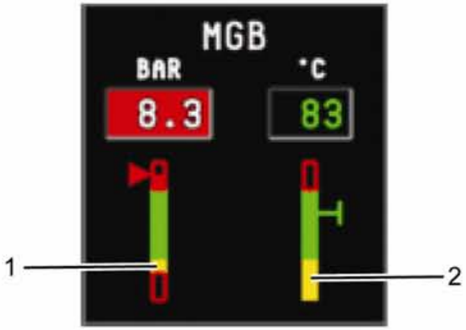
MFD CRUISE PAGE



MFD MAIN PAGE



COMPOSITE FORMAT



MAIN ROTOR DRIVE – INDICATIONS

MAIN ROTOR DRIVE – CONTROLS AND INDICATORS

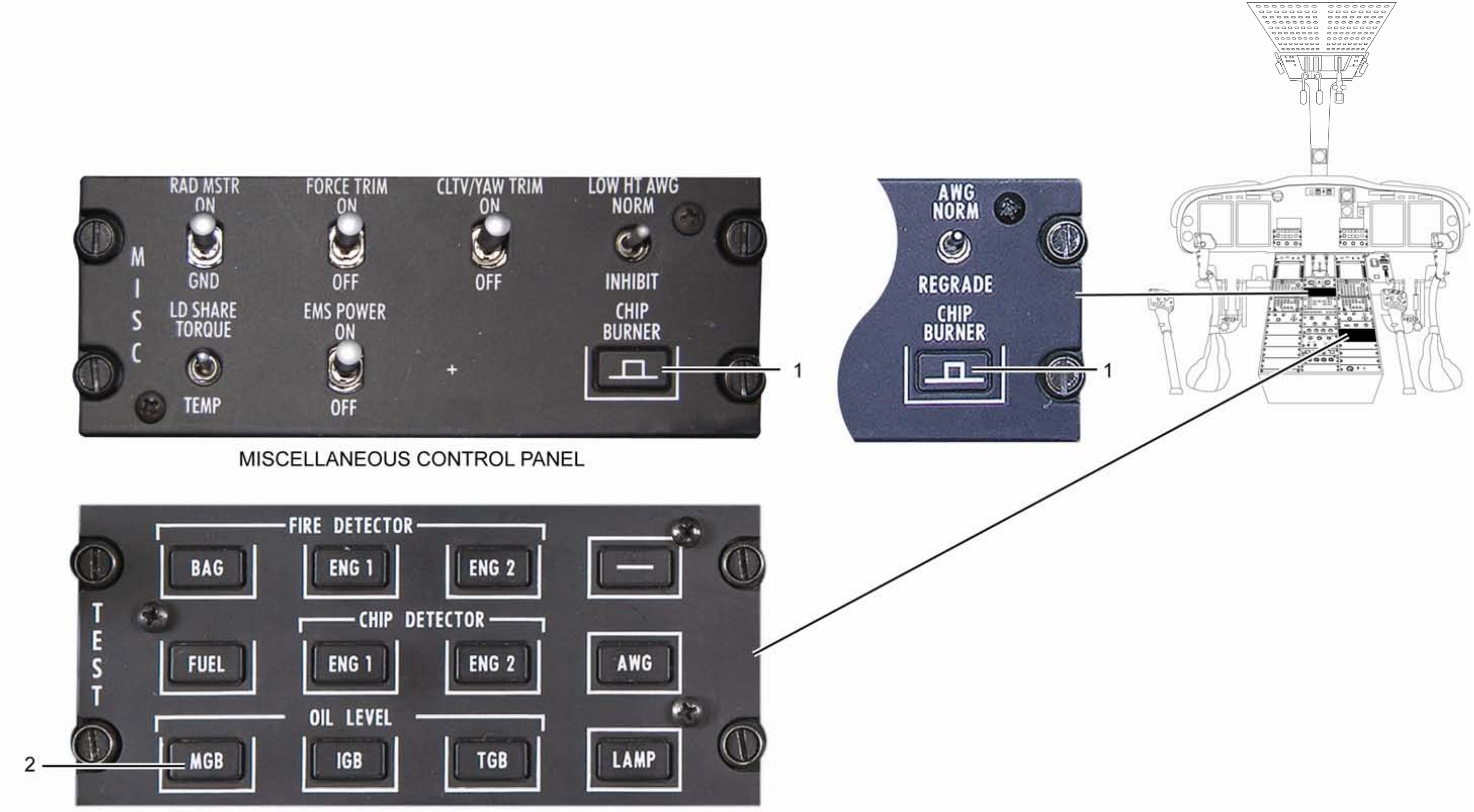
1. CHIP BURNER push-button

   PRESSED ..…….   a chip burning attempt is performed

2. OIL LEVEL MGB push-button (on the TEST panel)

   PRESSED ………   the caution MGB OIL LOW comes on in inverse video in the CAS window

   The test is possible only when the helicopter is on the ground and the NR is less than 2%



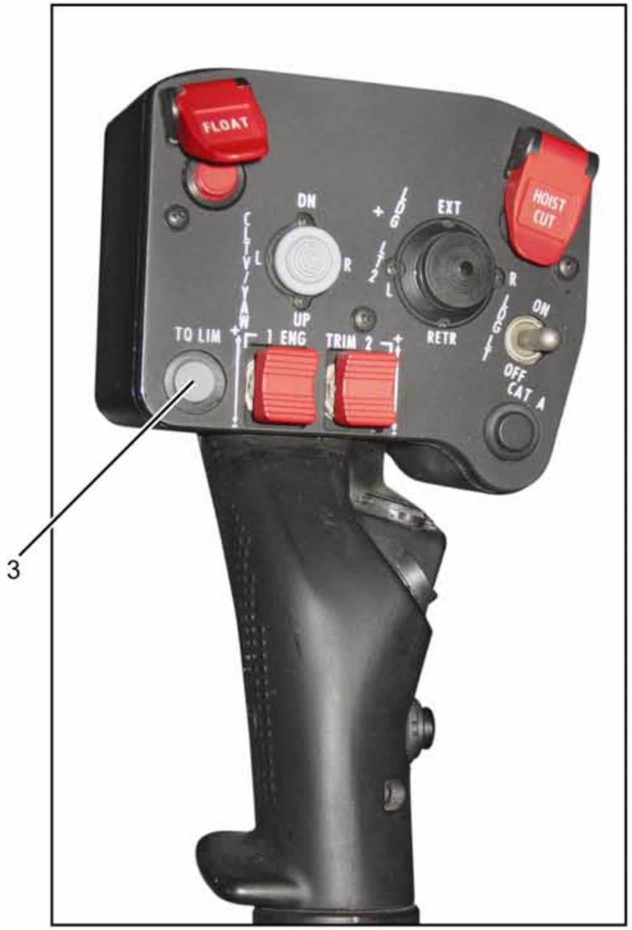
MISCELLANEOUS CONTROL PANEL

TEST CONTROL PANEL

MAIN ROTOR DRIVE – CONTROLS AND INDICATORS

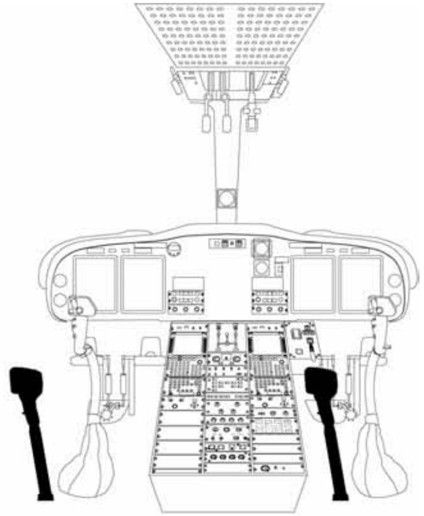
3. TQ LIM toggle switch

PRESSED ..……. engages / disengages the torque limiter



COPILOT

PILOT



MAIN ROTOR DRIVE – CONTROLS AND INDICATORS

TORQUE LIMITER

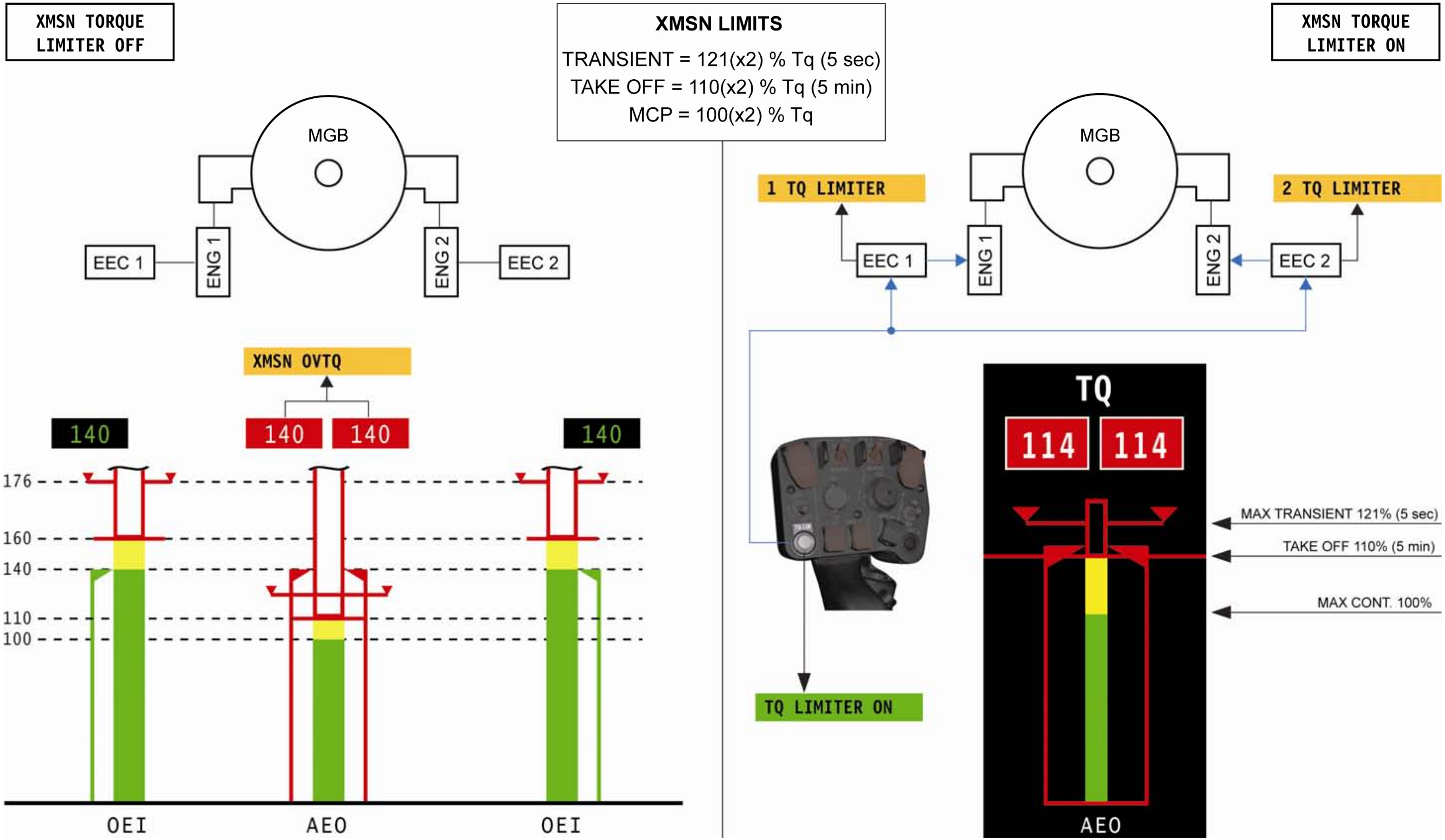
On the AW139 helicopter, the torque limiter function is set to OFF as default.

If the pilot wants to engage the torque limiter function, the TQ LIM push-button on the collective grip, must be pressed. After selection, the green advisory message TQ LIMITER ON will be displayed in the CAS window.

When the torque limiter function is set to ON, the AEO total torque will be limited to be combined torque value of TQ 228%. The OEI engine torque will not be affected.

When pressed a second time, the torque limiter function is deactivated and full power is available from engines.

To perform the OEI TRAINING, the TQ LIM function must be set to ON.



TRANSMISSION TORQUE LIMITS

MGB LUBRICATION

The MGB is lubricated by an oil system where all oil ducts are contained inside the MGB casing to avoid the possibility of oil leaks.

The lubricating system comprises two hydraulic pumps, check valves, a filter, an oil cooler and a fan which sends fresh air through the oil cooler. The lower part of the MGB case acts as a tank for the lubricating oil: in fact the lubrication pumps suck the oil from the sump and deliver it to the distribution system. The distribution of the lubricating oil is achieved by a series of calibrated nozzles (jets).

The outlet pressure of each pump is limited by a pressure relief valves that return the excess of oil to the oil sump. After the pressure relief valves the oil passes through a check valve and then to the oil filter.

The filter includes a by-pass and an impending bypass switch. The filter element will be bypassed if the pressure drop across the filter reaches a preset value.

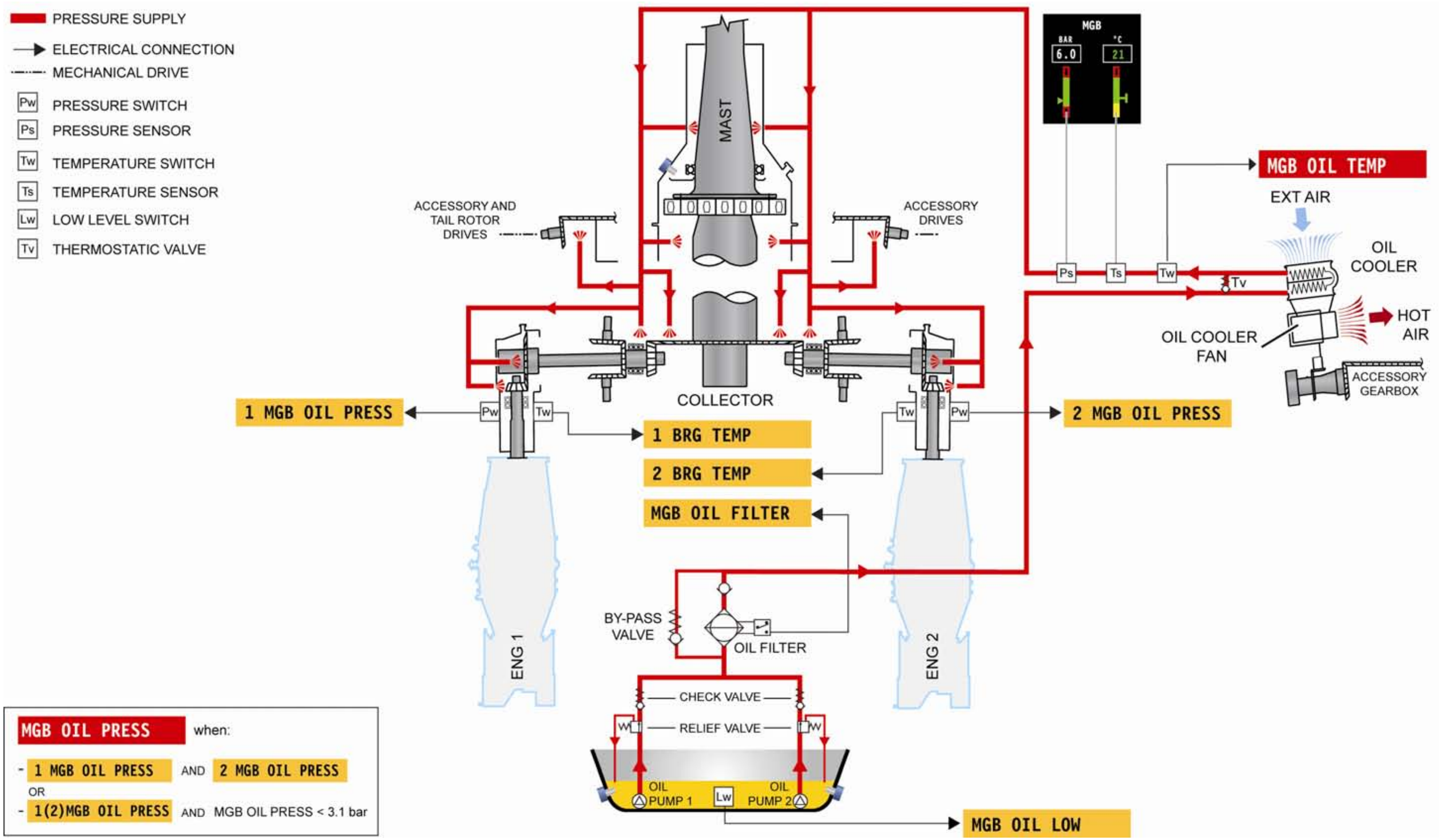
An oil level sight glass is located in front of the MGB.

An oil low level sensor permits to have a caution message related to the oil level condition.

No indications are displayed in case of pump failure.

OIL LOW LEVEL SENSOR

The oil low level sensor is an optical sensor type which gives a signal to MAU 1 and MAU 2 when the oil level in the sump is less the minimum level.



MAIN GEAR BOX LUBRICATING SYSTEM

CHIP DETECTOR SYSTEM

Five chip detectors and one Chip Detector Power Unit constitute the chip detector system.

The purposes of the chip detector system are the followings:

- detect ferrous particles in the lubricating oil
- burn the ferrous particles detected (when possible)

The chip detector system includes:

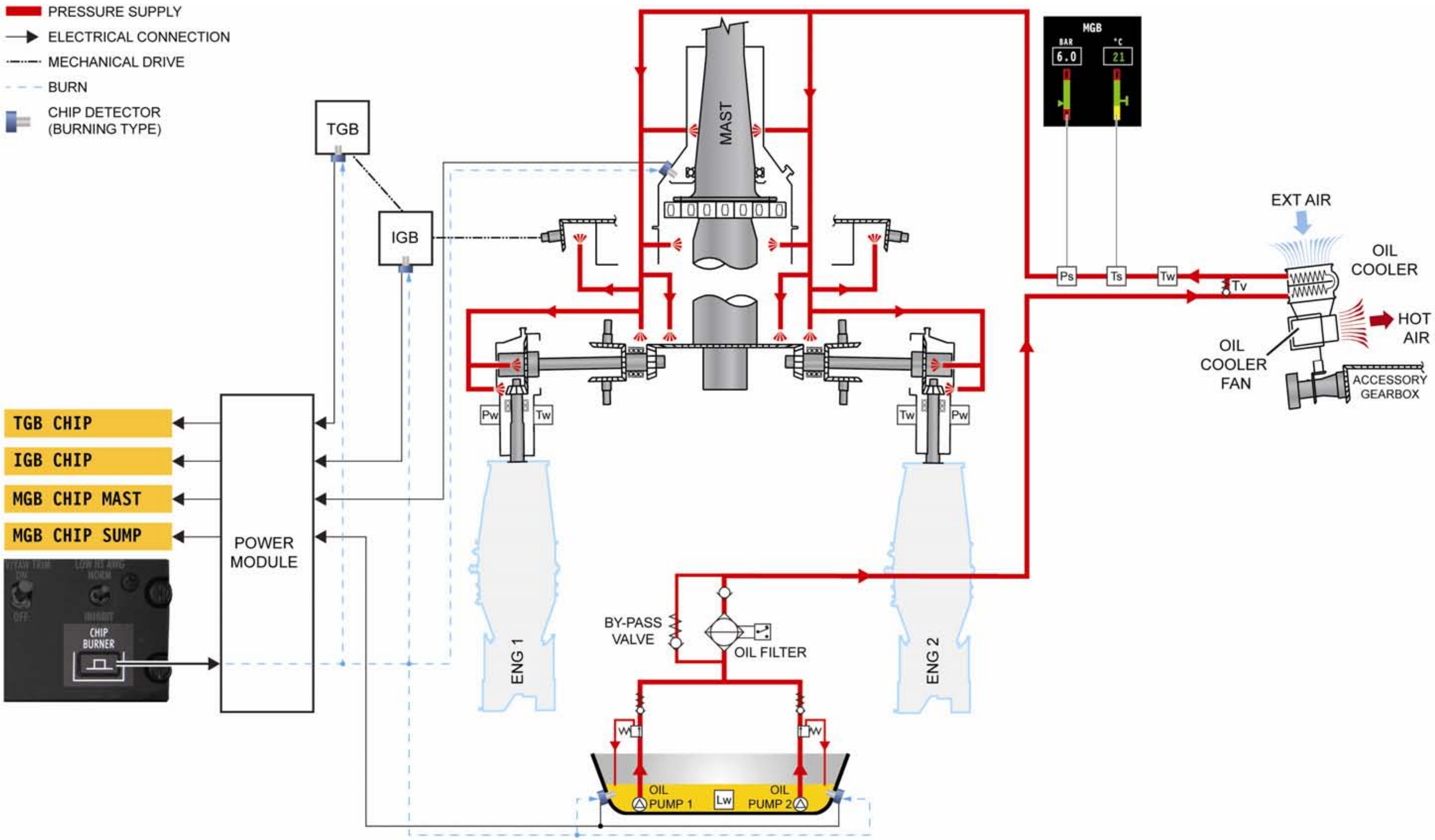
- a power module interfaced with the chip detectors, the two MAUs, the burn command on the MISC control panel;
- five chip detectors: three chip detectors are located on the MGB; one chip detector is located on the Intermediate Gear Box (IGB); and one chip detector is located on the Tail Gear Box (TGB).

When the MGB CHIP MAST or MGB CHIP SUMP caution messages are displayed in the CAS window of the MFD, the pilot can try to burn the ferrous particle(s) pushing the CHIP BURNER switch located on the MISC control panel.

If the burning is successful the particle is classified as “small” and the relevant caution extinguishes.

In the opposite case, the particle is considered “large” and cannot be burned.

When the particle is burned, a message is written in the Non Volatile Memory (NVM) for maintenance purposes.



CHIP DETECTOR SYSTEM

CAS WARNING MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
MGB OIL PRESS	Oil pressure below limits at one or both engine MGB inputs and in MGB oil system. (less than 3.1 bar) (resets above 3.4 bar) *– voice warning: WARNING-WARNING repeated once*	OIL PRESSURE LOW	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES MAIN GEARBOX
MGB OIL TEMP	MGB oil temperature above limit (greater than 109°C) (resets below 97°C) *– voice warning: WARNING-WARNING repeated once*	OIL TEMPERATURE HIGH	

CAS CAUTION MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
XMSN OVTQ	Transmission TQ limit exceeded, Take Off rating 5 min limit (either engine above 110%) or 2.5 min OEI limit (160%)	MAIN GEARBOX OVERTORQUE	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES DRIVE SYSTEM
MGB CHIP MAST MGB CHIP SUMP	Activated CHIP burner (on MISC control panel) It is permitted to activate the CHIP BURNER up to 3 times to clear a chip in one flight. On the 4th CHIP caution Land As Soon As Practicable	MAIN GEARBOX CHIP DETECTOR	
MGB OIL FILTER	MGB oil filter blockage and in bypass (differential pressure over filter exceeds 1.25 bar)	MAIN GEARBOX OIL FILTER	
MGB OIL LOW	MGB oil level low (caution only active with engines shut down and NR below 2%)	MAIN GEARBOX OIL LOW	
1(2) MGB OIL PRESS	oil pressure at MGB input 1 or 2 low (less than 3.1 bar) possible blockage in oil duct to engine NOTE If both 1 MGB OIL PRESS and 2 MGB OIL PRESS are displayed, then the red warning MGB OIL PRESS will come on and suppresses the two previous cautions	MAIN GEARBOX INPUT OIL PRESSURE	

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
CHIP DET UNIT **CHIP DET TEST**	Chip detect unit malfunction	GEARBOX CHIP DETECT UNIT MALFUNCTION	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES DRIVE SYSTEM
CHIP MAST FAIL **CHIP SUMP FAIL**	Associated gearbox chip sensor failed	GEARBOX CHIP DETECTOR SENSOR FAILURE	
1(2) BRG TEMP	Associated MGB engine input bearing over temperature	MAIN GEARBOX INPUT BEARING TEMPERATURE	
1(2) TQ LIMITER	Associated engine torque limiter system not functioning	TORQUE LIMITER	

CAS ADVISORY MESSAGES

CAS CAPTION	MESSAGE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
TQ LIMITER ON	Torque limiter activated		Supplement 2 NORMAL PROCEDURES

TRANSMISSION FAILURES

The most common transmission failures are

- lubrication system failure (oil pump, ducts, nozzles etc)
- transmission component failure (gears, bearings, etc)
- accessory component failure (hydraulic pumps, electrical generators, coolers, etc)

The transmission is monitored by oil pressure and oil temperature sensors and switches, chip detectors and CAS messages. It is probable that one or more of these indications will be present if a mechanical transmission failure is imminent. However, whether these indications are present or not, crew sensory perceptions such as

- abnormal mechanical noise and/or
- heavy vibration levels and/or
- odour of hot metal fumes
- play an important part in diagnosis of impending transmission system failures and assist the pilot in determining what actions are required

In general

a single failure dictates	LAND AS SOON AS PRACTICABLE
a double failure dictates	LAND AS SOON AS POSSIBLE
a multiple failure dictates (including abnormal noise and/or vibration)	LAND IMMEDIATELY

MAIN ROTOR DRIVE - LIMITATIONS

Refer to AW139-RFM-4D Section 1.

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CHAPTER
63
MAIN ROTOR DRIVE

SECTION 50 – ROTOR BRAKE

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ROTOR BRAKE – GENERAL

The Rotor Brake (RB) is described in SECTION 5 – OPTIONAL EQUIPMENT SUPPLEMENTS of the AW139–RFM–4D.

The Rotor Brake allows:

- a rapid deceleration of the rotors after engines shut down
- to maintain the rotors stopped up to 8 hours
- the manual restore of the parking pressure

ROTOR BARKE – MAIN COMPONENTS

ROTOR BRAKE CONTROL LEVER (RBCL)

The pilot manually operates the Rotor Brake Control Lever for the selection of one of the following positions:

- OFF (RB release)
- PUMPING from forward detent and backward stop (pumping limit)
- BRAKE

The RBCL transmits mechanically the control to the Rotor Brake Control Module (RBCM). The trigger with the spring lets the pilots to unlock the lever and move it between the OFF and BRAKE position.

ROTOR BRAKE CONTROL MODULE (RBCM)

The Rotor Brake Control Module (RBCM) transforms the action of the pilot in hydraulic power of the RB circuit. The RBCM includes:

- a dual-stage hydraulic pump that supplies the necessary pressure and fluid displacement;
- an outlet valve which discharges the pressure in the hydraulic reservoir;
- an hydraulic accumulator which provides the hydraulic fluid displacement during the brake application;
- a shut-off valve which manages the RB circuit;
- an interlock pressure switch which detects the pressure in the circuit;
- a pressure transducer which provides monitoring of pressure values and gives a visual indication on the RBPI.

ROTOR BRAKE PRESSURE INDICATOR (RBPI)

The Rotor Brake Pressure Indicator (RBPI) is installed inside the cockpit and gives:

- rotor brake pressure indications (on the display),
- the position of the caliper.

ROTOR BRAKE ASSEMBLY (RBA)

The Rotor Brake Assembly (RBA) comprises the caliper, the pads and the pistons.

The RBA provides for the braking action when pistons are hydraulically actuated to clamp the rotating disc bolted to the tail output drive shaft flange.

ROTOR BRAKE ACTUATION ASSEMBLY (RBAA)

The Rotor Brake Actuation Assembly (RBAA) provides for to move and to maintain the RBA caliper on its operative UP and DOWN positions.

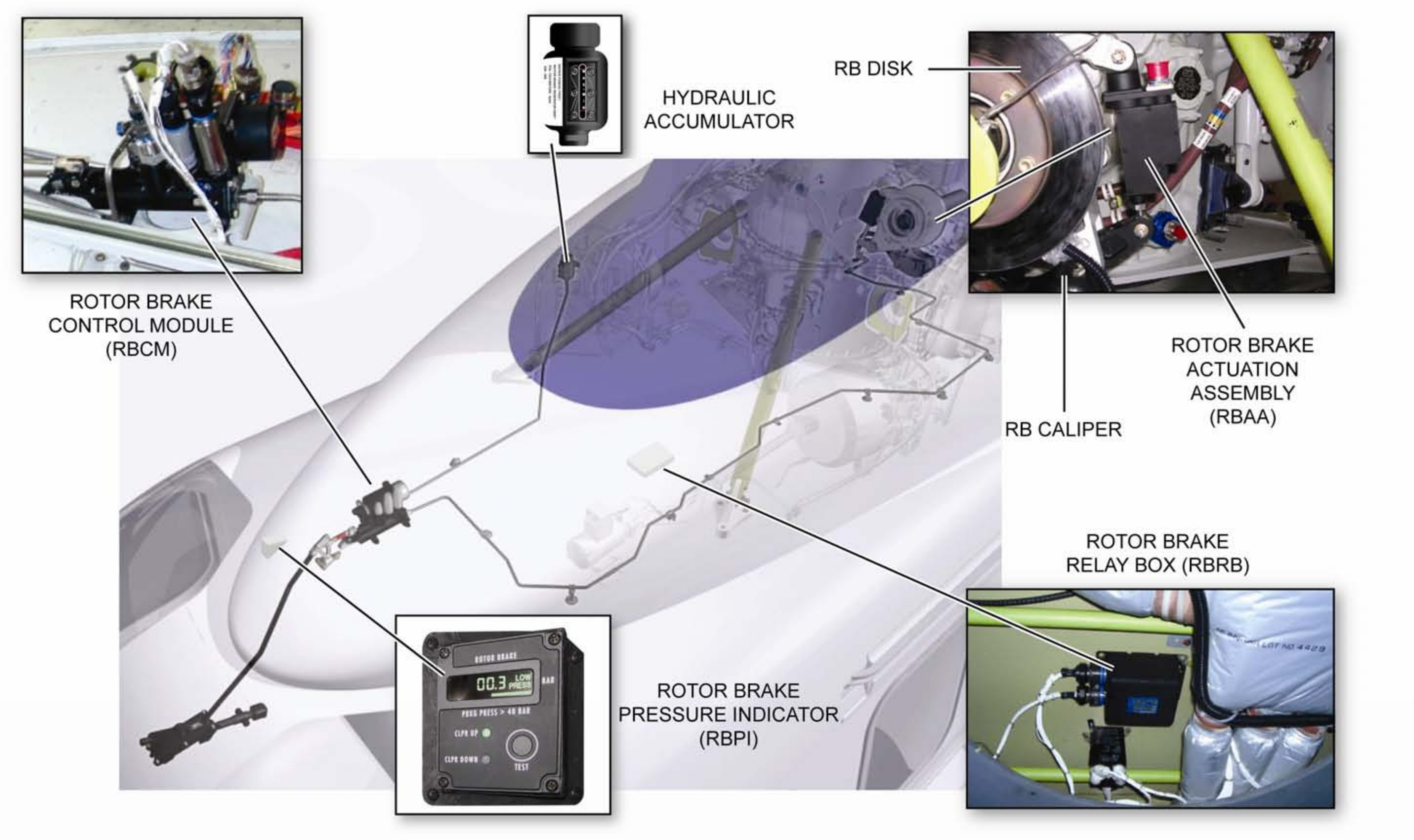
ROTOR BRAKE RELAY BOX (RBRB)

The Rotor Brake Relay Box (RBRB) controls the UP and DOWN movements of the caliper according to this logic:

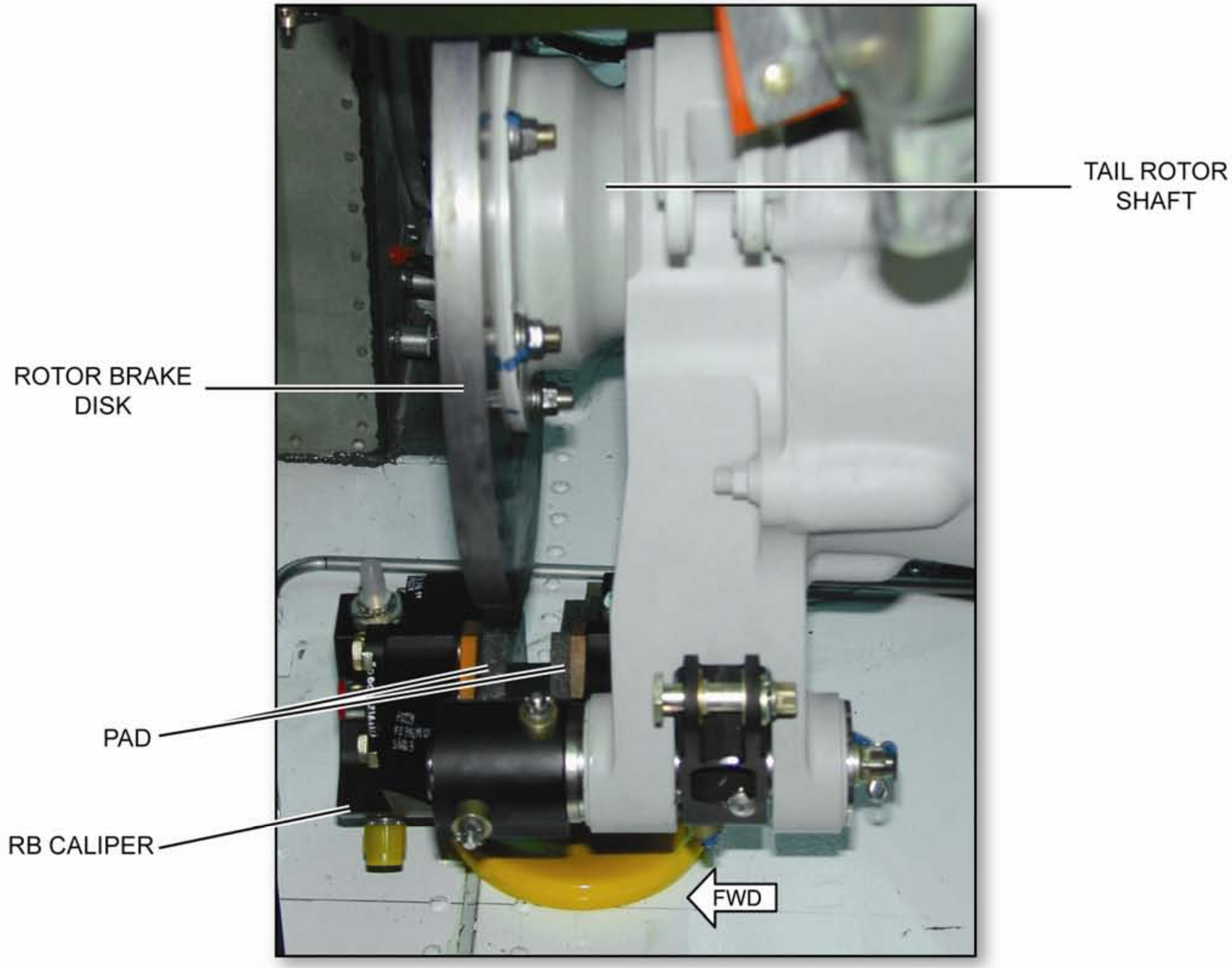
- during flight the caliper is maintained in DOWN position far from the rotor brake disc;
- on the ground with the engines OFF the caliper is moved in UP position.

The electrical signals used for the braking action are provided by:

- the Rotor Brake Control Lever position;
- the Rotor Brake Actuation Assembly UP/DOWN position;
- the engine mode selector position;
- the Weight–On–Wheels (WOW) microswitch.



ROTOR BRAKE – MAIN COMPONENTS (1 OF 2)



ROTOR BRAKE – MAIN COMPONENTS (2 OF 2)

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ROTOR BRAKE – CONTROLS AND INDICATORS

1. ROTOR BRAKE CONTROL LEVER (RBCL) at the right side of the overhead panel

OFF…………………............ the rotor brake is released

BRAKE ……………………. - NORMAL BRAKING

Allows to pressurize the system at 26-28 bar for dynamic braking by filling the accumulator

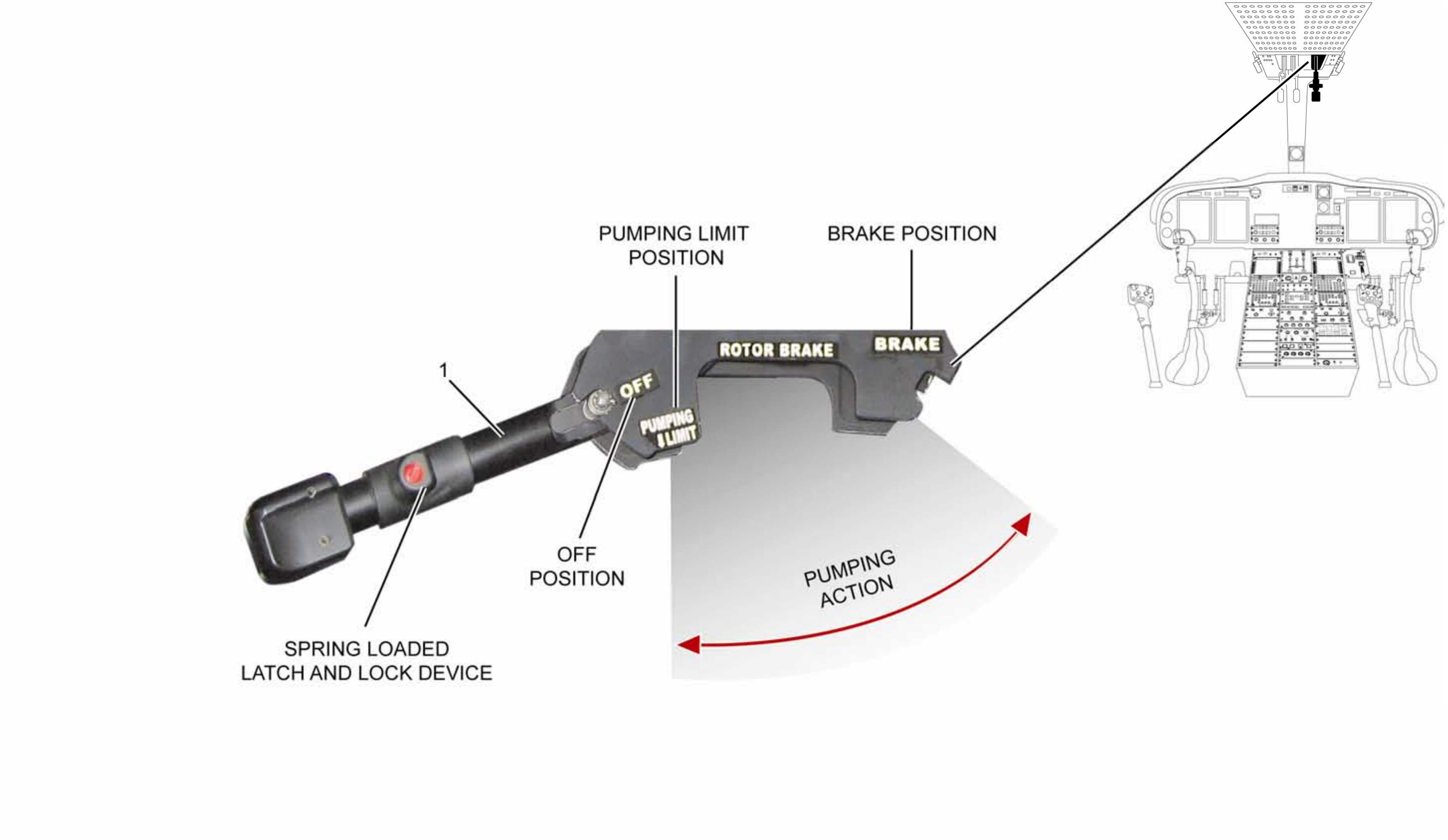
- PARKING

to perform a pumping action by moving the lever between PUMPING and BRAKING positions, increases the pressure in the circuit and in the accumulator up to a maximum parking pressure of 46.5 bar

PUMPING action ..……… (forward detent and backward step) permit to increase the pressure inside the system

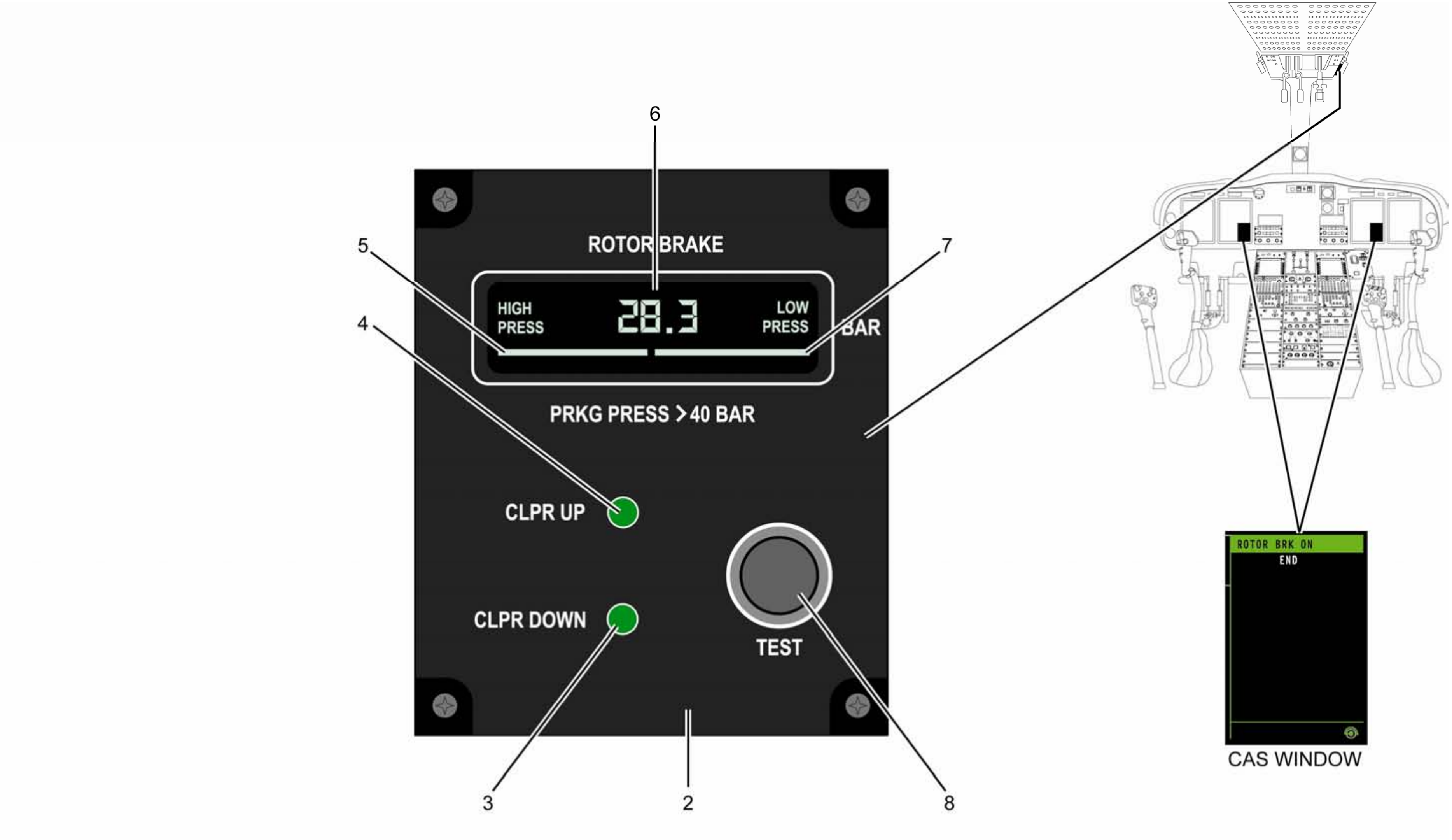
NOTE.

The minimum parking brake pressure is 26 bar and may be achieved after 8 hrs. Below this pressure is possible to restore the correct one in the circuit, pumping again



ROTOR BRAKE – CONTROLS AND INDICATORS (1 OF 2)

2. ROTOR BRAKE PRESSURE INDICATOR (RBPI) at the right side of the overhead panel
3. CLPR DOWN .... green led. Lit when the caliper is down; flashes during the transition
4. CLPR UP green led ...... green led. Lit when the caliper is up; flashes during the transition
5. HIGH PRESS .... red underlined. Lit when the pressure is equal or above 50 bar
6. LOW PRESS .... green underlined. Lit when the pressure is equal or less than 20 bar
7. PRESS digital readout .. displays the operating pressure in green
8. TEST pushbutton .... when pressed the RBPI test light is performed



ROTOR BRAKE – CONTROLS AND INDICATORS (2 OF 2)

ROTOR BRAKE – OPERATIONS

The braking action is manually controlled by the Rotor Brake Control Lever (RBCL). The movement of the lever allows to generate the required hydraulic pressure, using a mechanical pump that is installed inside the Rotor Brake Control Module (RBCM). The RBCL and the pump are connected via a control rod. The stroke between PUMPING and BRAKE positions, produce the pressure for:

NORMAL BRAKING

- When the pilot moves the RBCL to BRAKE, the RBCM supplies the hydraulic pressure needed to stop the rotor disk (≈ 26-28 bar). The rotors stop in 11 ÷ 15 sec approx. (DYNAMIC BRAKING).

PARKING

- To perform a "pumping action" the RBCM permit to reach a value of 46 bar required to maintain the rotors stopped for approximately 8 hours.

When the lever is set to OFF, the outlet valve (mechanically connected to the lever), goes in open position and permit to de-pressurize the circuit.
For safety reason, the RBA can move in the up and down position. This feature, permit to maintain the Rotor Brake Assembly (RBA) away from the RB disk when all the engines are running and the helicopter is in flight. The RBA is moved by an electrical motor, called Rotor Brake Actuation Assembly (RBAA).
This function is managed by the Rotor Brake Relay Box (RBR) that uses the following signals:

- RBCL / RBAA UP/DOWN positions;
- Engine Control Panel selectors positions;
- WOW (Weight-On-Wheels).

The RBA moves in UP position only when the helicopter is on the ground and the engines mode selector in set to OFF.
The electrical shut-off valve prevents inadvertent pressurization of the RBA (valve open) when:

ON GROUND

- all engines are OFF and WOW ON and hydraulic circuits are pressurized (pressure detected by the Interlock Pressure Switch)

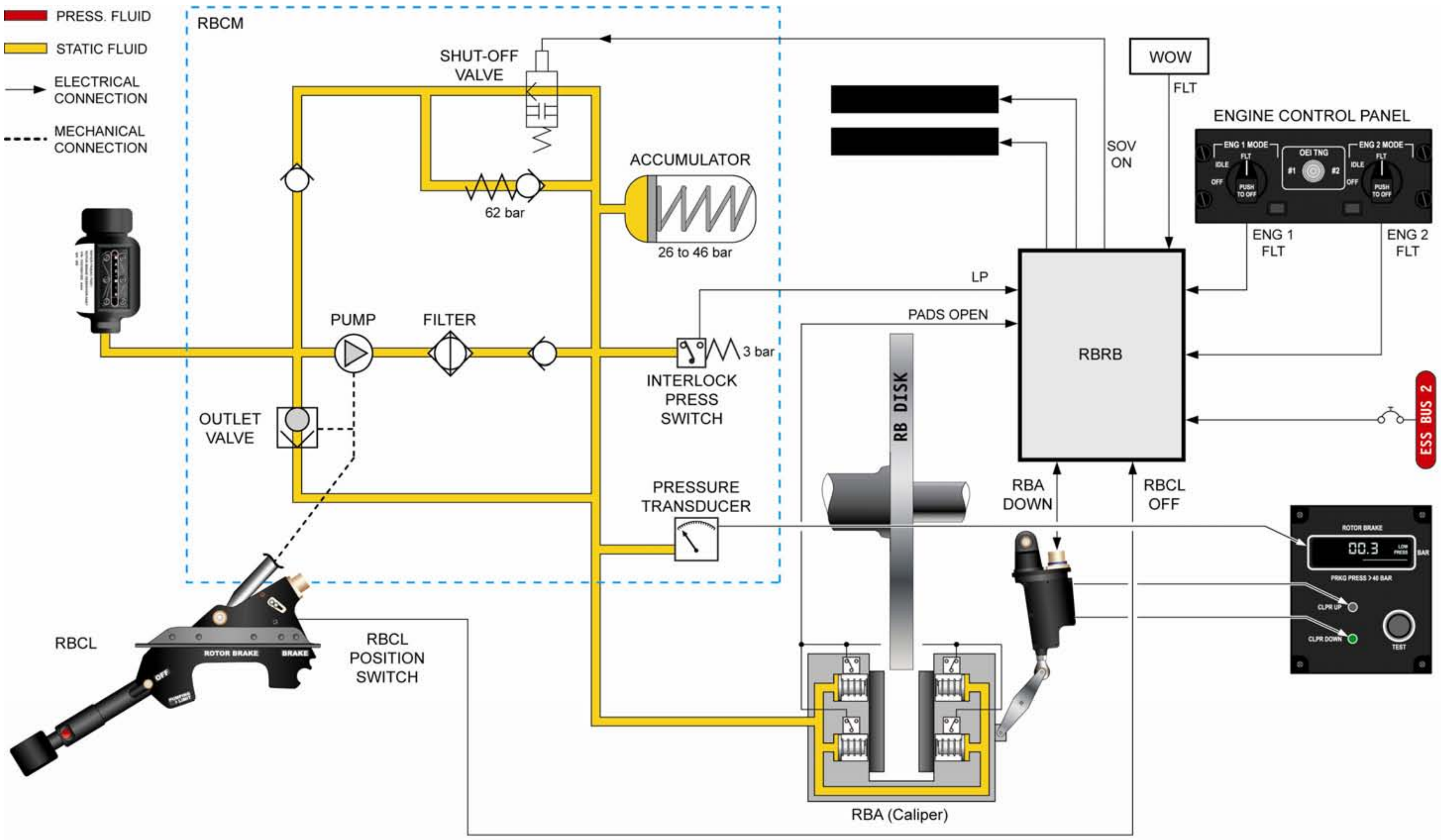
IN FLIGHT

- all engines ON and WOW OFF.

In case the system is already pressurized and the "ENGINE START-UP PROCEDURE" begins, the shut-off valve automatically opens and de-pressurizes the hydraulic circuit.

IN FLIGHT OPERATIONS

- the RBCL is in OFF position
- no hydraulic pressure is furnished to RB circuit
- the caliper is DOWN
- the CLPR DOWN green LED on the Rotor Brake Pressure Indicator (RBPI) is lighted



IN-FLIGHT OPERATIONS

ON GROUND OPERATIONS

WITH ONE OR BOTH ENGINES IN FLT OR GI

- the RBCL is in OFF position
- no hydraulic pressure is supplied to rotor brake circuit
- the caliper is DOWN
- the CLPR DOWN green LED on the RBPI is lighted

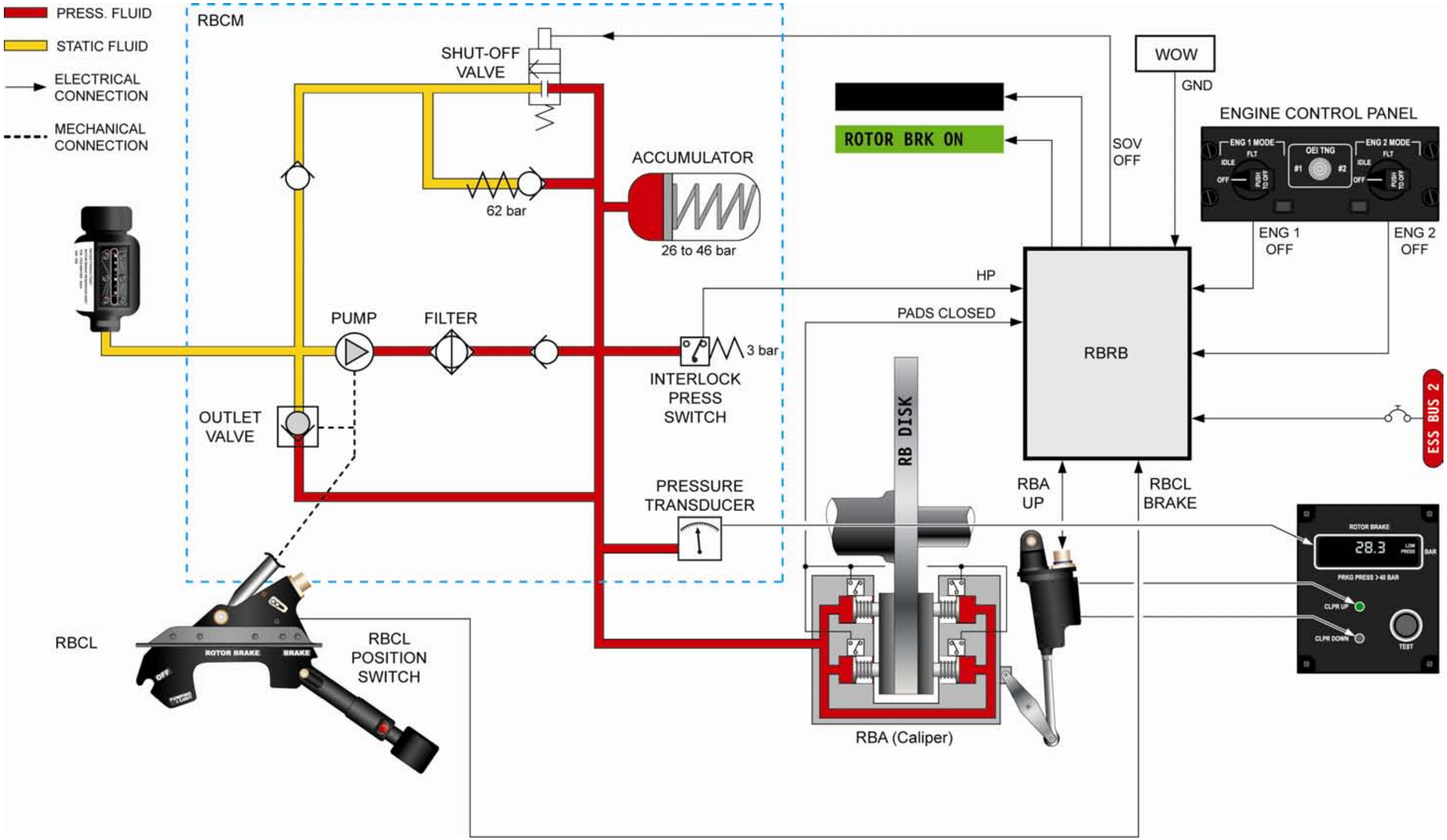
ON GROUND AFTER ENGINES SHUT-DOWN

- the RBCL is in OFF position
- no hydraulic pressure is furnished to RB circuit
- the caliper rises UP automatically
- the RBPI changes the status of the indicators as follows:
  - the CLPR UP and CLPR DOWN flash during transition
  - the CLPR UP lights on
  - LOW PRESS lights on

BRAKE ACTION

When the engines stop the displacement of the RBCL from OFF to BRAKE positions causes the braking action and

- the RB circuit is pressurized at 26-28 bar
- LOW PRESS on the RBPI goes off
- the digital display of the RBPI shows the braking pressure
- the message ROTOR BRAKE ON is displayed on the CAS window



ON GROUND OPERATIONS

PARKING OPERATIONS

Using pumping action of the RBCL sets the system to PARKING condition.

When the pressure becomes grater than 40 bar but less than 62 bar:

- the RBPI digital display shows the braking pressure;
- ROTOR BRAKE ON is displayed on the CAS window;
- up to 8 hours of parking pressure are guaranteed;

When the pressure decreases below 20 bar:

- LOW PRESS is displayed on the RBPI;
- a pumping action is required to re-pressurize the circuit until HIGH PRESS is displayed on the RBPI.

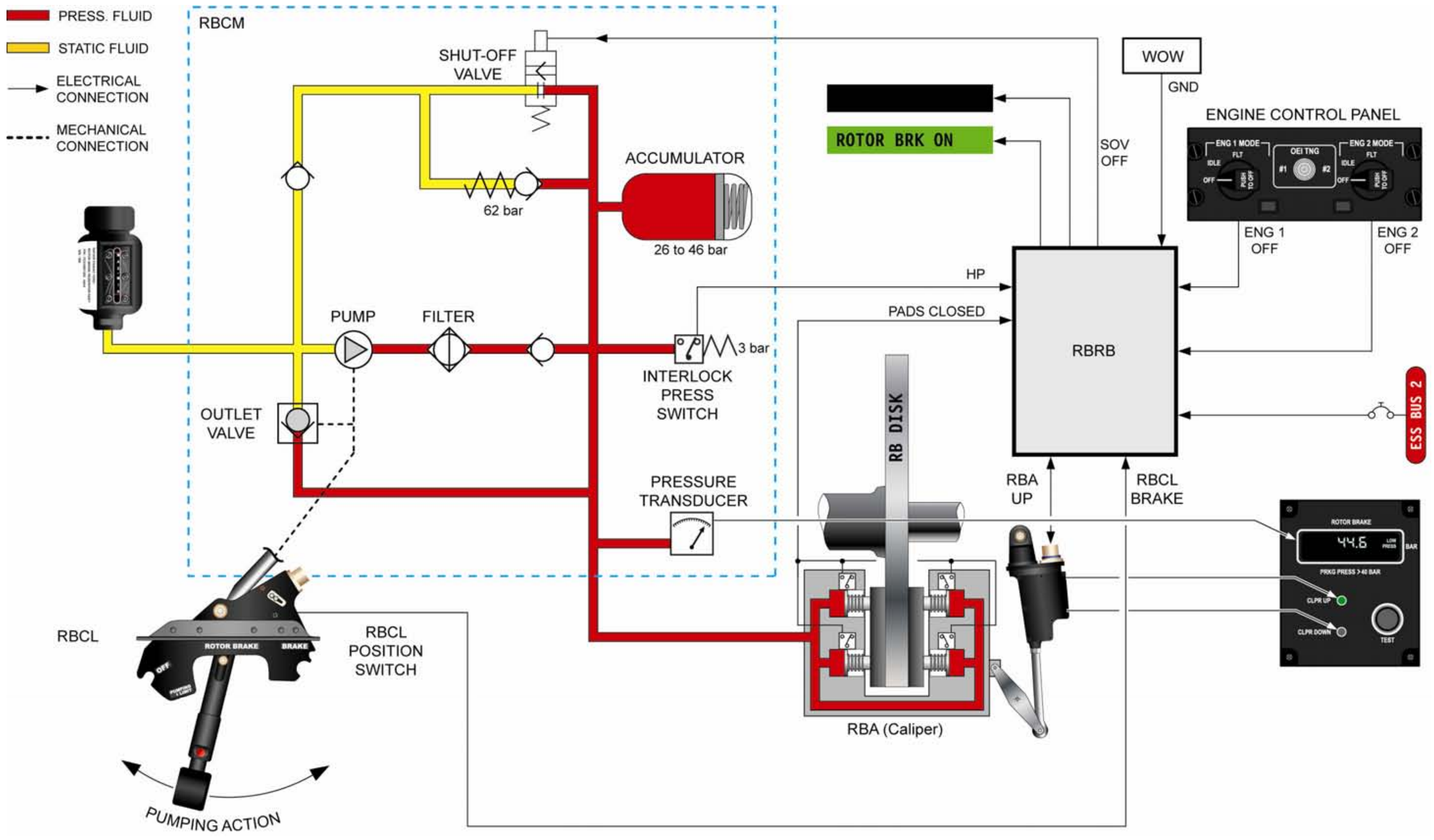
BRAKE RELEASE

To release the brake the pilot must set the RBCL to OFF:

- the CLPR UP green LED on the RBPI remains lighted;
- LOW PRESS is displayed on the RBPI.

AFTER ENGINE STATING:

- the RBCL is in OFF position;
- no hydraulic pressure is supplied to rotor brake circuit;
- the caliper moves automatically DOWN;
- on the RBPI
    - o the CLPR DOWN and CLPR UP flash during transition;
    - o CLPR DOWN green LED lights on.

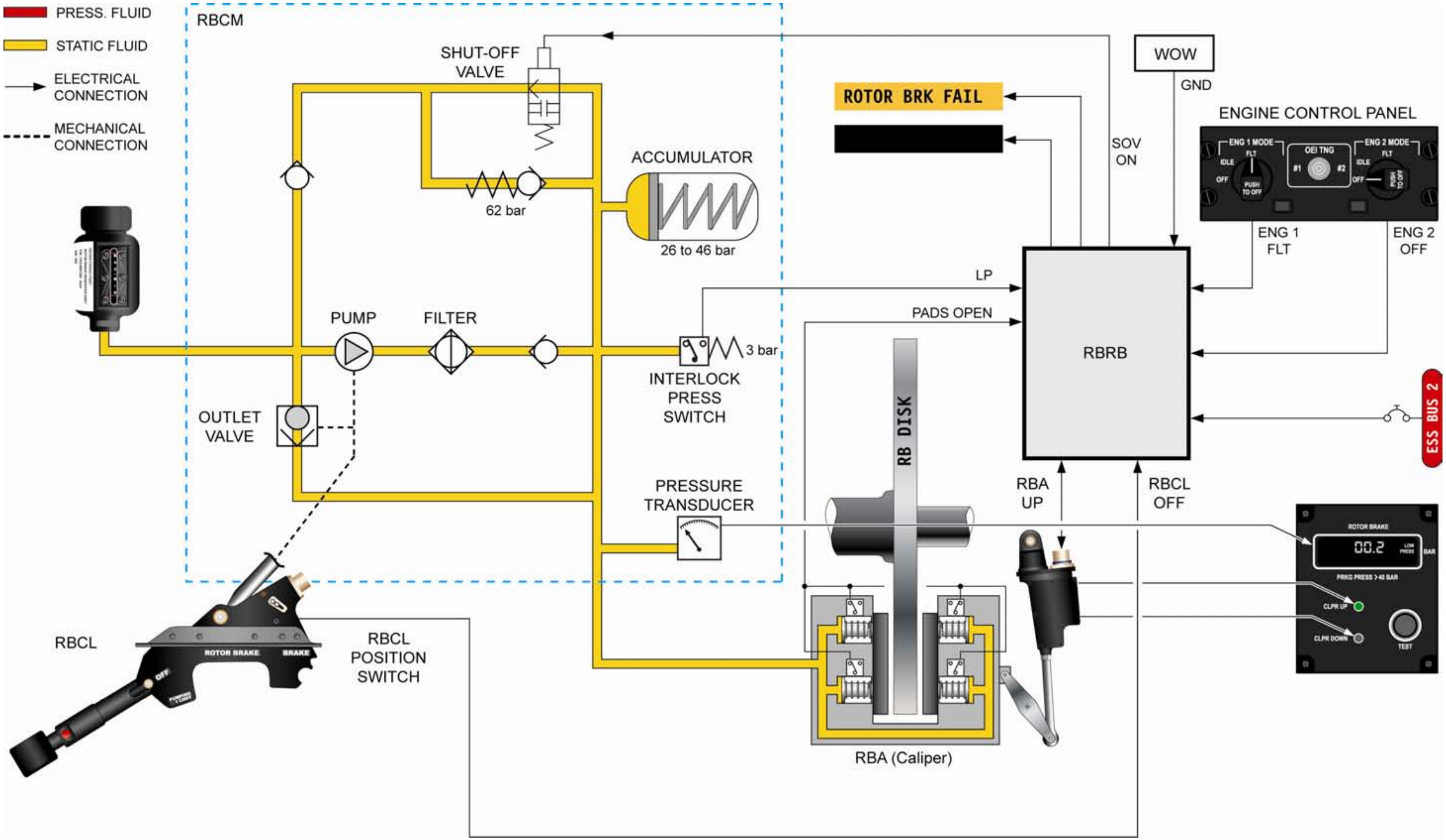


PARKING OPERATIONS

ROTOR BRAKE FAILURES

The ROTOR BRK FAIL caution message is displayed in the CAS window when one of the following condition occurs:

1. The pads are engaged and there is no pressure in the system;
2. When, with both engines OFF, after 10 sec from the shut-down of the last engine, the caliper is not in UP position. The caliper may be in DOWN position or in any position between DOWN and UP);
3. When, with one or both engines in GI/FLT condition after 10 sec the caliper is not in DOWN position. The caliper may be in UP position or in any position between UP and DOWN);
4. When one or both engines are started in GI/FLT condition with the RBCL not in OFF position;
5. When the system is not pressurized but the RBCL is not in OFF position.



ROTOR BRAKE FAILURE (e.g. caliper up with engines not OFF)

CAS CAUTION MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
ROTOR BRK FAIL	Rotor brake system in one of the failure conditions noted • Brake pads not withdrawn • With both engines OFF brake caliper not UP • With an engine in FLT, brake caliper not DOWN • With an engine in GI, brake caliper not DOWN • Rotor brake lever not in OFF, one engine to FLT or in GI (displayed on ground only)	ROTOR BRAKE FAIL	Supplement 1 Section 3 EMERGENCY AND MALFUNCTION PROCEDURES ROTOR BRAKE

ROTOR BRAKE ADVISORY MESSAGES ON ROTOR BRAKE MONITOR PANEL

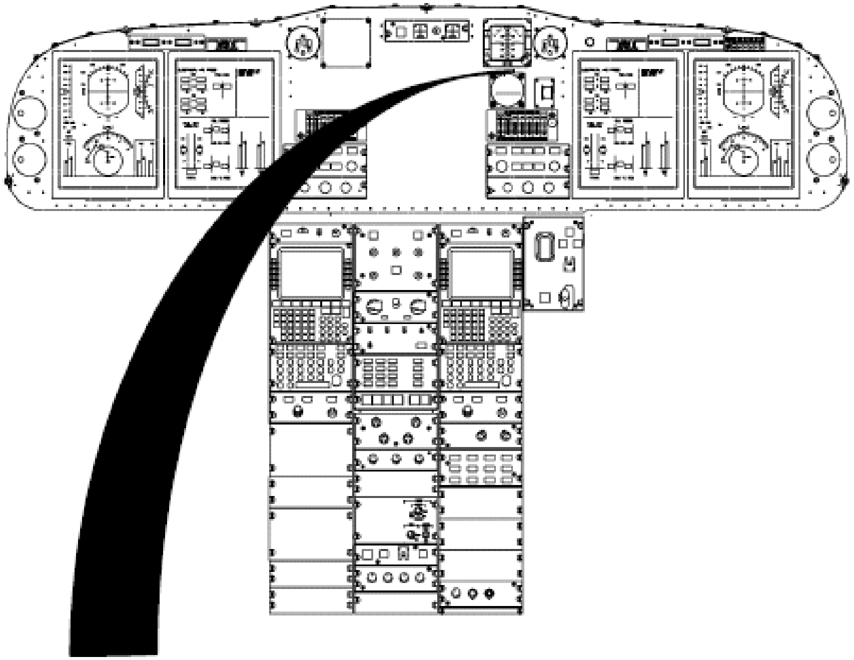
CLPR UP	Rotor brake caliper ready for brake (only on ground)
CLPR DOWN	Rotor brake caliper out of braking position (only on ground)

CAS ADVISORY MESSAGES

CAS CAPTION	MESSAGE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
ROTOR BRK ON	Rotor brake selected to BRAKE position		Supplement 1 Section 2 NORMAL PROCEDURES

ROTOR BRAKE - LIMITATIONS

Refer to AW139-RFM-4D.



APPLY ROTOR BRAKE AT 40% NR

CHAPTER
64
TAIL ROTOR

SECTION 00 – GENERAL

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TAIL ROTOR – GENERAL

The Tail Rotor (TR) compensates the torque caused by the main rotor and enables to control the aircraft in the yaw axis.

The TR is a four blade fully articulated rotor with elastomeric bearings that allow flapping, lead-lag and pitch change.

TAIL ROTOR – MAIN COMPONENTS

TAIL ROTOR HEAD

The tail rotor head is installed on the Tail Gear Box (TGB) mast.

HUB

The hub that connects the TGB shaft to the blades. The hub is constituted by four arms. An upper and a lower limiter are used to limit the flapping movements.

ELASTOMERIC BEARINGS

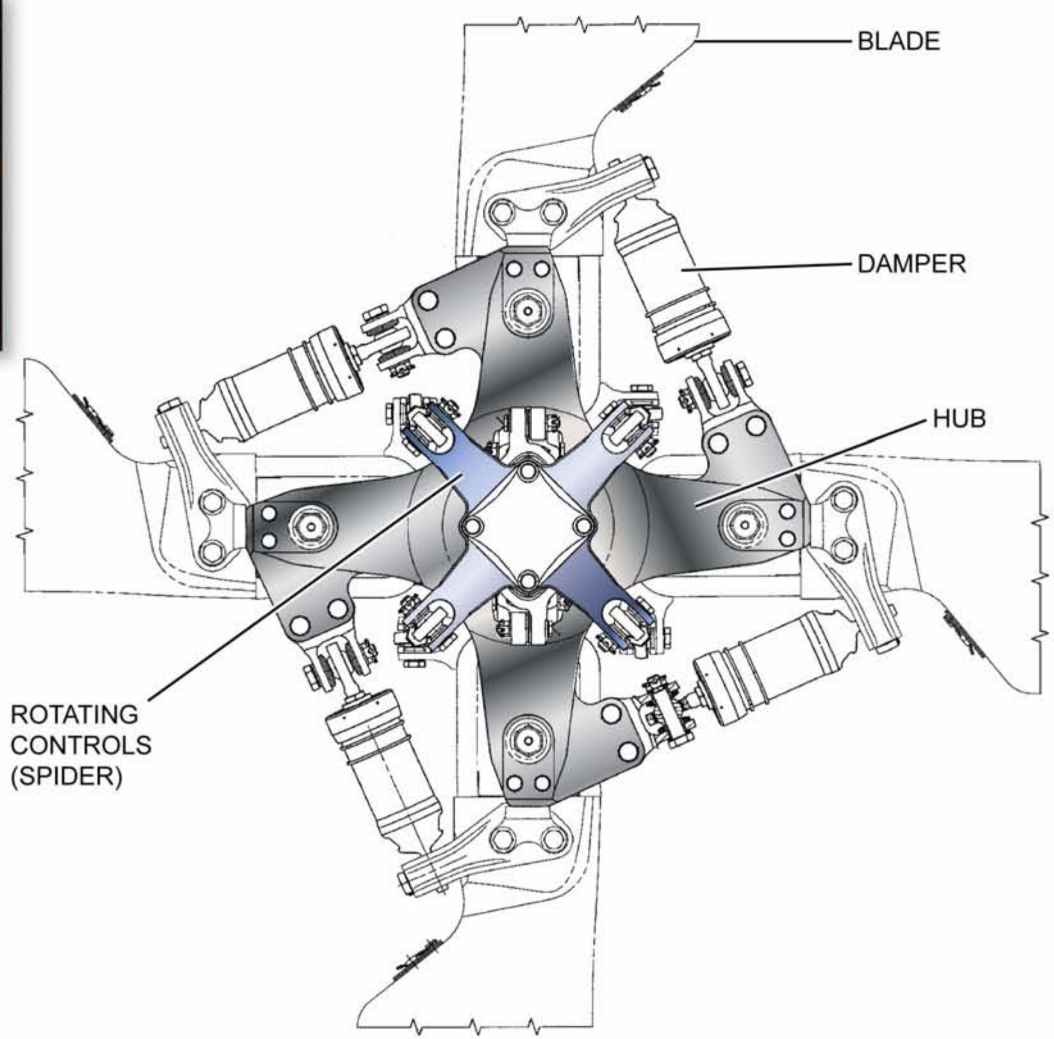
Four elastomeric bearings make the connection between the blades and the hub and permit lead-lag, flap, and pitch changes of the blades.

DAMPERS

The dampers are installed between the blades and the hub to damper the lag movements of the blades. The stops limit the flap movement of the blades.

COVER

A cover, made by aluminium alloy, is installed on the top of the rotating controls.



TAIL ROTOR – MAIN COMPONENTS

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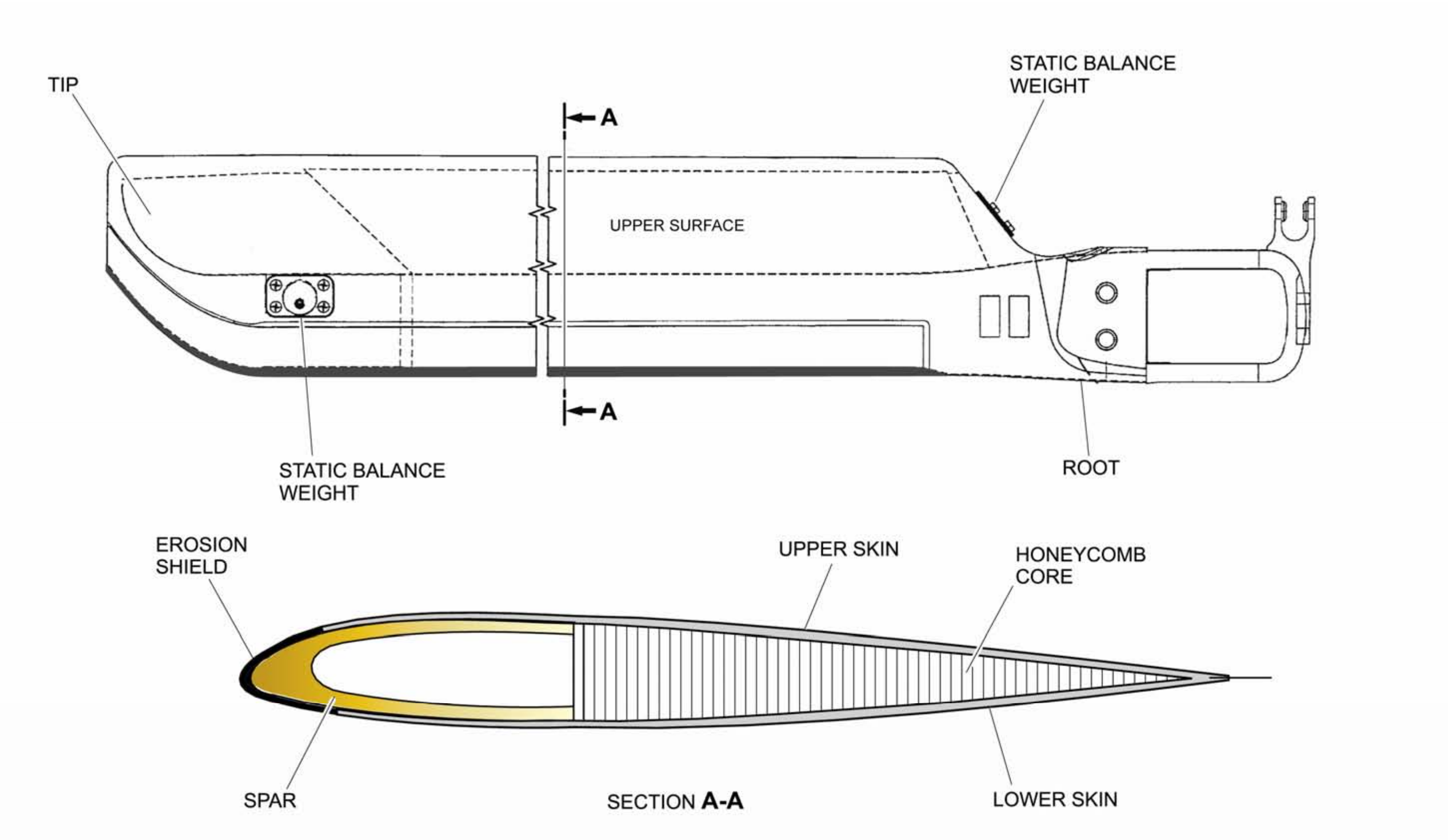
BLADES

The blades are made of composite material except for the erosion shield (that is made of stainless steel) and some minor part.
The blades have a very long service life and are statically balanced as final manufacturing process to achieve an individual interchangeability. In fact the blade tip includes a pocket where masses are introduced to statically balance the blade; the same occur in the root section.
The blades are composed of a constant chord profile with a parabolic tip.
The metallic pitch change arm lever is installed on the blade.
A lightning conductor jumper is attached to the top of each blade attachment.

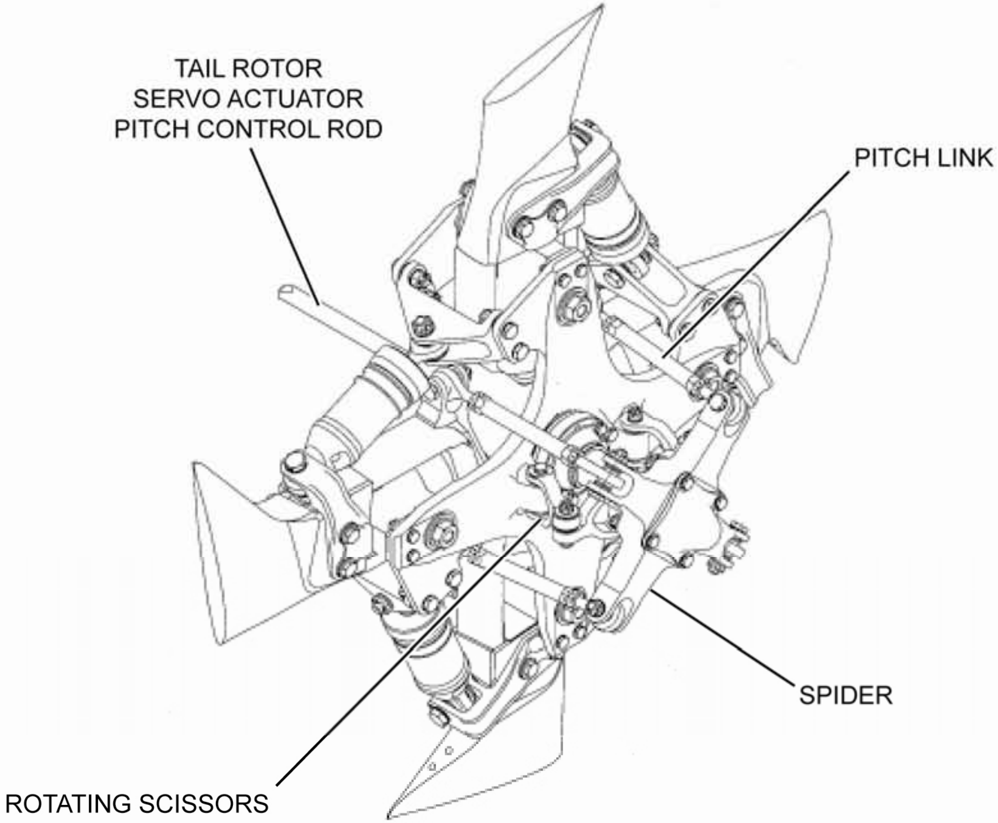

TIP
STATIC BALANCE
WEIGHT
A
UPPER SURFACE
A
STATIC BALANCE
WEIGHT
ROOT
EROSION
SHIELD
UPPER SKIN
HONEYCOMB
CORE
SPAR
SECTION A-A
LOWER SKIN

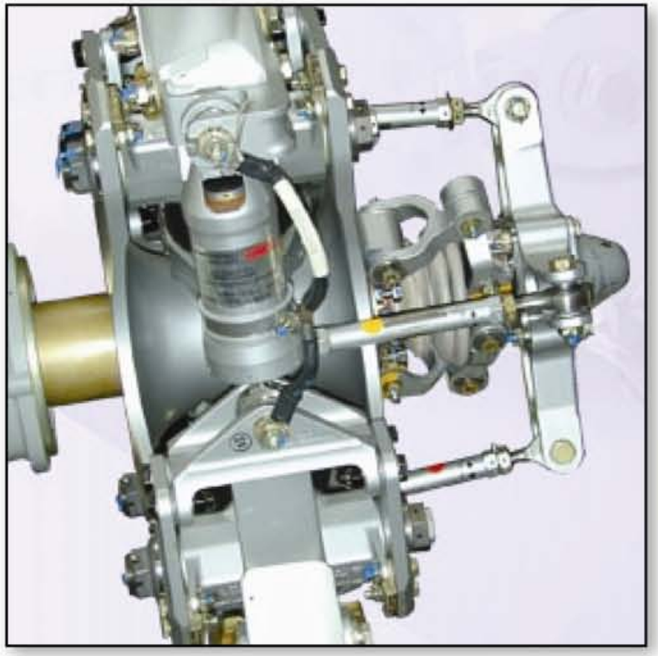
BLADES

ROTATING CONTROLS

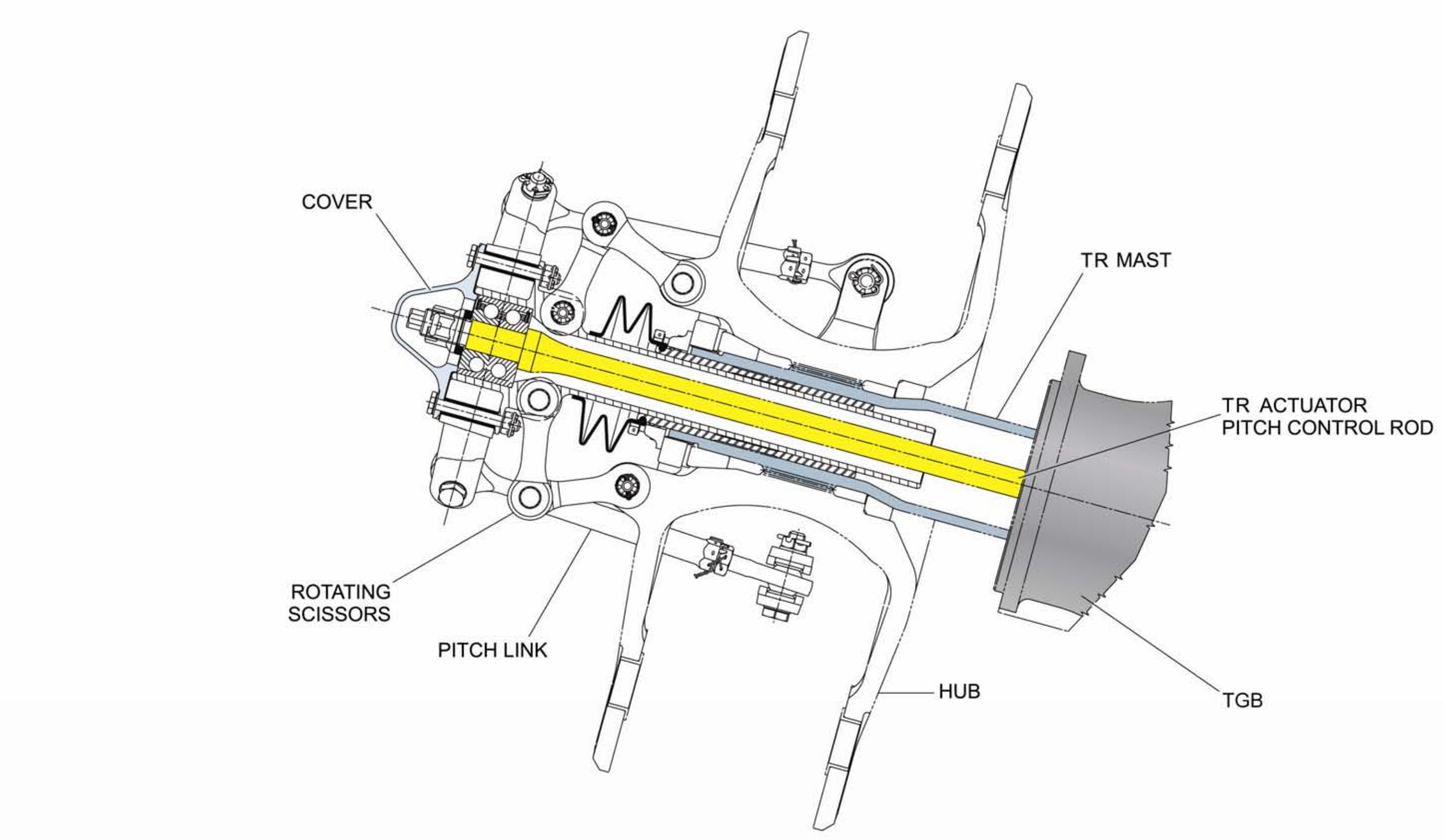
The rotating controls receive inputs from the TR actuator and transform give the required blade pitch-angle changes.
The TR controls include:

- the spider mounted over a sliding tube. The sliding tube is installed on the TGB mast and supplies the attachments for one end of the rotating scissors;
- the rotating scissors that supply a rotary drive force from the hub to the spider. At the same time the rotating scissors turn to allow lateral pitch-change movement;
- four pitch links, one for each blade, that connect the TR blades to the spider.





TAIL ROTOR – ROTATING CONTROLS (1 OF 2)



TAIL ROTOR – ROTATING CONTROLS (2 OF 2)

CHAPTER 65
TAIL ROTOR DRIVE

SECTION 00 – GENERAL

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TRANSMISSION

The transmission supplies the drive from the engines to the Main Rotor (MR), the Tail Rotor (TR) and the accessories (pumps, ECS compressor, and alternators - if installed).

The transmission is composed by:

- the Main Rotor Drive System (MRDS)
- the Tail Rotor Drive System (TRDS)

as visualized in the schematic that follows. This chapter describes the Tail Rotor Drive System.

TAIL ROTOR DRIVE SYSTEM – GENERAL

The Tail Rotor Drive System transmits the rotational drive from the Main Gear Box to the Tail Rotor. The TRDS comprises:

- the tail rotor drive shafts;
- the Intermediate Gear Box (IGB);
- the Tail Gear Box (TGB).

The Intermediate Gearbox (IGB) and the Tail Gearbox (TGB) transmit the drive while changing the drive angle. The gearboxes also reduce the rotational speed between the input and output drive. Each gearbox has a self-contained splash lubrication and condition-monitoring oil system.

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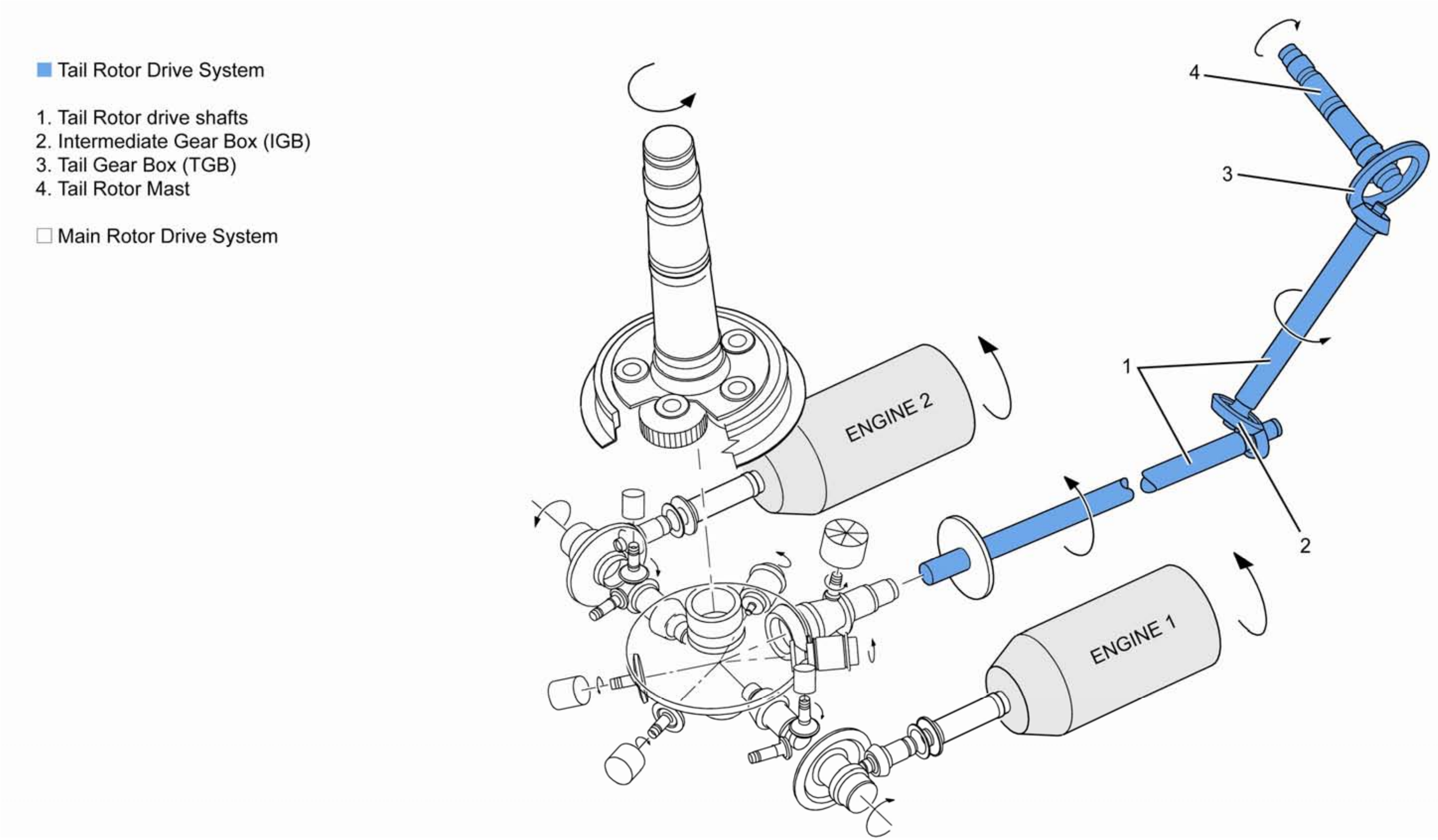


TRANSMISSION – GENERAL

TRDS – MAIN COMPONENTS

TAIL ROTOR DRIVE SHAFTS

The tail rotor drive shafts make the connection and transmit the torque between the gearboxes. Two different length shafts connect the Main Gear Box (MGB) to the IGB. The drive shaft no. 1 starts at the forward end of the drive-train and connects the MGB tail flange coupling to a support bearing assembly. The same support assembly makes a connection to the forward end of shaft no. 2. The rear end of this no.2 shaft connects to the IGB input drive pinion.
One slanted shaft connects the IGB to the TGB.
Shafts no.1 and no.2 have damper-assemblies used to dampen the shaft flexing movements that can happen at critical rotational speeds.
The drive shaft no.2 is a critical component because of a resonance can occur. For this reason an anti-flail assembly is installed in case of failure or disconnection of shaft no. 2.

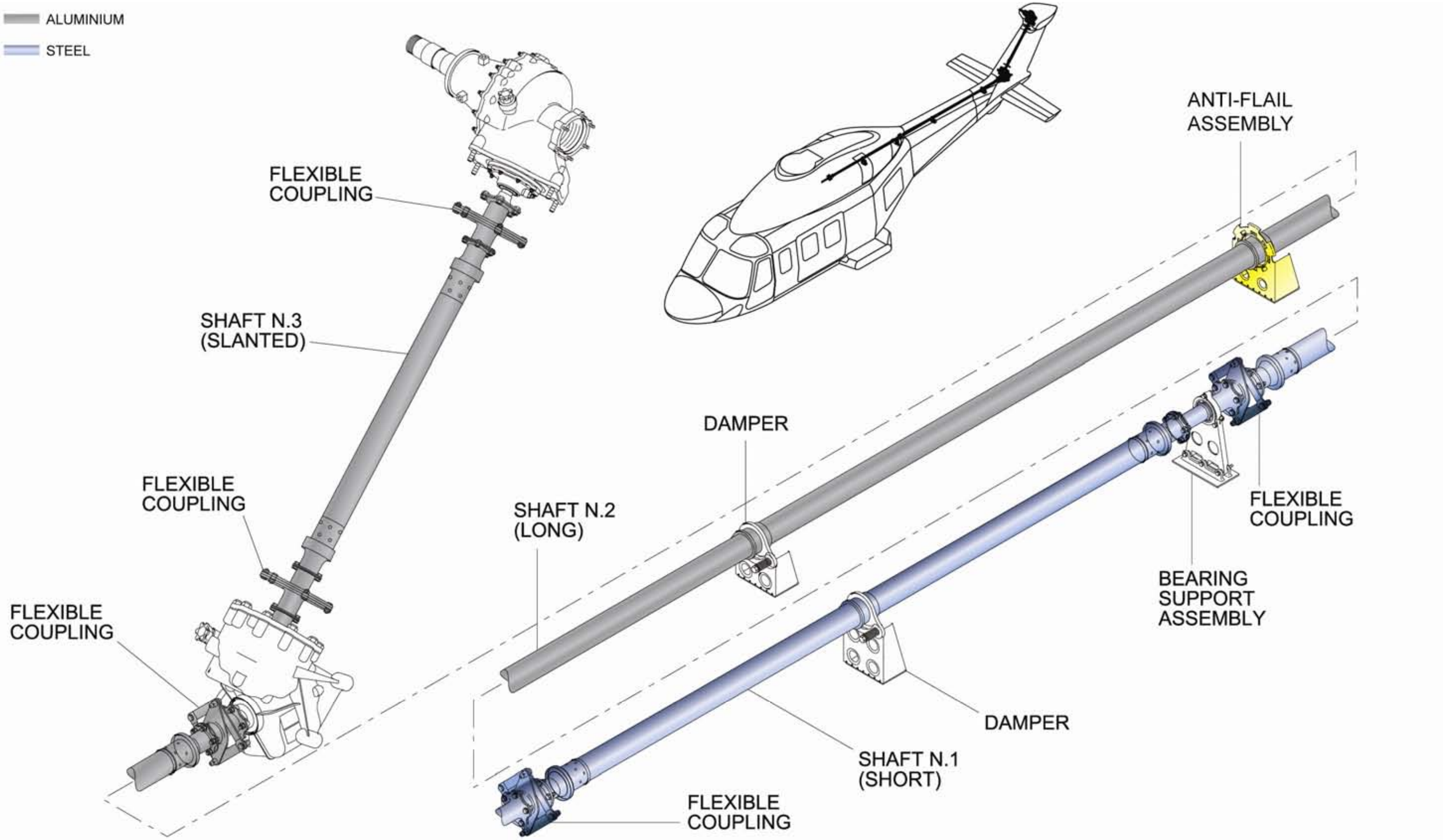
The damper no.1 is installed on the drive shaft no.1 and decreases the vibrations transmitted to the airframe.

INTERMEDIATE GEAR BOX (IGB)

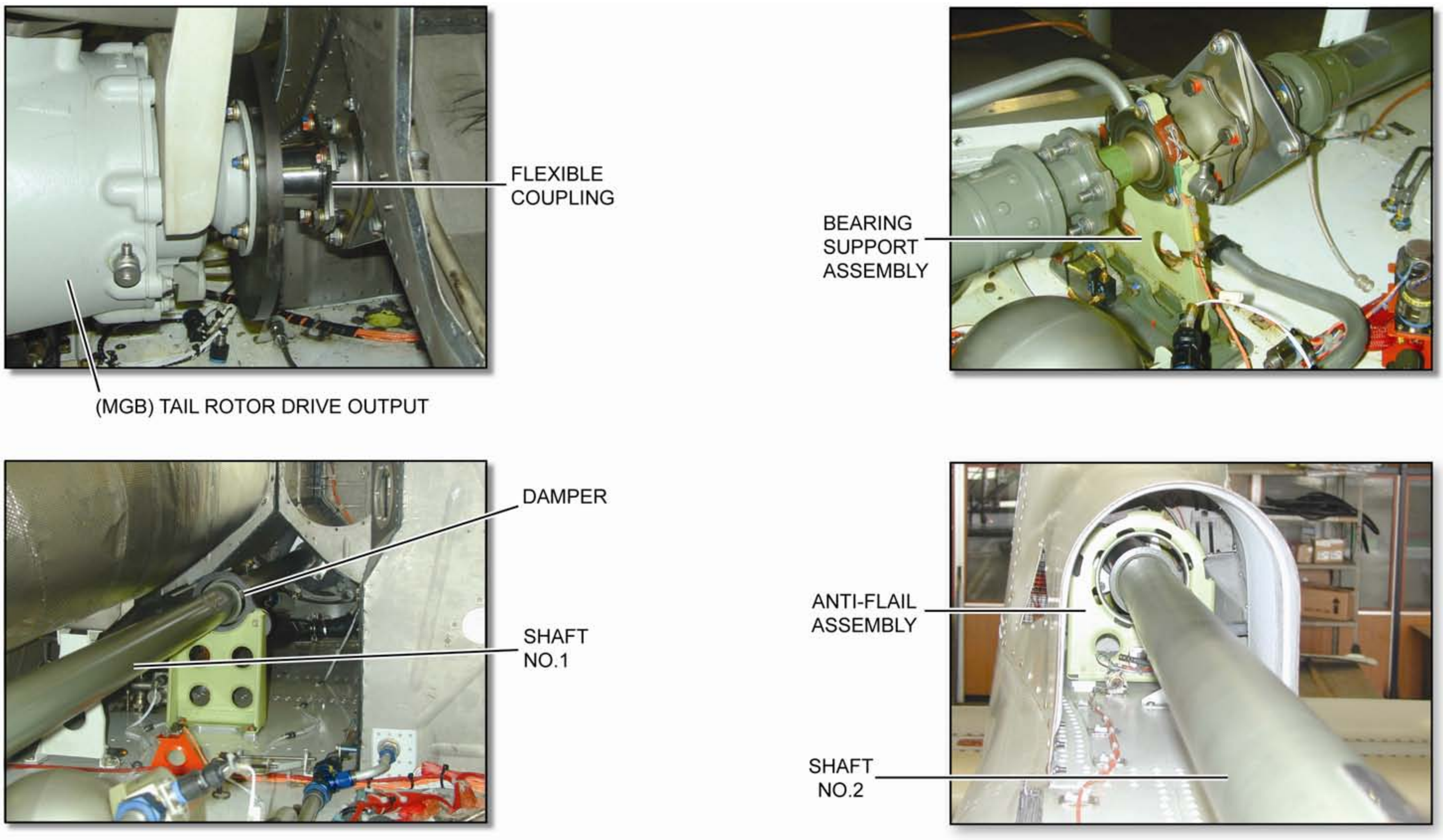
The Intermediate Gear Box (IGB) is installed on the lower tail-fin structure and changes the direction of the drive to the TGB. The IGB reduces the rotational speed from the input speed of 4532 rpm to the output speed of 3458 rpm.

The IGB is splash lubricated and includes one oil low level sensor, one oil temperature sensor, one accelerometer and one chip detector.

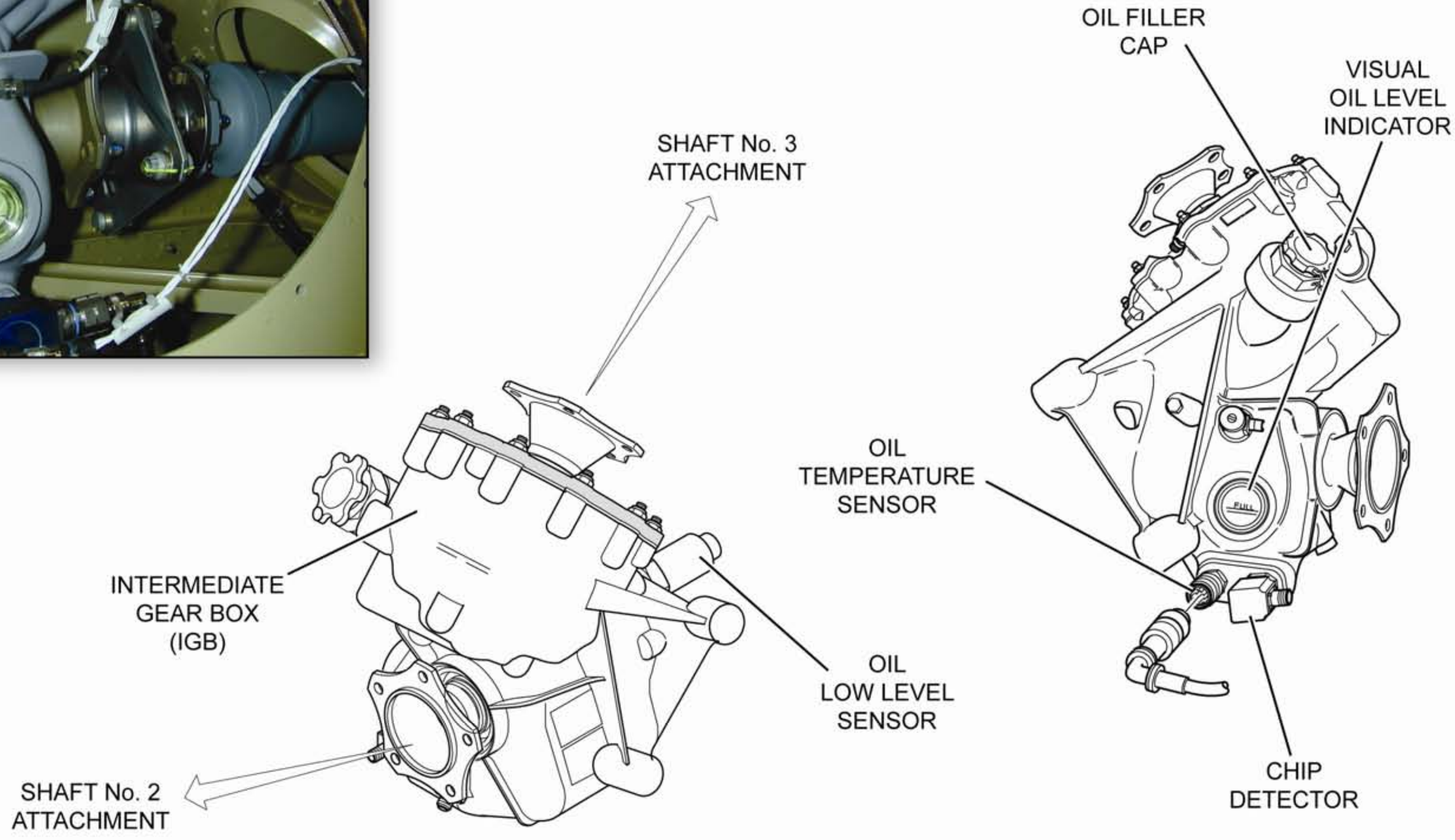
A visual oil level indicator allows to monitor the oil level for maintenance purposes.



TAIL ROTOR DRIVE SHAFTS



TAIL ROTOR DRIVE SYSTEM – MAIN COMPONENTS



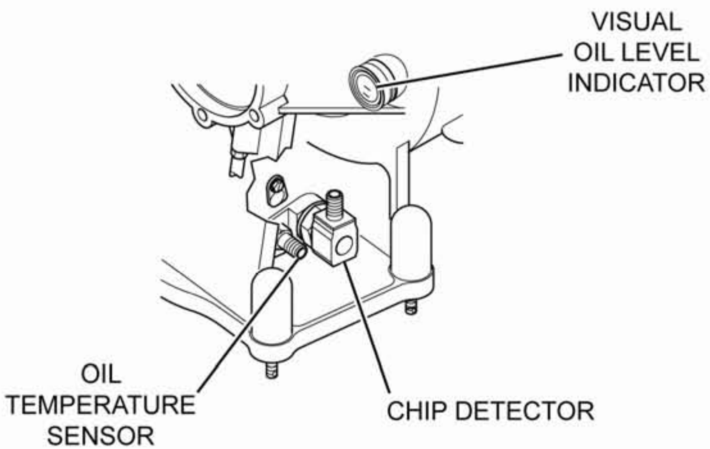
INTERMEDIATE GEAR BOX (IGB)

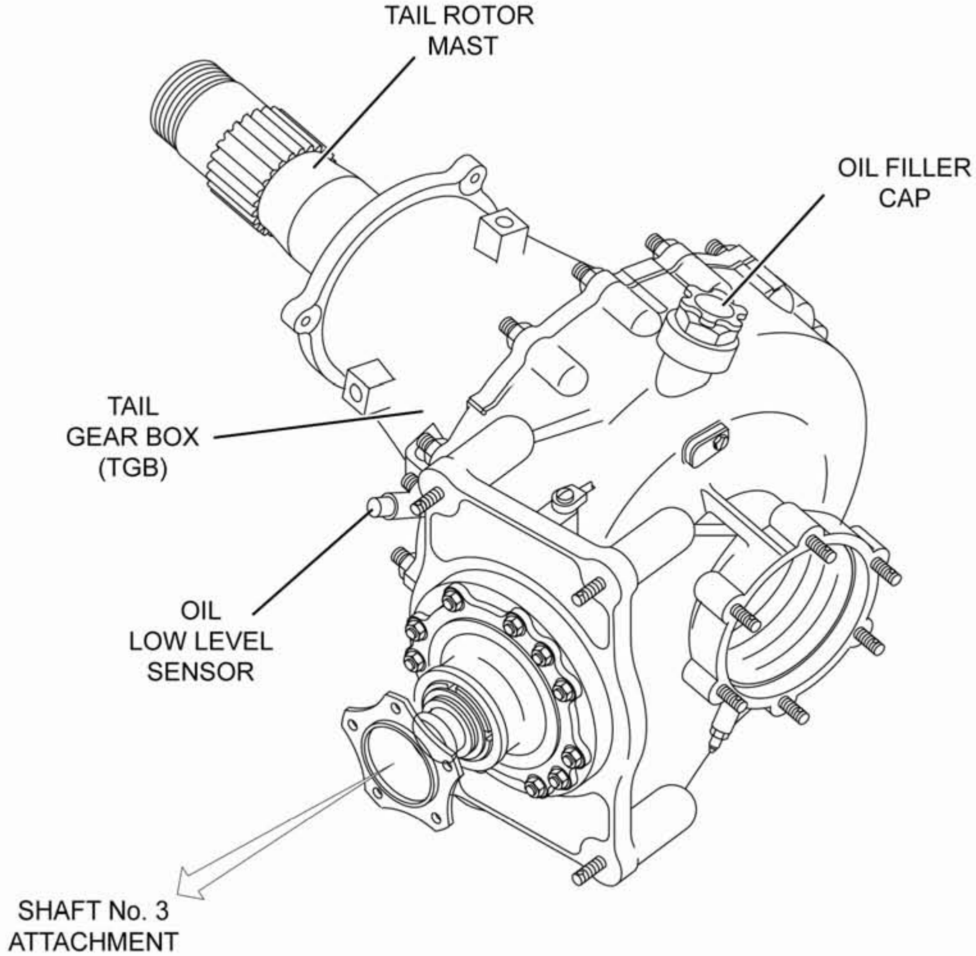
TAIL GEAR BOX (TGB)

The Tail Gear Box is installed on the top of the fin. The TGB reduces the input speed of 3458 rpm to the output speed of 1435 rpm which is the nominal speed of the TR.

The TGB is splash lubricated and includes one oil low level sensor, one oil temperature sensor, one accelerometer and one chip detector.

A visual oil level indicator allows to monitor the oil level of the TGB for maintenance purposes.





TAIL GEAR BOX (TGB)

TAIL ROTOR DRIVE – INDICATIONS

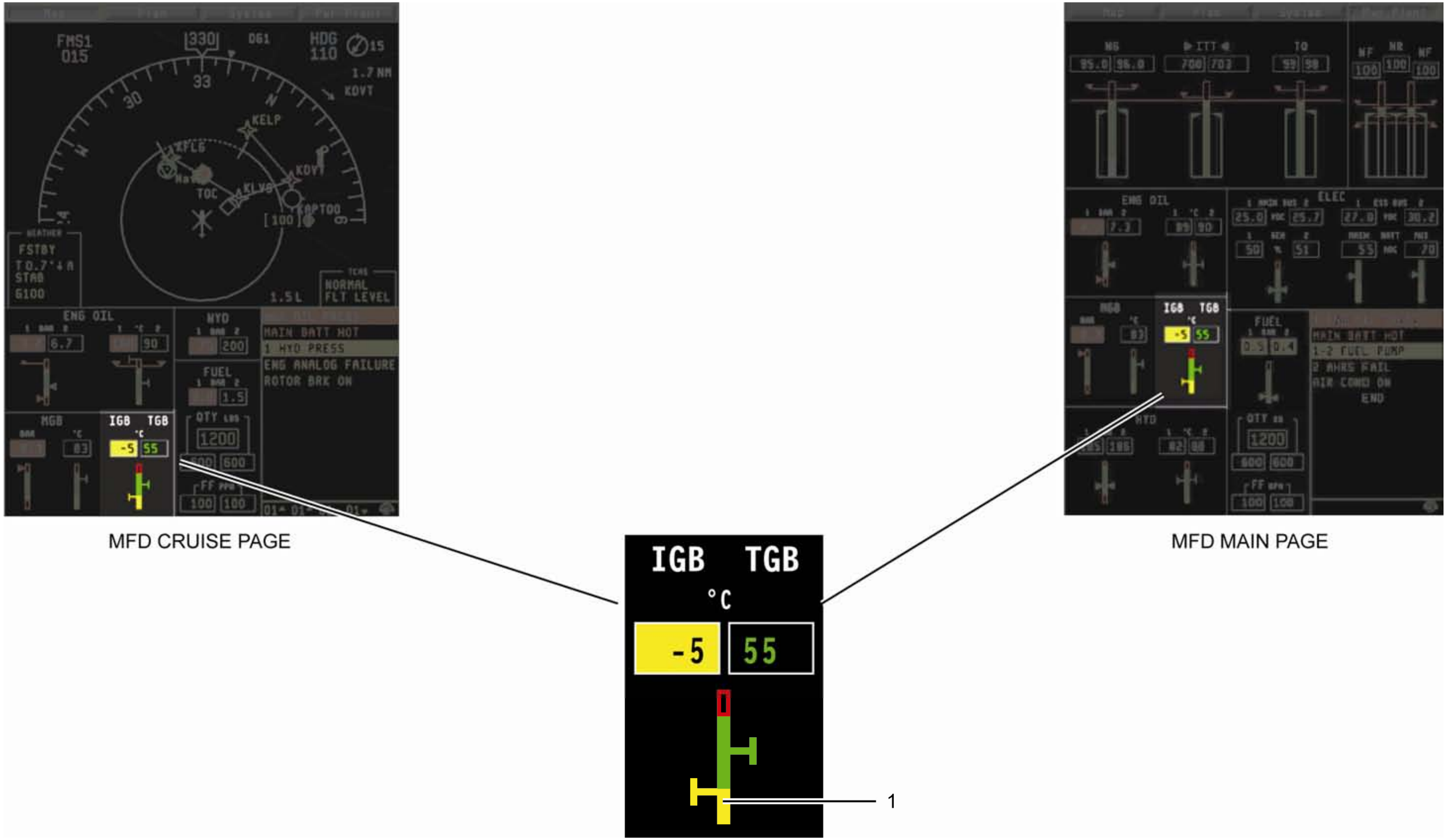
The PWR PLANT PAGE displays the values of temperature in the IGB (Intermediate Gear Box) and TGB (Tail Gear Box) areas.

1. TEMPERATURE

   The oil temperature values in the IGB and in the TGB are represented by two digital readouts under the label °C (on the left are displayed the IGB values of temperature; on the right are displayed the TGB values of temperature). Graphically the values are represented on a vertical scale by means of pointers (T symbols) that match the color of the area on the scale.

   Green band of the analogue vertical scale represents a normal condition for the oil temperature and so the associated digital readout values in Celsius degrees.

   The amber band represents a caution condition while the red band is associated to a warning condition.



MAIN ROTOR DRIVE – INDICATIONS

TAIL ROTOR DRIVE – CONTROLS AND INDICATORS

1. CHIP BURNER push-button

   PRESSED ..……. a chip burning attempt for the IGB CHIP and TGB CHIP is performed

2. OIL LEVEL IGB push-button

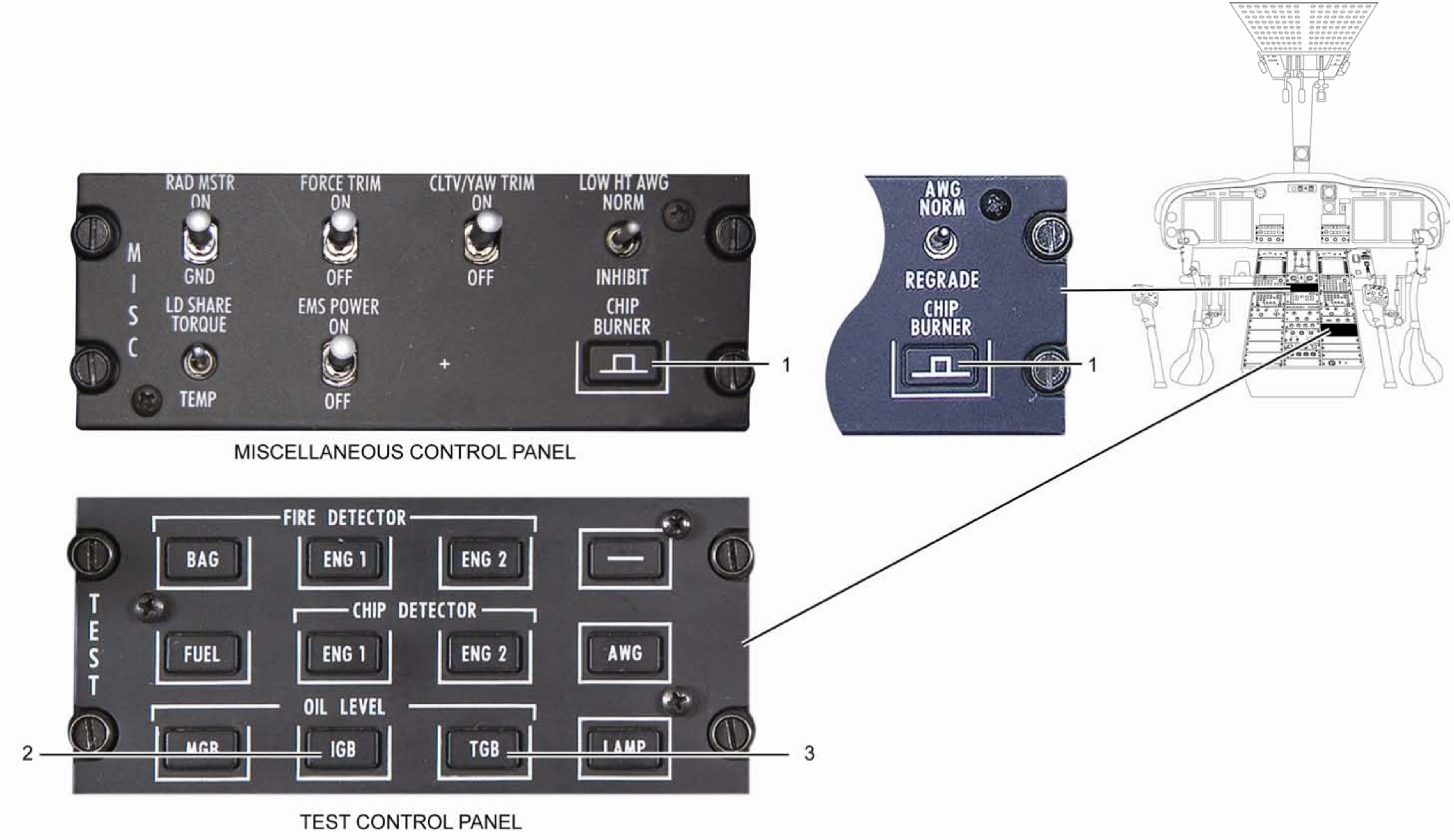
   PRESSED ……… the caution IGB OIL LOW in inverse video is displayed in the CAS window

   The test is possible only when the helicopter is on the ground and the NR is less than 2%

3. OIL LEVEL TGB push-button

   PRESSED ……… the caution TGB OIL LOW in inverse video is displayed in the CAS window

   The test is possible only when the helicopter is on the ground and the NR is less than 2%



TAIL ROTOR DRIVE – CONTROLS AND INDICATORS

CHIP DETECTOR SYSTEM

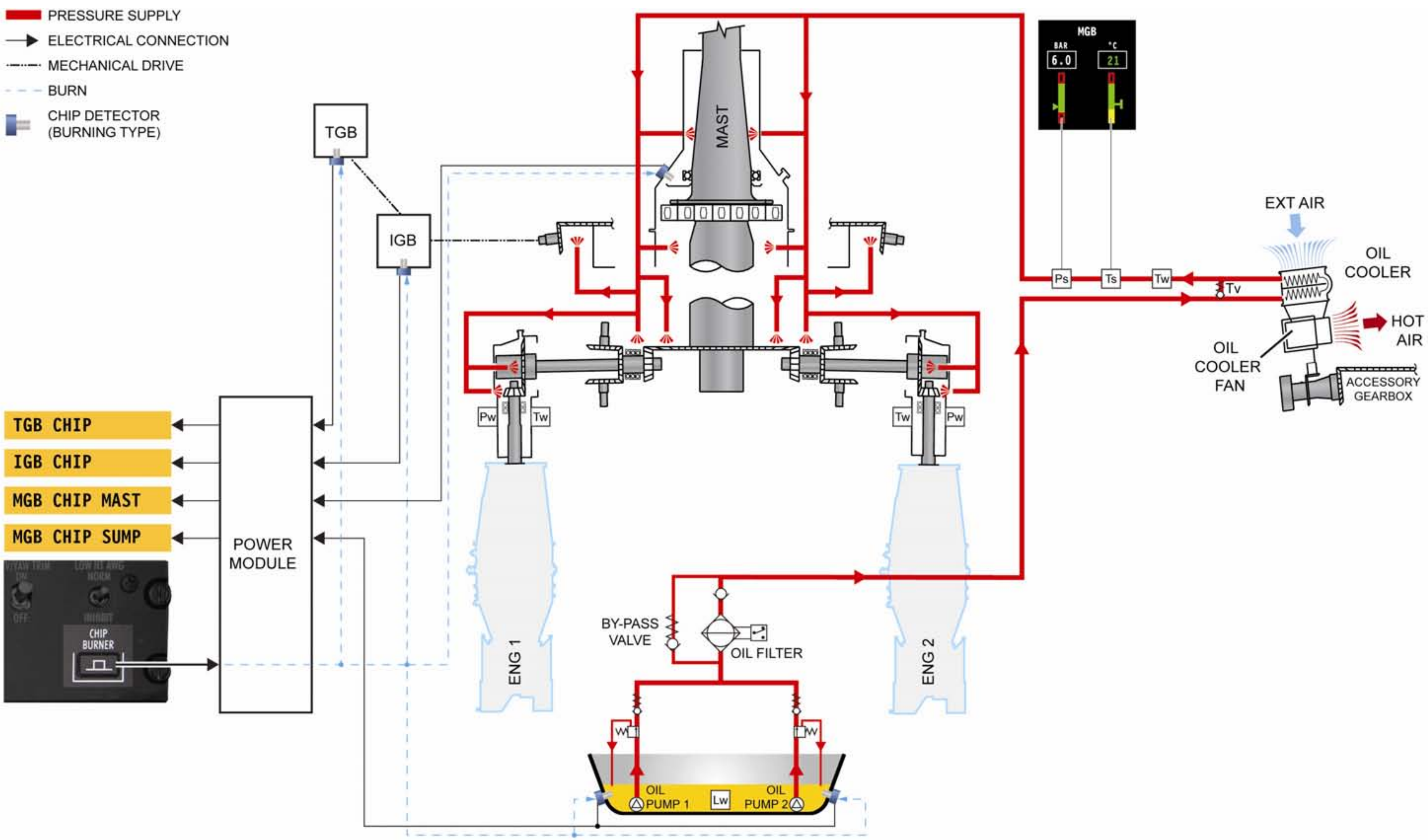
The chip detector system is described in the Ch.63-00-00. On the TRDS there are two chip detectors: one chip detector is installed on the IGB and one chip detector is installed on the TGB.

When the IGB CHIP or the TGB CHIP caution messages are displayed in the CAS window of the MFD, the pilot can try to burn the ferrous particle(s) pushing the CHIP BURNER switch located on the MISC control panel.

If the burning is successful the particle is classified as “small” and the relevant caution extinguishes.

In the opposite case, the particle is considered “large” and cannot be burned.

When the particle is burned, a message is written in the Non Volatile Memory (NVM) for maintenance purposes.



CHIP DETECTOR SYSTEM

CAS CAUTION MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
IGB CHIP	Activated CHIP burner (on MISC control panel) It is permitted to activate the CHIP BURNER up to 3 times to clear a chip in one flight. On the 4th CHIP caution Land As Soon As Practicable	INTERMEDIATE GEARBOX CHIP	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES DRIVE SYSTEM
TGB CHIP		TAIL GEARBOX CHIP	
IGB OIL LOW	IGB oil temperature above limit (greater than109°C)	INTERMEDIATE GEARBOX OIL LOW	
TGB OIL LOW	IGB oil temperature above limit (greater than109°C)	TAIL GEARBOX OIL LOW	
IGB OIL TEMP	IGB oil level low (caution only active with engines shut down and NR below 2%)	INTERMEDIATE GEARBOX OIL TEMPERATURE HIGH	
TGB OIL TEMP	TGB oil level low (caution only active with engines shut down and NR below 2%)	TAIL GEARBOX OIL TEMPERATURE HIGH	
CHIP DET UNIT CHIP DET TEST	Chip detect system malfunction	GEARBOX CHIP DETECT UNIT MALFUNCTION	
IGB CHIP FAIL TGB CHIP FAIL	Associated gearbox chip sensor failed	GEARBOX CHIP DETECTOR SENSOR FAILURE	

CHAPTER
67
ROTOR FLIGHT CONTROLS

SECTION 00 – GENERAL

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ROTOR FLIGHT CONTROLS – GENERAL

The rotor flight controls allow to manage the flight attitude, altitude and direction of the aircraft. The control is transmitted by means of mechanical linkages that interface with manual input controls (collective, cyclic and pedals both pilot and copilot) and Automatic Flight Control System (AFCS) input controls.

The rotor flight controls include:

- the main rotor controls system
- the tail rotor controls system
- the rotor flight controls indicating system

ROTOR FLIGHT CONTROLS - GENERAL

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MAIN ROTOR CONTROLS SYSTEM – GENERAL

The main rotor controls system includes

- the collective control
- the cyclic control

The two controls are mixed, before to be sent to the main rotor servo actuators, to obtain the desired attitude and altitude.

COLLECTIVE CONTROL – GENERAL

The collective control is a conventional metallic rod and bellcrank type. The control is composed of a lever for pilot and copilot that make a common input to a mixing unit. The collective control sticks are made of aluminium with a control grip at the top. The levers are connected together by a torsion tube.

At the right end of the tube is installed an adjustable friction used to increase the collective control resistance or hold the stick during manual flight.

At the left end of the tube are installed the two Linear Variable Differential Transducers (LVDT) which provide power anticipation to the engines.
From the control tube a rod extends vertically to the roof and them, through another torsion tube inputs, to the mixing unit.

COLLECTIVE TRIM ACTUATOR

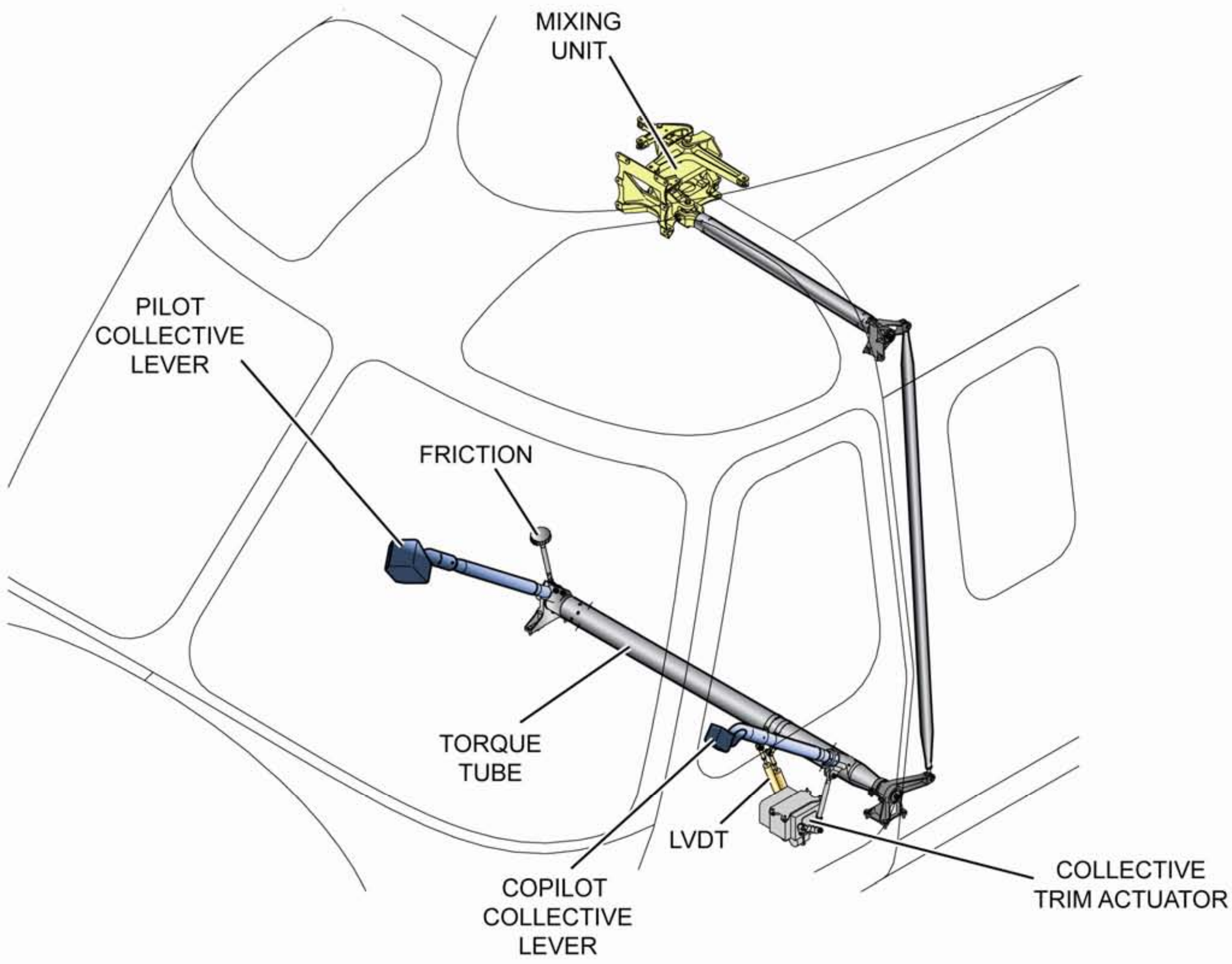
A collective trim actuator is installed in parallel with the torque tube. It has the same control travel authority as the pilot but with a low-limited rate of control for safety in case of a malfunction occurs. The trim actuator can be overridden by pilot commands via a spring-clutch.

COLLECTIVE CONTROL – PRINCIPLE OF OPERATION

The pilot and copilot collective levers are connected together to give a common input to the mixing unit.

The mixing unit, in turn, gives a common output towards the three hydraulic main rotor servo actuators which change the pitch angle of the main rotor blades to obtain the desiderated change of attitude and altitude.

An electrical trim actuator connected to pilot and copilot collective provides an automatic input towards the AFCS.



COLLECTIVE CONTROL

CYCLIC CONTROL – GENERAL

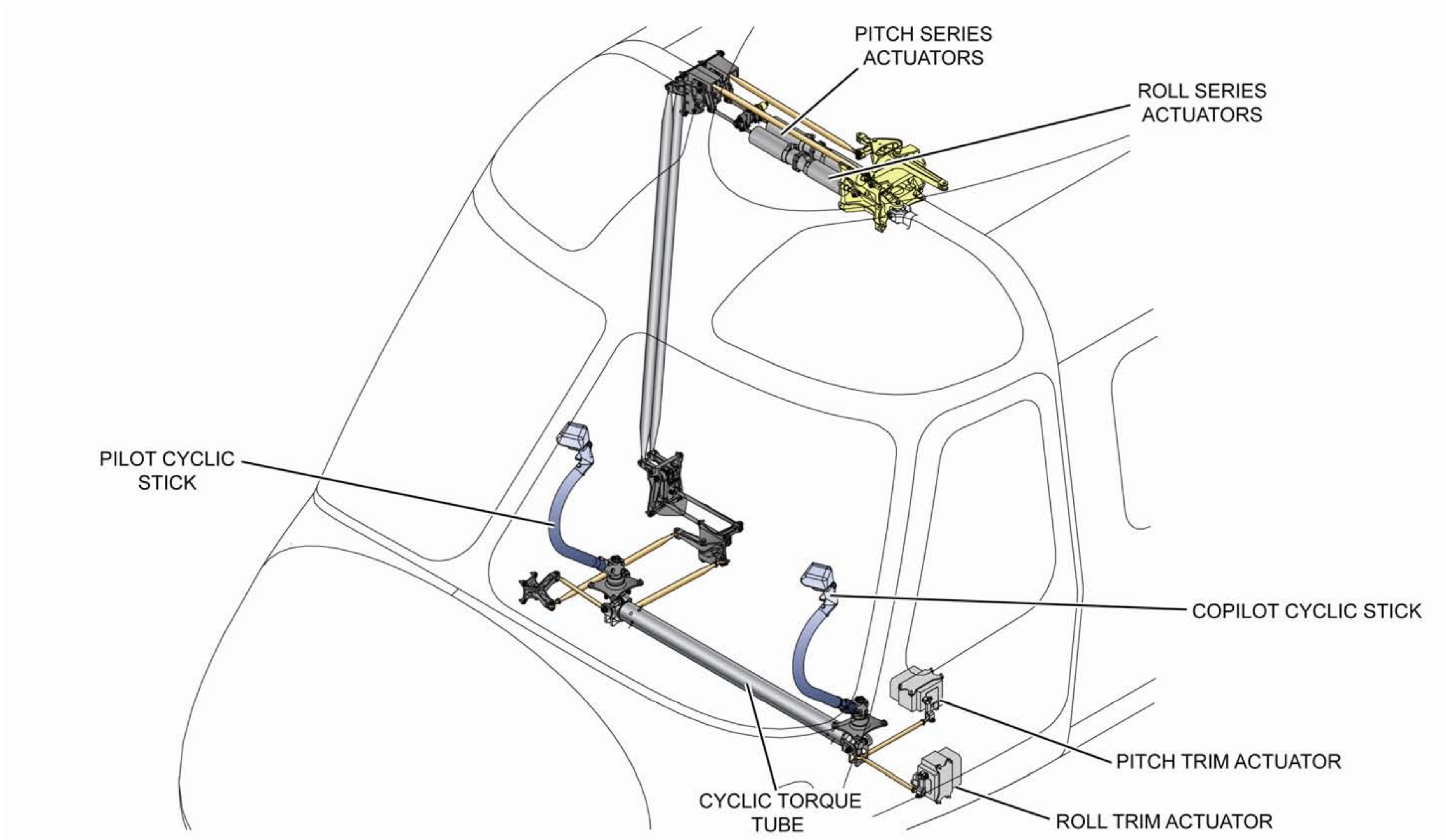
The cyclic control is a conventional metallic rod and bellcrank type. The control is composed of a stick for pilot and copilot that make a common input to a mixing unit. The collective control sticks are made of aluminium with a control grip at the top. The pilot and copilot stick are connected together by a torsion tube. The sticks are connected to the floor using an attachment plate containing a spherical bearing which is connected with a cross shaped, stainless steel fitting to a torsion tube.
The pilot attachment plate contains a shaped flange used to limit pitch and roll movements combination.
To statically balance the control linkage weight, springs are installed on the stick cross-shaped fitting.
At the left end of the torsion tube, the cross-shaped fitting allows for the connection with rods to the longitudinal and lateral trim actuators.
From the right side the control is connected with a lower bellcrank installation and then with rods that extend vertically to an upper bellcrank installation.
Each lower bellcrank installation has a friction clamp (fixed friction) for counteracting the output force of the series actuators, when the pitch/roll trim actuators are declutched, thus ensuring the movement is translated to the mixing unit.
At the upper bellcrank installation there is a compound linkage connection which transmits the longitudinal and lateral control to the mixing unit. Interposed between the compound linkage and the mixing unit are the AFCS series actuators. Acting on the rod that connects between each series actuators and the compound linkage is an anchor spring which provides a safety function for the event of an actuator disconnect, in that the spring will react the manual input as a pivot point to ensure the movement is translated to the mixing unit.
The compound linkage then mixes the pilot input with the series actuators (pitch and roll dual linear actuators) input and transmits this output to the mixing unit. Limit stops are present in the upper bellcrank to limit the longitudinal and lateral cyclic pitch to the limits of its envelope.

PITCH AND ROLL TRIM ACTUATORS

Pitch and roll trim actuators are installed in parallel with the cyclic stick. Trim actuators have the same control travel authority as the pilot but with a low-limit rate of control for safety in case of malfunctions.
Trim actuators can be overridden by pilot commands via a spring-clutch. Actuators are operated by pilot commands or by the AFCS. They convert the command to a mechanical output.

CYCLIC CONTROL – PRINCIPLE OF OPERATION

The cyclic control is a mechanical, dual linkage system, actuated by the cyclic control sticks.
The two cyclic inputs are transmitted to the mixing unit which sends the output to the three main rotor actuators.
The mechanical linkage also gives a connection to the pitch and roll trim actuators which can also give pitch and roll commands.



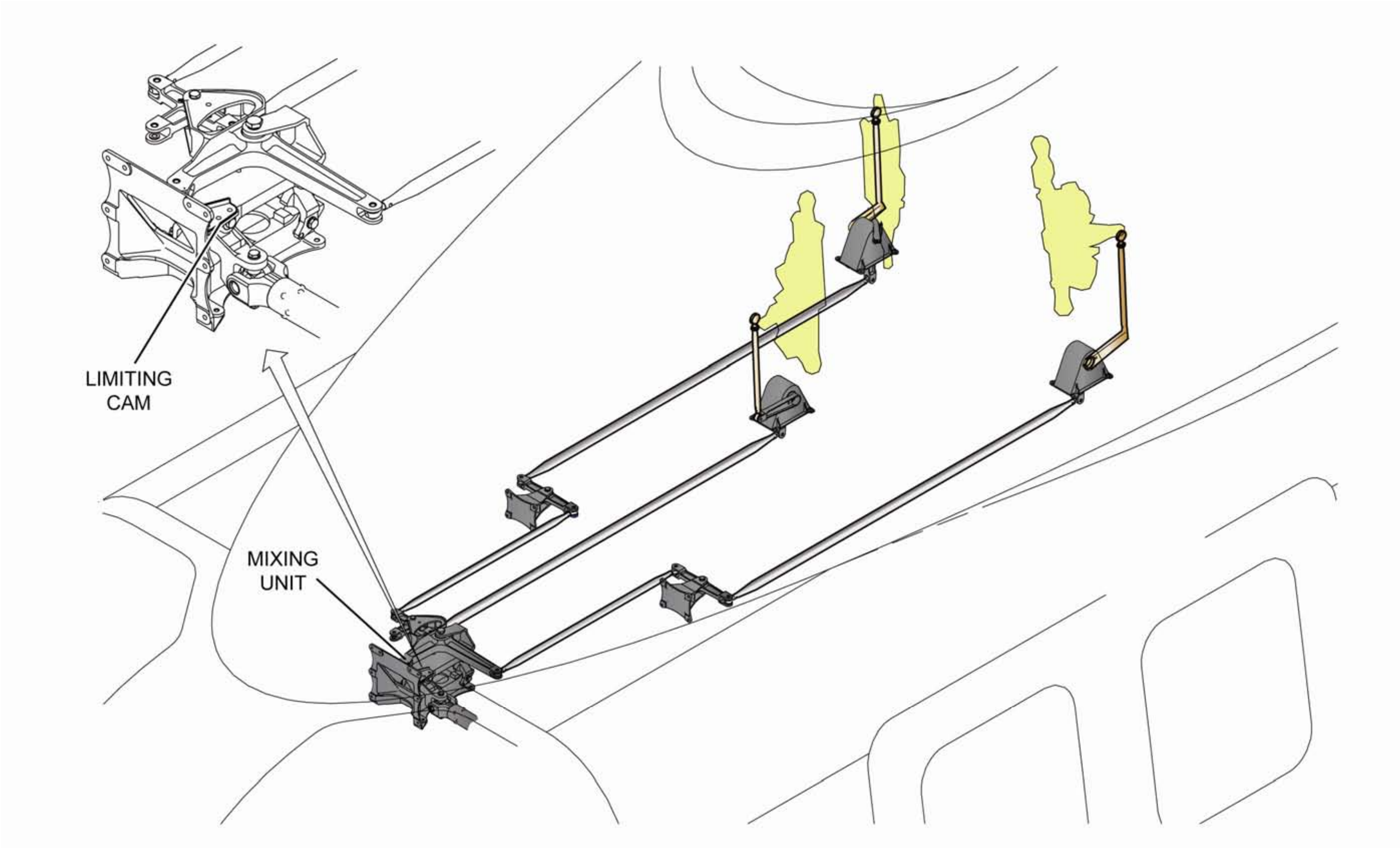
CYCLIC CONTROL

MIXING UNIT

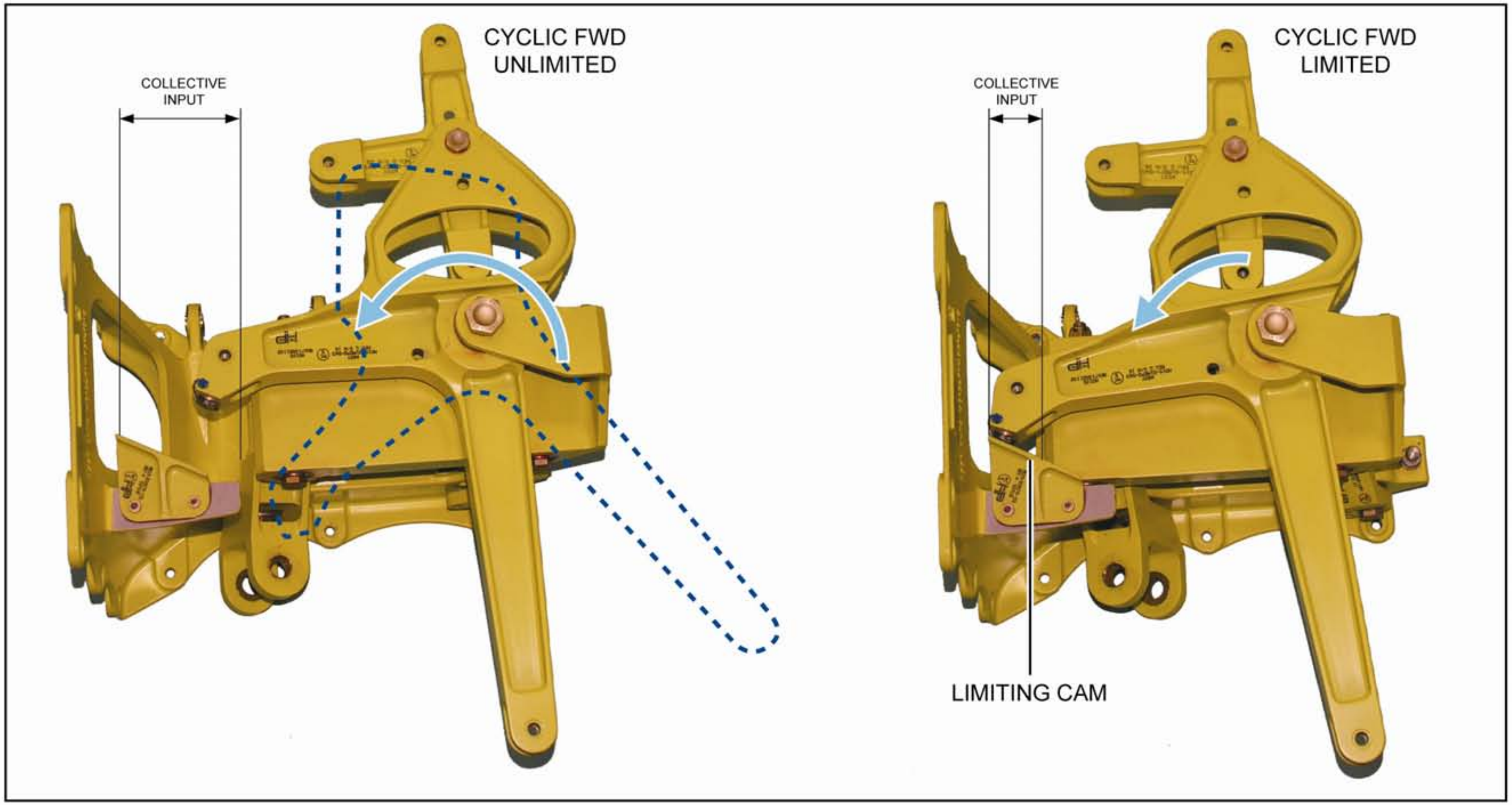
The mixing unit is a mechanical equipment made by a linkage on a movable support assembly.
The mixing unit receives a mechanical input from the collective controls and cyclic controls and, after mixing them, sends a mechanical output to main rotor servo actuators.

The mixing unit includes a mechanical stop that restricts the longitudinal pitch control as a function of collective displacement; full forward longitudinal cyclic control is achieved only if collective control is not at minimum.

The output from the mixing unit to the servo actuators comprises three rods that use a pivot lever and bell cranks to transmit the mechanical movement.



MIXING UNIT



COLLECTIVE UP

COLLECTIVE DOWN

COLLECTIVE DISPLACEMENT AND LONGITUDINAL PITCH CONTROL

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SERVO CONTROLS

MAIN ROTOR SERVO ACTUATORS - GENERAL

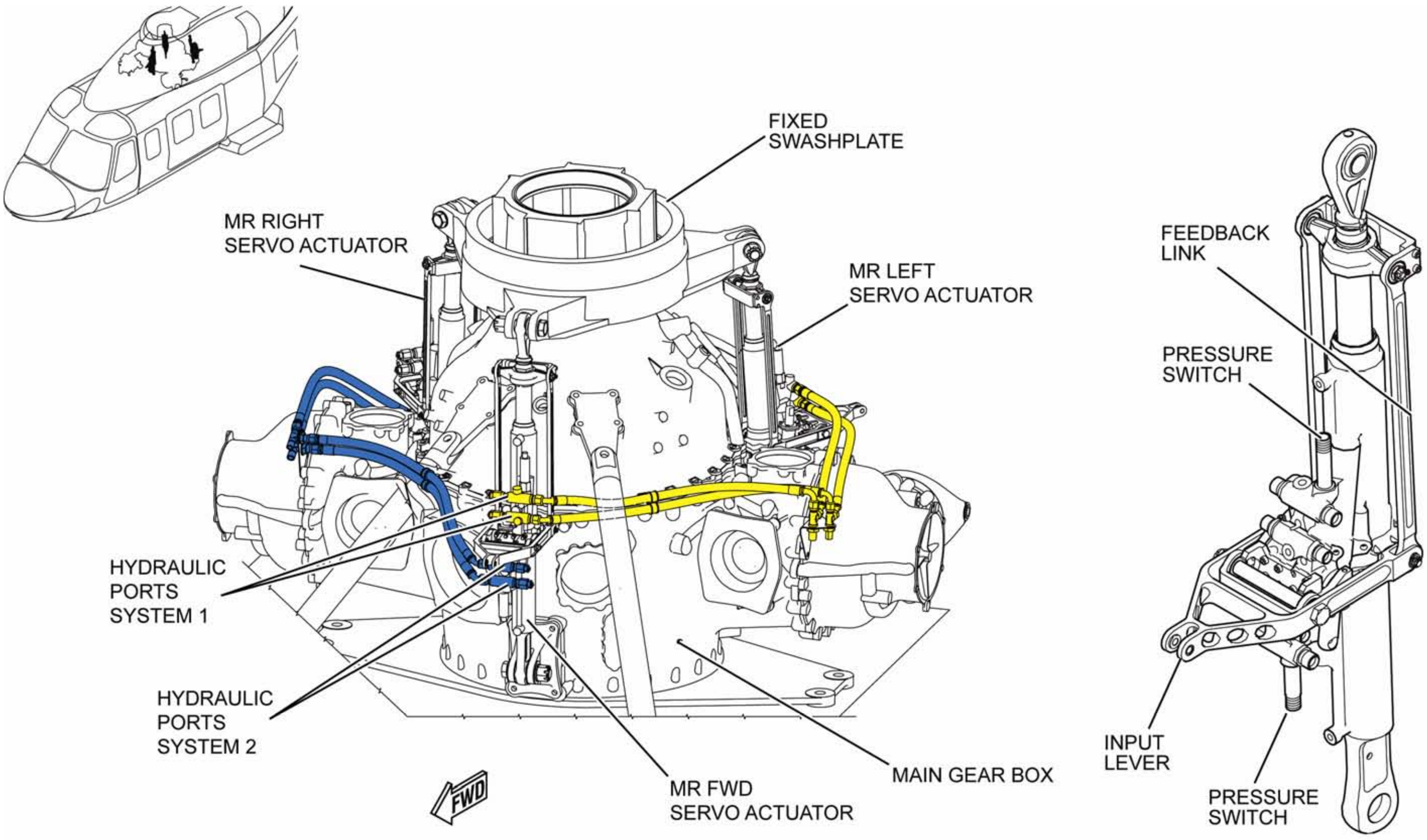
There are three Main Rotor (MR) servo actuators: LEFT, RIGHT and FORWARD (LH, RH and FWD).
Each main rotor servo actuator is a fixed body actuator comprising two separate cylinder assemblies bolted together at the actuator center and tandem pistons.

The upper piston is connected to the fixed swashplate; the lower end of the cylinder is connected with a support to the MGB.
Each cylinder assembly has an integral flow distributor that contains a dual concentric valve which provides the normal servo control and a jam tolerant function.
The flow distributor assembly is controlled through an input lever which receives the input from pilot's flight controls and through a feedback link which is connected to the output piston.

The anti-jamming device consists of a inner and outer sleeve that are held in place under the action of a spring. In case of a valve spool jam the force required at the input lever to operate the main valve exceeds a preset value thus compressing the spring and allowing the two sleeves to move relatively to control the hydraulic fluid within the cylinder chambers.
The actuator is designed to operate normally with two hydraulic systems (hydraulic system no.1 and hydraulic system no.2) which are completely separated within the actuator.

In case of failure of one system, the level of performances with only one hydraulic system operating ensures no degradation of handling qualities.

The indication system of the rotor flight controls is supplied by a pressure switch installed on each actuator control-valve.
If the control spool of an actuator control-valve jams or moves abnormally, the pressure switch will send an input to the indicating system which, in turn, will generate a caution and a 1 SERVO or 2 SERVO message depending on which system has had a jam.



MAIN ROTOR SERVO ACTUATORS

TAIL ROTOR CONTROLS SYSTEM – GENERAL

The tail rotor controls system includes the yaw control. The yaw control is obtained by changing the pitch angle of the tail rotor (TR) blades.

YAW CONTROL – MAIN COMPONENTS

PILOT PEDAL ASSY

The pilot pedal assembly is mounted on an articulated support that allows to fit the physical characteristics of the pilot.
The pilot pedal assembly is connected to the copilot pedal assembly to give a common input to the TR servo actuator.
The yaw pedal incorporates a microswitch used to operate the trim actuator friction in order to disengage the anchor point.

COPILOT PEDAL ASSY

The copilot pedal assembly is mounted on an articulated support that allows to fit the physical characteristics of the copilot.
The yaw pedal incorporates a microswitch used to operate the trim actuator friction in order to disengage the anchor point.

YAW TRIM ACTUATOR

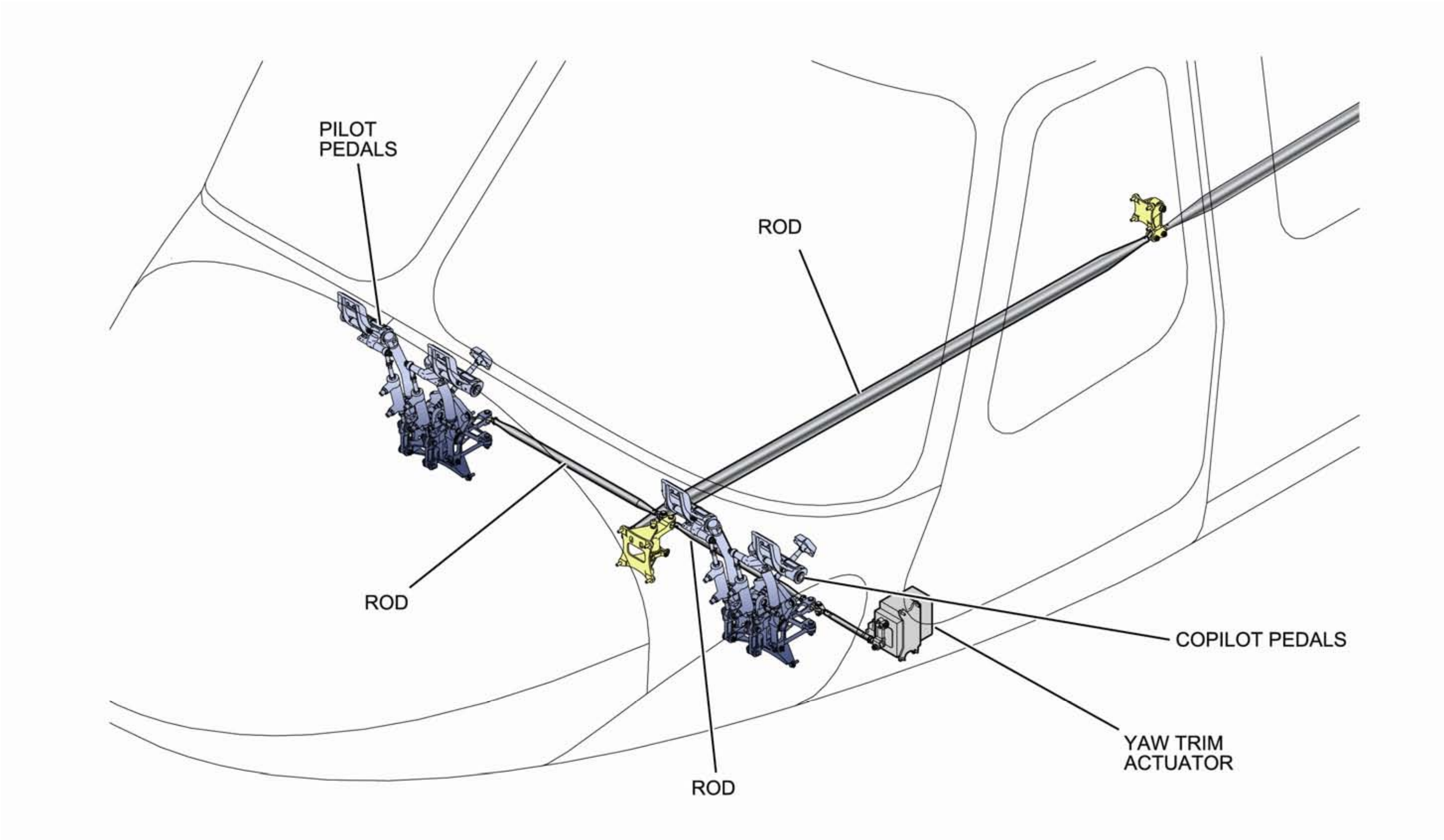
The yaw trim actuator receives electrical commands from the pilots or from the AFCS and convert these commands into a mechanical output used to change the position of the yaw control linkage.

YAW DUAL LINEAR ACTUATOR

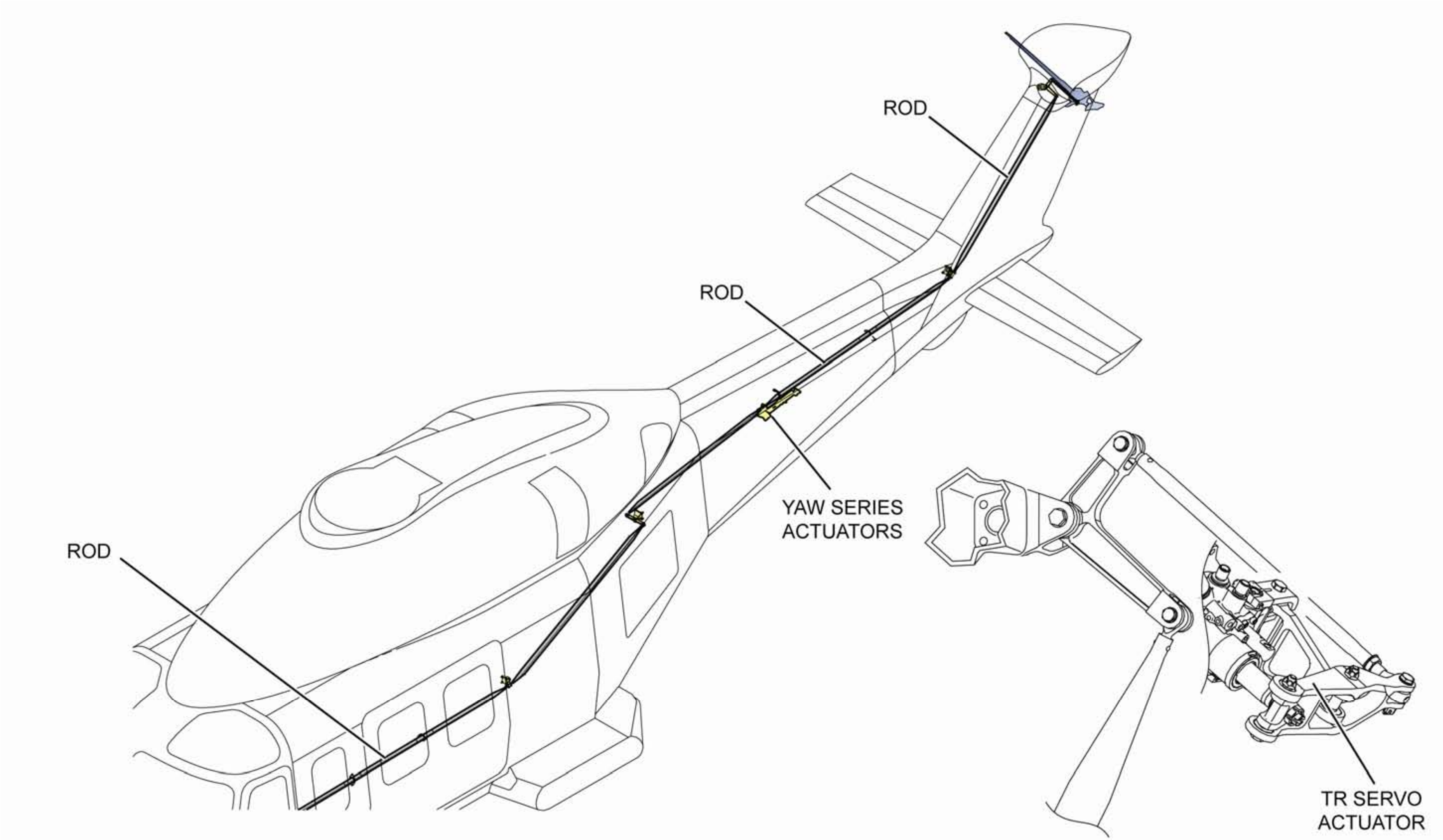
The yaw dual linear actuator is installed in series with the rods of the mechanical linkage inside the tail boom.
It is connected with the linkage through an anchor spring which provides a safety function in case of the actuator disconnects: in this case the spring reacts to the manual input as a pivot point to ensure the movement to the tail rotor servo actuator.

YAW CONTROL – PRINCIPLE OF OPERATION

The tail rotor control is operated by means of a mechanical linkage that gives an input to the dual-channel tail rotor hydraulic servo actuator. The TR servo actuator acts the pitch-change mechanism to give the required yaw action.
The mechanical linkage is operated by pilot and copilot pedals.
The mechanical linkage is also controlled by a yaw trim actuator.



YAW CONTROL – MAIN COMPONENTS (1 OF 2)



YAW CONTROL – MAIN COMPONENTS (2 OF 2)

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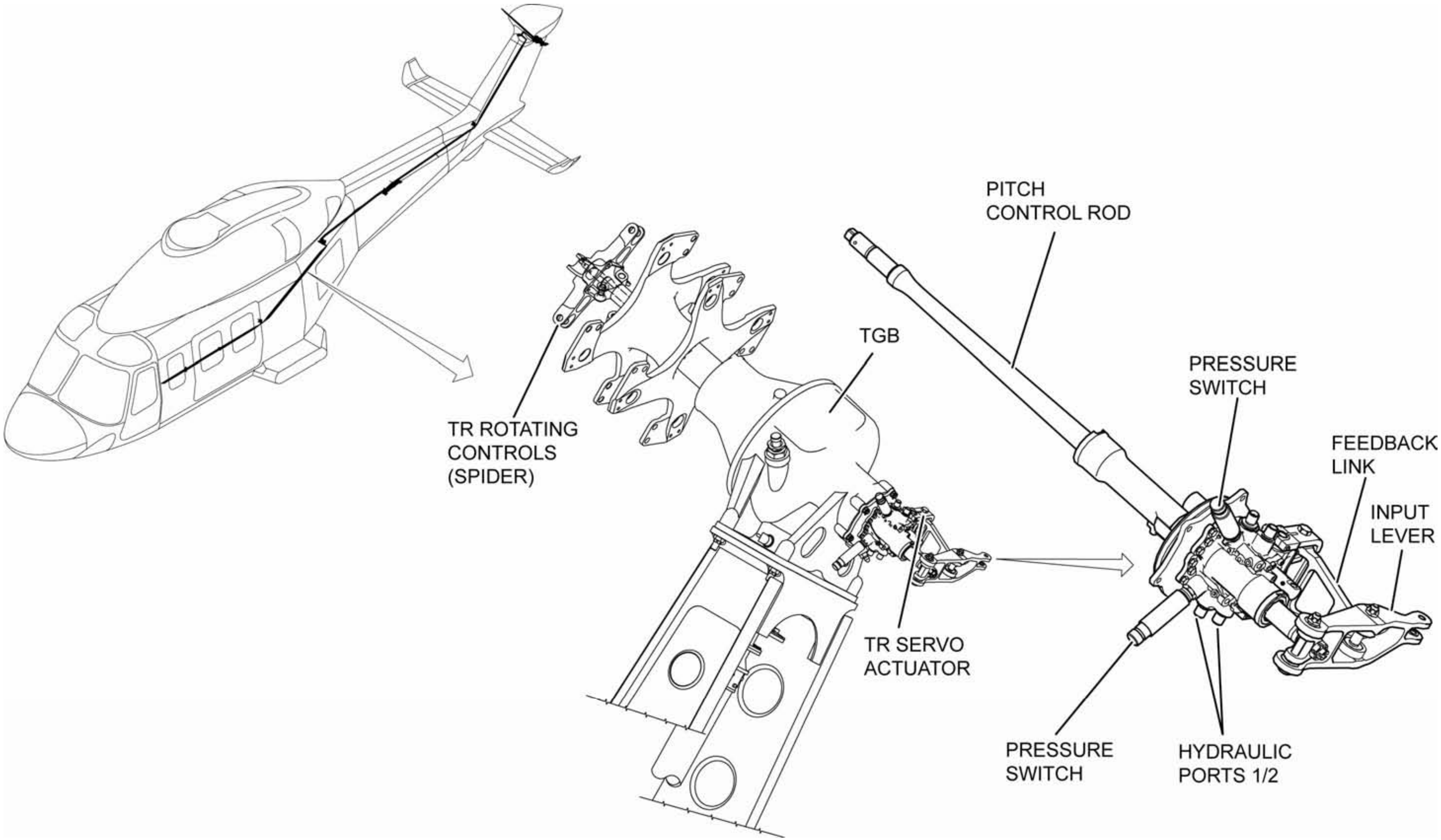
TAIL ROTOR SERVO ACTUATOR - GENERAL

The tail rotor servo actuator is a fixed body actuator comprising two separate cylinder assemblies, which are bolted together at the actuator center and tandem pistons.

TR SERVO ACTUATOR – PRINCIPLE OF OPERATION

The actuator body is provided with a mounting flange for the installation to the left side of the TGB. One piston end extends with a control rod which is connected to the rotating control spider; the other piston end is connected with the input lever.
Each cylinder assembly has an integral flow distributor that contains a dual concentric valve which provides the normal servo control and a jam tolerant function.
The flow distributor assembly is controlled through an input lever which receives the input from pilot's flight controls and through a feedback link which is connected to the input lever and hence to the output piston.
The anti-jamming device consists of a inner and outer sleeve that are held in place under the action of a spring.
In case of a valve spool jam the force required at the input lever to operate the main valve exceeds a preset value thus compressing the spring and allowing the two sleeves to move relatively to control the hydraulic fluid within the cylinder chambers.
The actuator is designed to operate normally with two hydraulic systems which are completely separated within the actuator. In case of failure of one system, the level of performances with only one hydraulic system operating ensures no degradation of the helicopter handling qualities.



TAIL ROTOR SERVOACTUATOR

ROTOR FLIGHT CONTROLS – CONTROLS AND INDICATORS

1. FORCE TRIM switch

OFF ……..…. the cyclic force trim actuators are disengaged

ON ……………..….. the cyclic force trim actuators are engaged. When the pilot moves the cyclic stick out of the detent position in the pitch and the roll axis, he feels the spring force applied on the cyclic flight control

2. CLTV / YAW TRIM switch

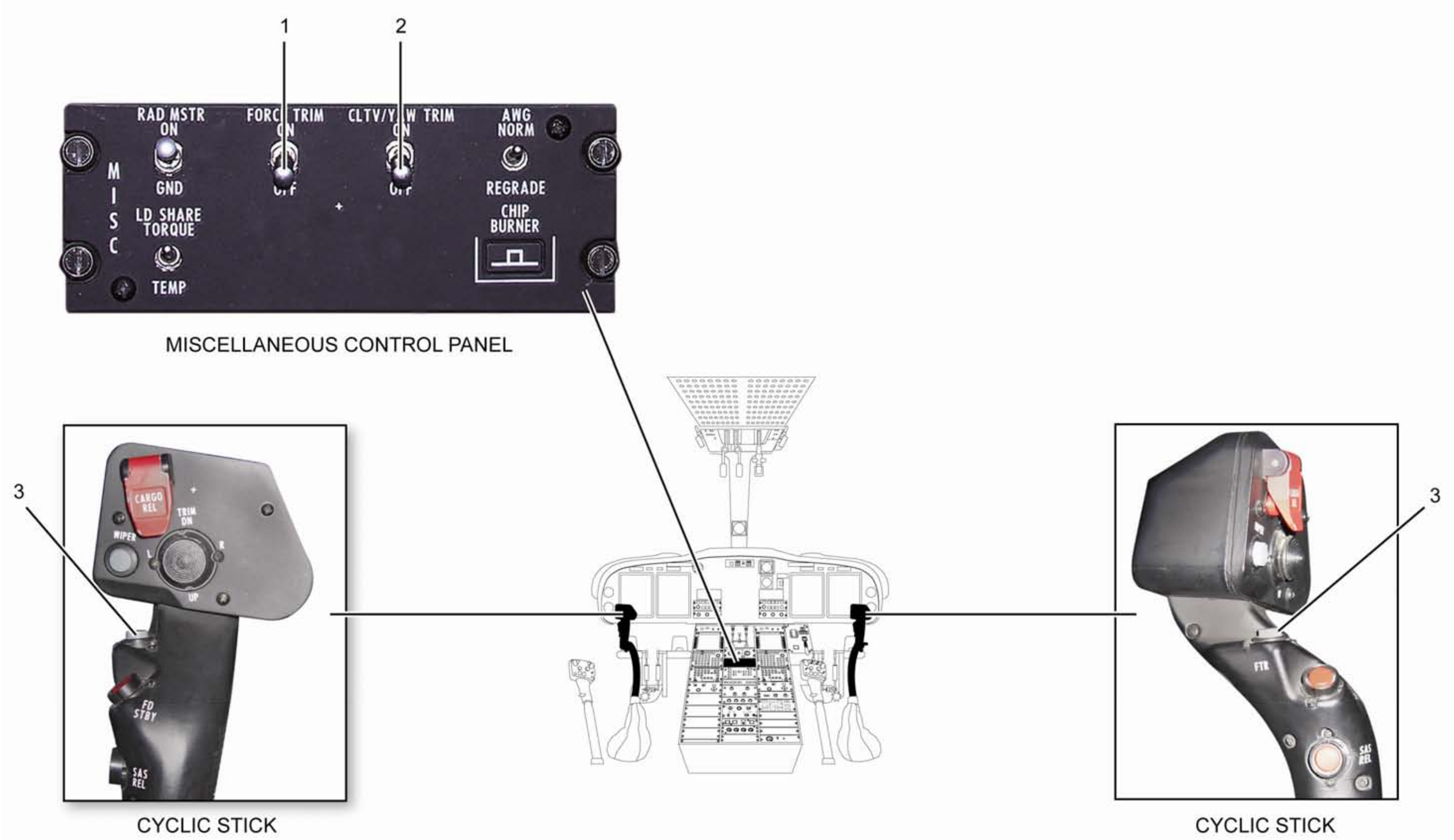
OFF ………..……... the collective and yaw pedals force trim actuators are disengaged

ON …………….….. the collective and yaw pedals force trim actuators are engaged. When the pilot moves the collective lever and the yaw pedals from their detent positions, he feels the spring force applied on the collective and yaw pedals flight controls

3. FTR push-button switch

PRESSED ...……... disengages the clutch of the cyclic trim actuator. When the pilot operate the cyclic stick, he feels the flight controls free to move and the spring force is not applied

RELEASED .... engages the clutch of the cyclic trim actuators. The pilot feels the spring force applied to the cyclic stick



ROTOR FLIGHT CONTROLS – CONTROLS AND INDICATORS (1 OF 2)

4. FTR push-button switch (CLTV PLT)

PRESSED ………. disengages the clutch of the collective and yaw pedals trim actuators.

NOTE. When the pilot operate the collective lever and the yaw pedals, he feels the flight controls free to move and the spring force is not applied.

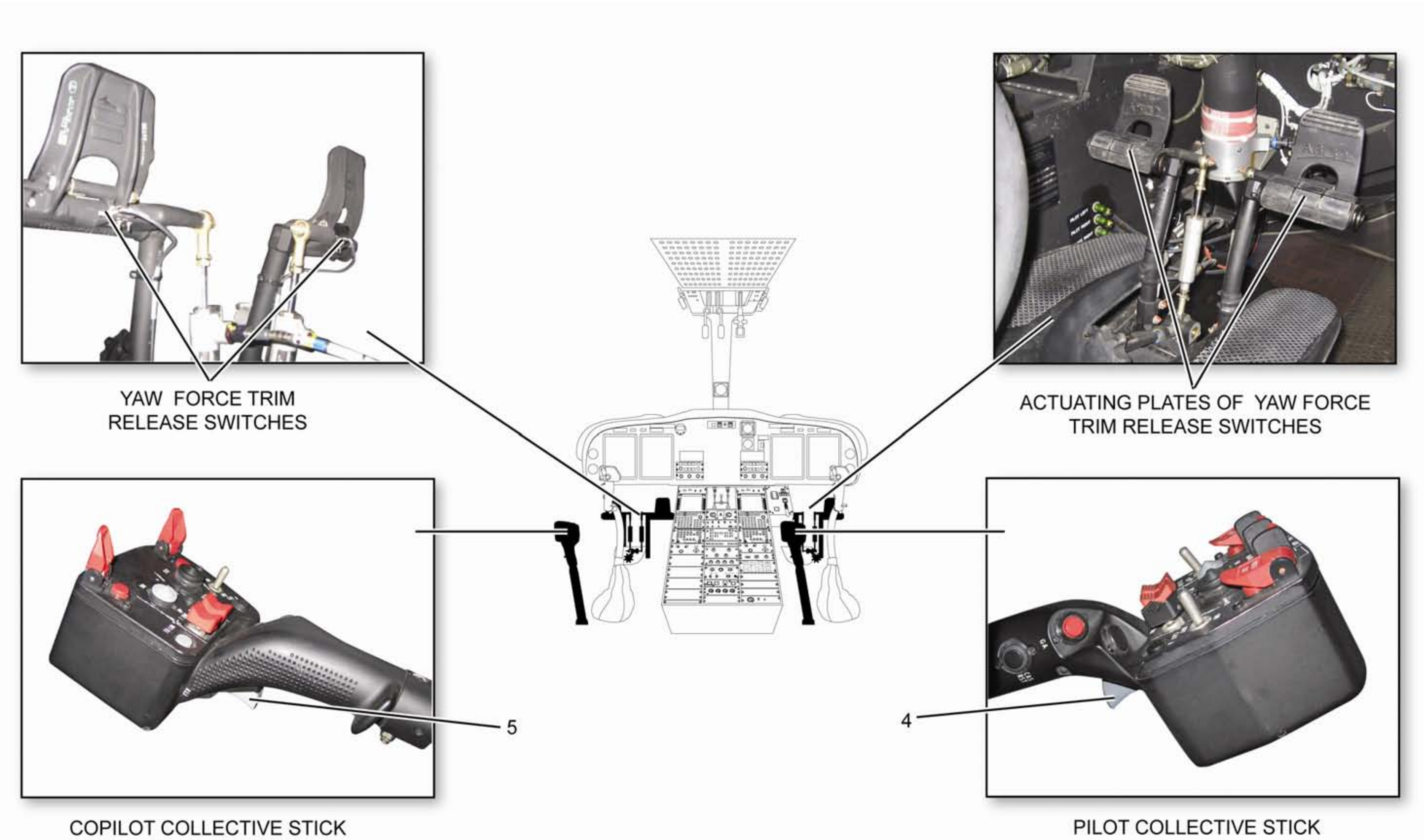
RELEASED …….. engages the clutch of the collective trim actuator and yaw pedals trim actuators. The pilot feels the spring force applied to the collective lever and yaw pedals

5. FTR push-button switch (CLTV CPLT)

PRESSED ….. disengages the clutch of the collective and yaw pedals trim actuators.

NOTE. When the copilot operate the collective lever and the yaw pedals, he feels the flight controls free to move and the spring force is not applied.

RELEASED … engages the clutch of the collective trim actuator and yaw pedals trim actuators. The copilot feels the spring force applied to the collective lever and yaw pedals.



ROTOR FLIGHT CONTROLS – CONTROLS AND INDICATORS (2 OF 2)

TRIM ACTUATOR - GENERAL

The trim actuator includes

- a feel spring which provides a feed-back feel load to the pilot when actuated
- a magnetic friction (clutch) which provide an anchor point of the control and the connection/disconnection to the feel spring
- a dual position sensor which provide a feed-back position signal of the control to the AFCS
- a detent microswitch used to inhibit AFCS control in case of pilot input
- an electrical motor which convert the AFCS electrical signals to mechanical output on the control
- a damper which reduces vibration on the control due to engagement / disengagement of the anchor point using the feel spring
- the output shaft is provided with a shear section

TRIM ACTUATOR – PRINCIPLE OF OPERATION

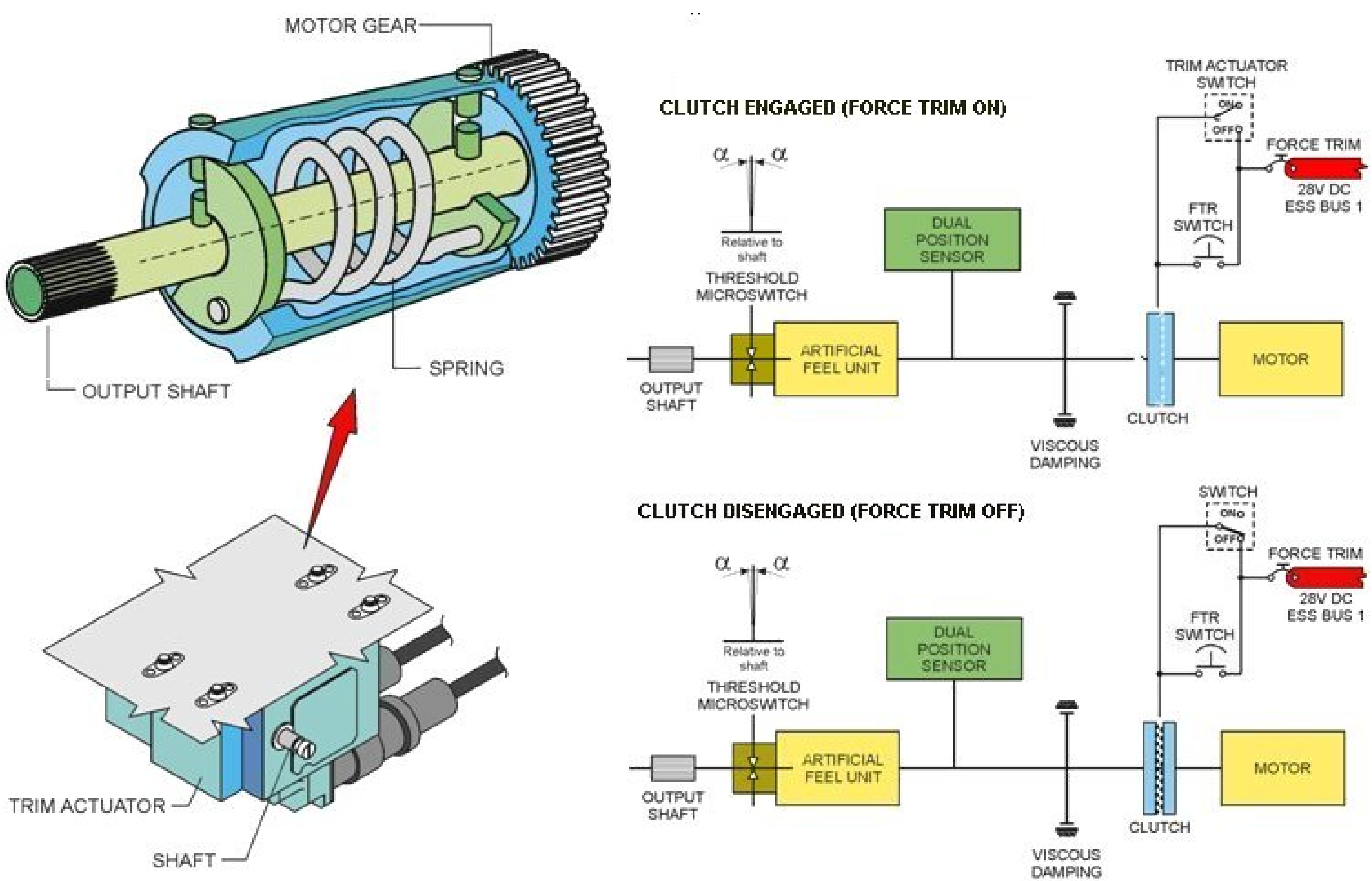
The pilot can set the force feel system ON and OFF with the FTR switch that engage or disengage the clutch in the trim mechanism.

When the clutch is disengaged then

- the force feel system is disengaged
- the actuator drive is disengaged
- the auto-trim cannot send commands to the controls

The clutch is disengaged when one of the following conditions is satisfied

- the 28 V DC power supplied by the trim actuator switch is set to OFF
- the 28 V DC power supplied by the trim actuator switch is set to ON and the FTR switch is pressed



TRIM ACTUATOR – PRINCIPLE OF OPERATION

TRIMMING

COLLECTIVE TRIMMING

The "push and hold" collective lever trim release button (FTR) is depressed to disengage the spring feel mechanism which enables the collective to be moved freely. When the button is released, the spring feel is re-datum to zero force.

CYCLIC TRIMMING

The force trim release button (FTR) on the cyclic stick should be kept depressed during all large stick movements. Upon releasing the trim release button, the attitude hold is restored.
For small attitude adjustments in the hover or in forward flight (± 2 to 3 knots), the system beep trim mode (TRIM) can be used. Operation of the cyclic beep trim switch causes the trim system to change the reference at 3 degrees per second in roll and 2°/sec in pitch or 1°/sec in pitch for airspeed above 120 KIAS. When the new desired attitude is reached, the switch is released.
The trim method most commonly used is a combination of trim release and beep trim.

YAW TRIMMING

Lateral operation of the collective lever CLTV/YAW 4-way trim switch alters the slip or skid command to either offset a small accelerometer misalignment, or to purposely offset the tail alignment.
Additionally, at speeds above 40-45 KIAS the coordinated turn facility enables the aircraft to carry out a balanced turns, below 60 KIAS the Low Speed Heading hold is active. Feet should not be rested on the pedals if heading hold or turn coordination facilities are required.

When used for the trimming of the cyclic and pedal controls, the actuators are controlled to keep the linear-series-actuators output-shaft at center. This is necessary to have the best response to the control inputs from the high-performance linear-actuators

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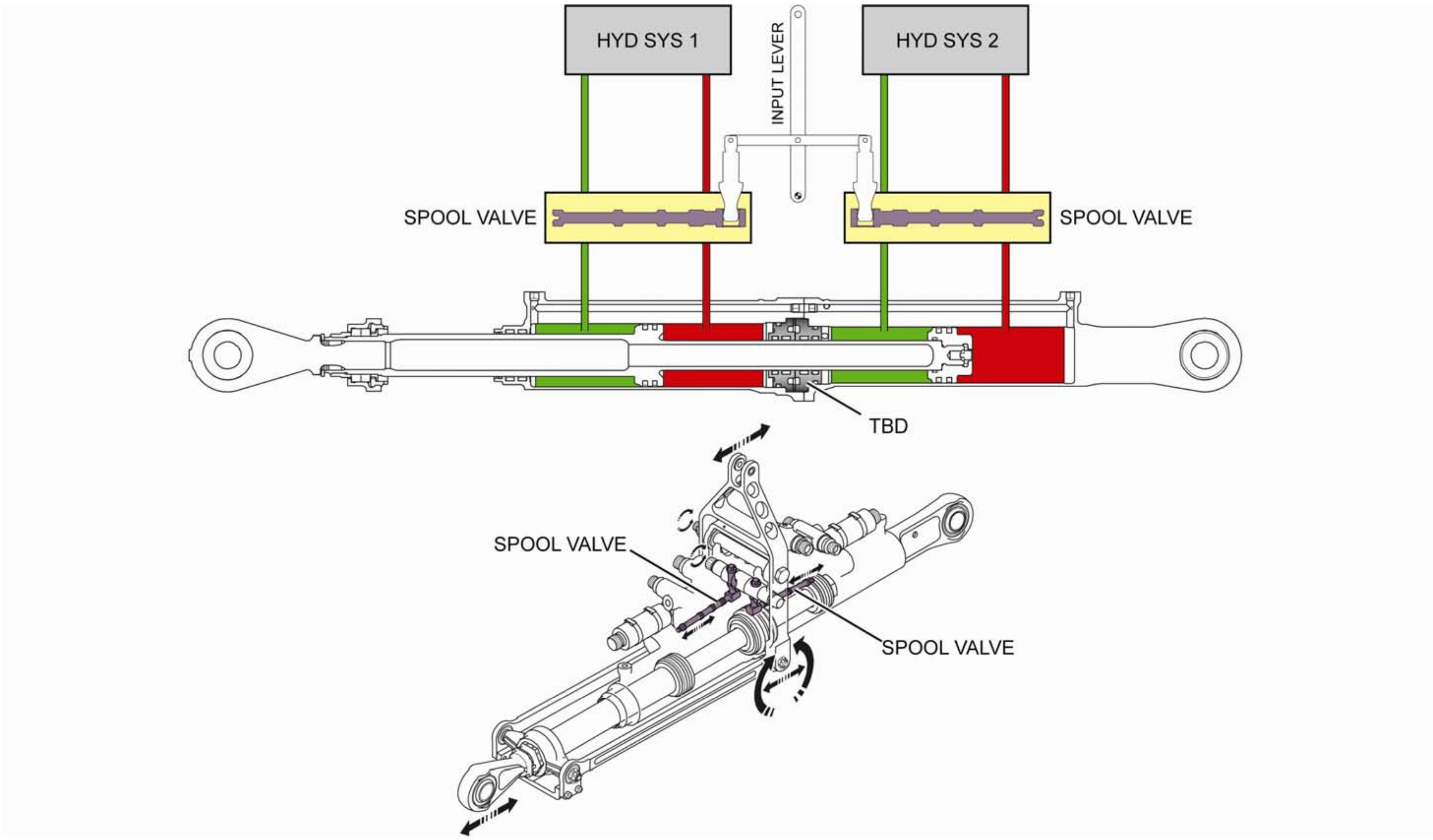
SERVO ACTUATORS – PRINCIPLE OF OPERATION

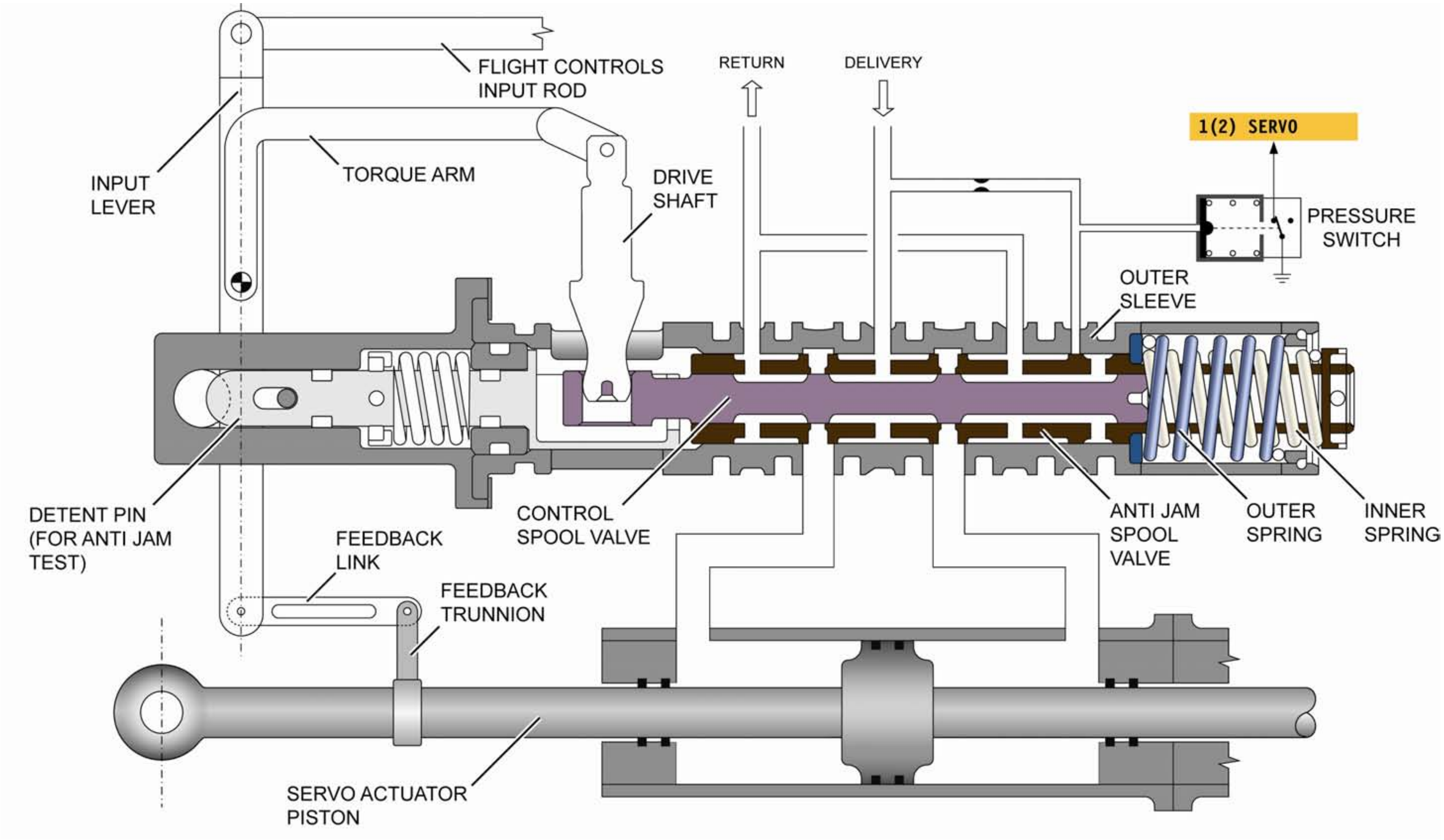
The following schemes explain the normal and the emergency operation that can occur into any one of the two MR servo actuators cylinders.

NORMAL OPERATION

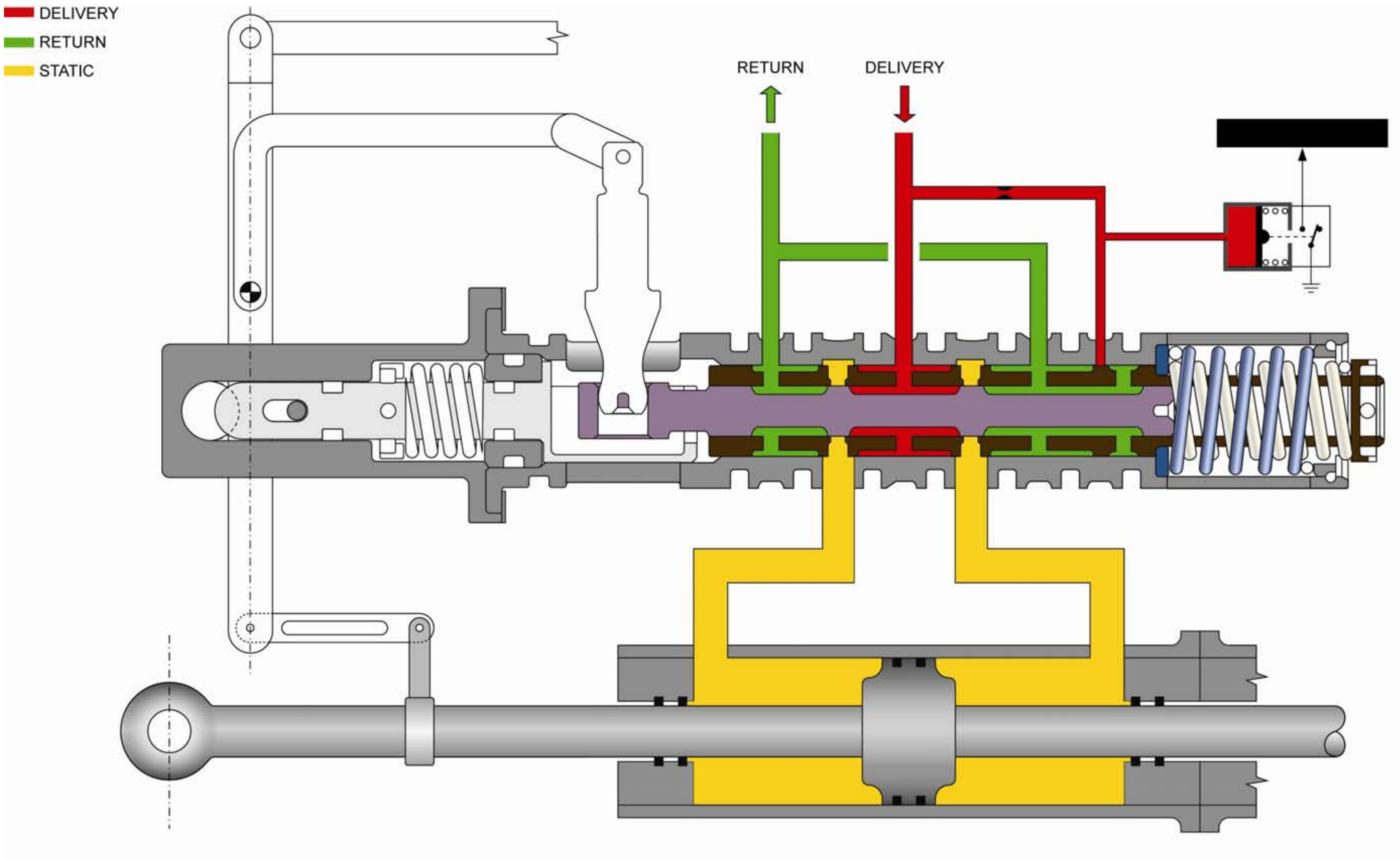
- the input lever transmits the pilot's input through a linkage and lay shaft to the flow distributor assembly in each half of the actuator
- displacement of the input lever results in movement of the servo valves from the neutral position routing the high pressure hydraulic fluid into the appropriate cylinder chamber
- the resulting piston motion drags a dual load path feedback link to rotate the input lever and reset the valve command
- movement of the input lever results in the piston moving in the opposite direction



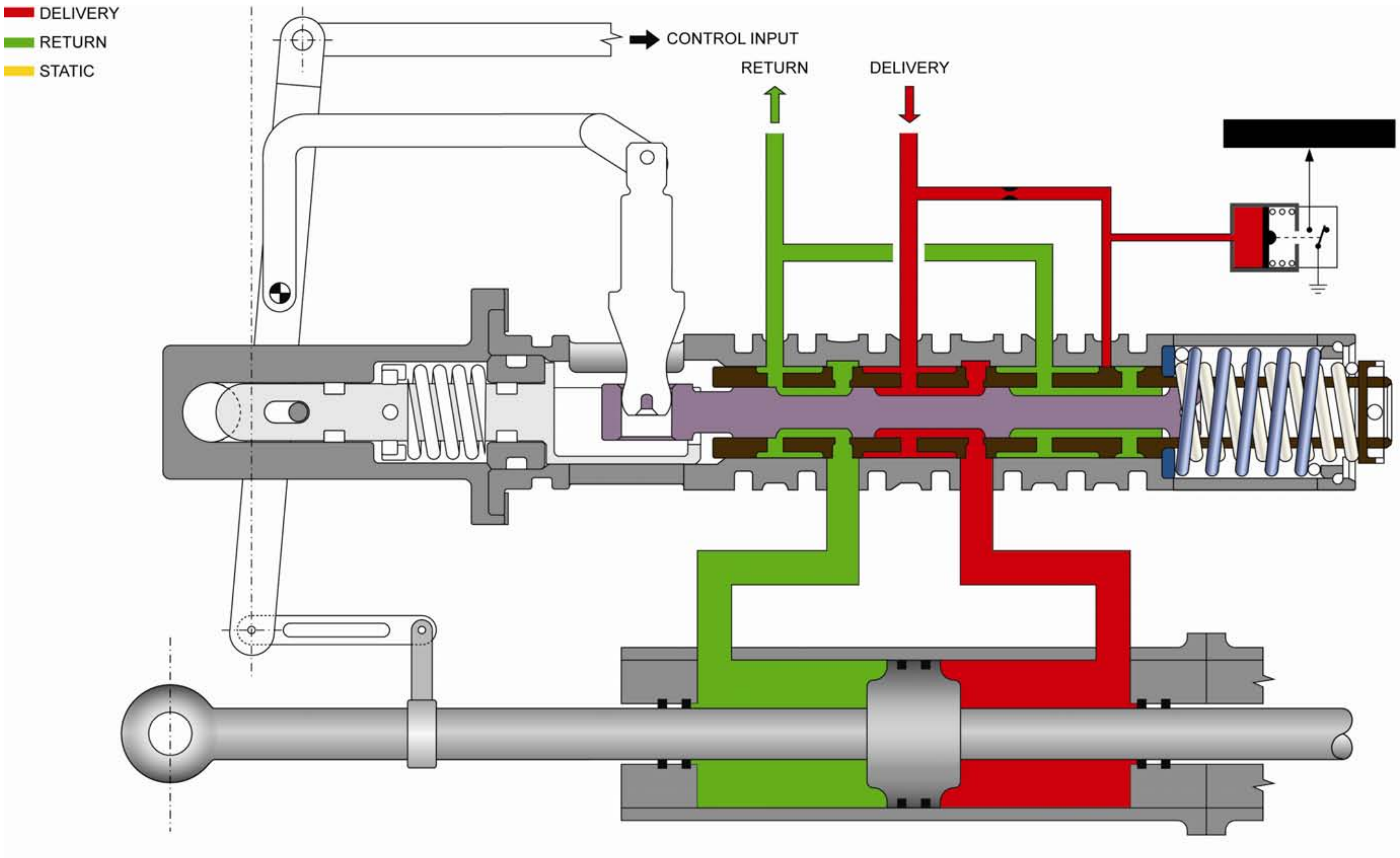
SERVO ACTUATOR SCHEMATIC



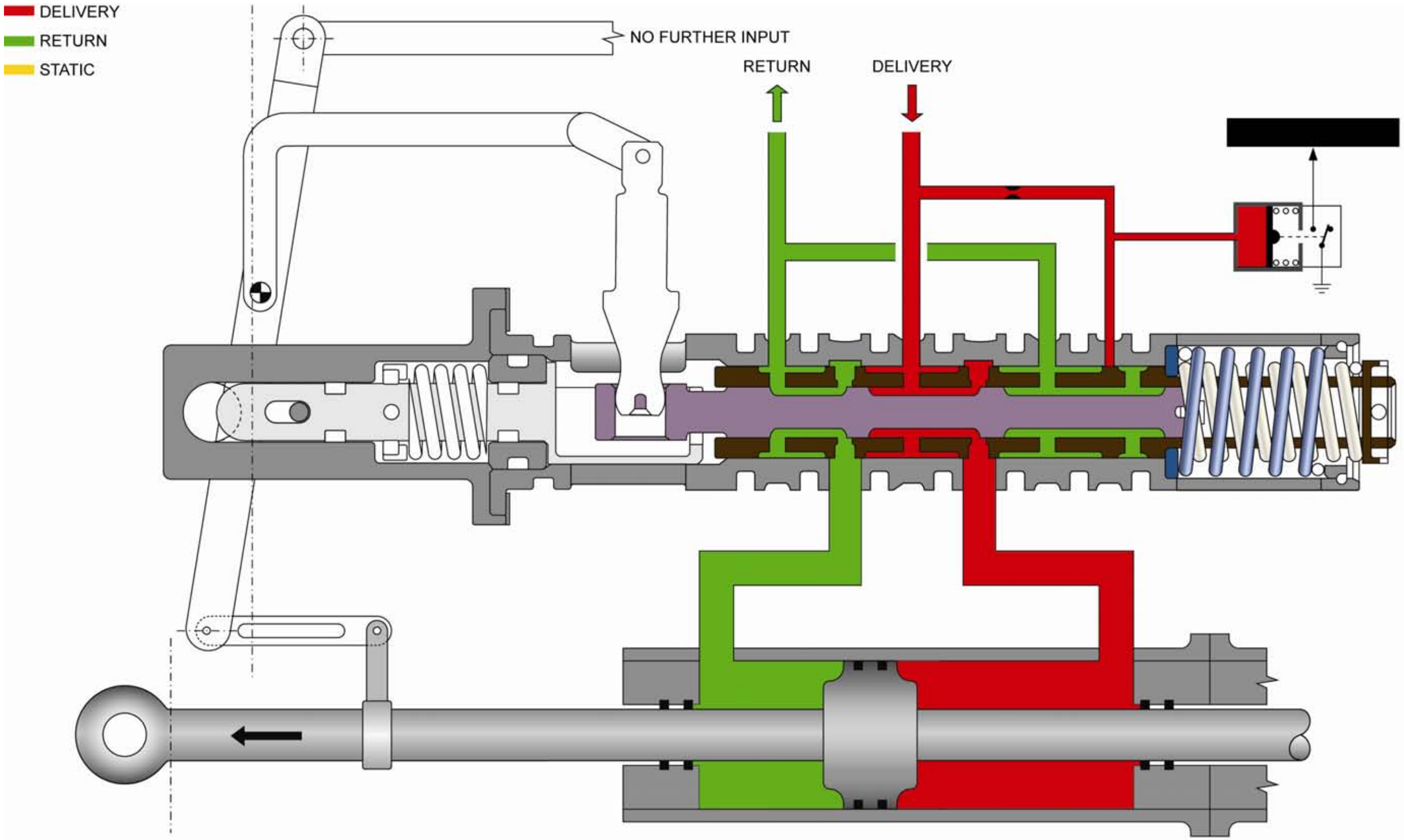
SERVO ACTUATOR – SCHEMATIC



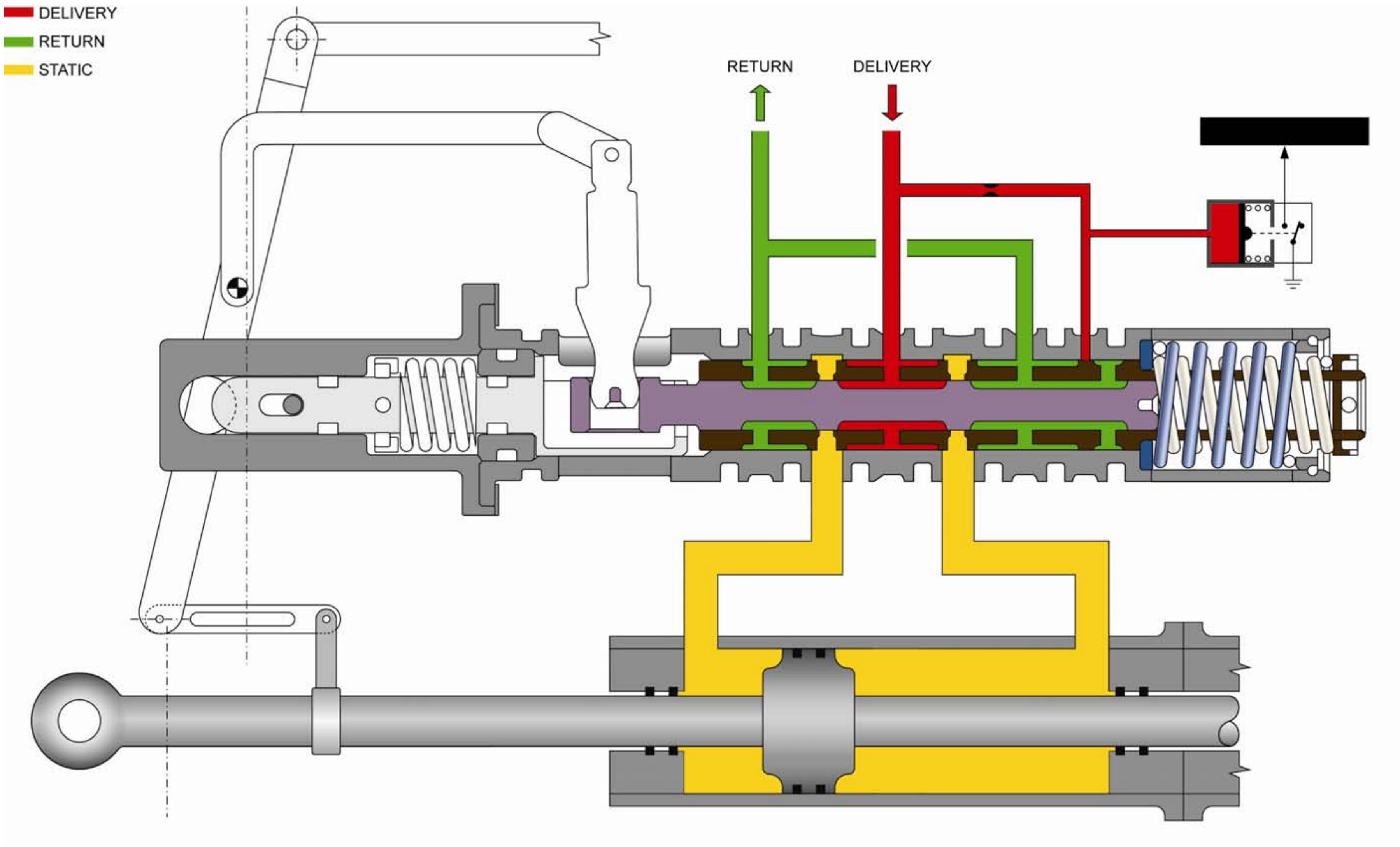
CONTROLS STATIONARY AND CENTERED



CONTROL INPUT



CONTROLS STATIONARY IN NEW POSITION - ACTUATOR MOVING



CONTROL SPOOL VALVE RESET - ACTUATOR IN NEW POSITION

EMERGENCY OPERATION - HYDRAULIC FAILURE

In this case the servo valve ports inside the failed half of the actuator (pressure loss) are controlled in the normal way allowing hydraulic fluid to be forced from the dead cylinder by the active half.

This condition is monitored on the CAS window (in addition to pressure indication) by the microswitch installed on the flow distributor which provides the caution message 1(2) SERVO.

EMERGENCY OPERATION – VALVE JAMMED

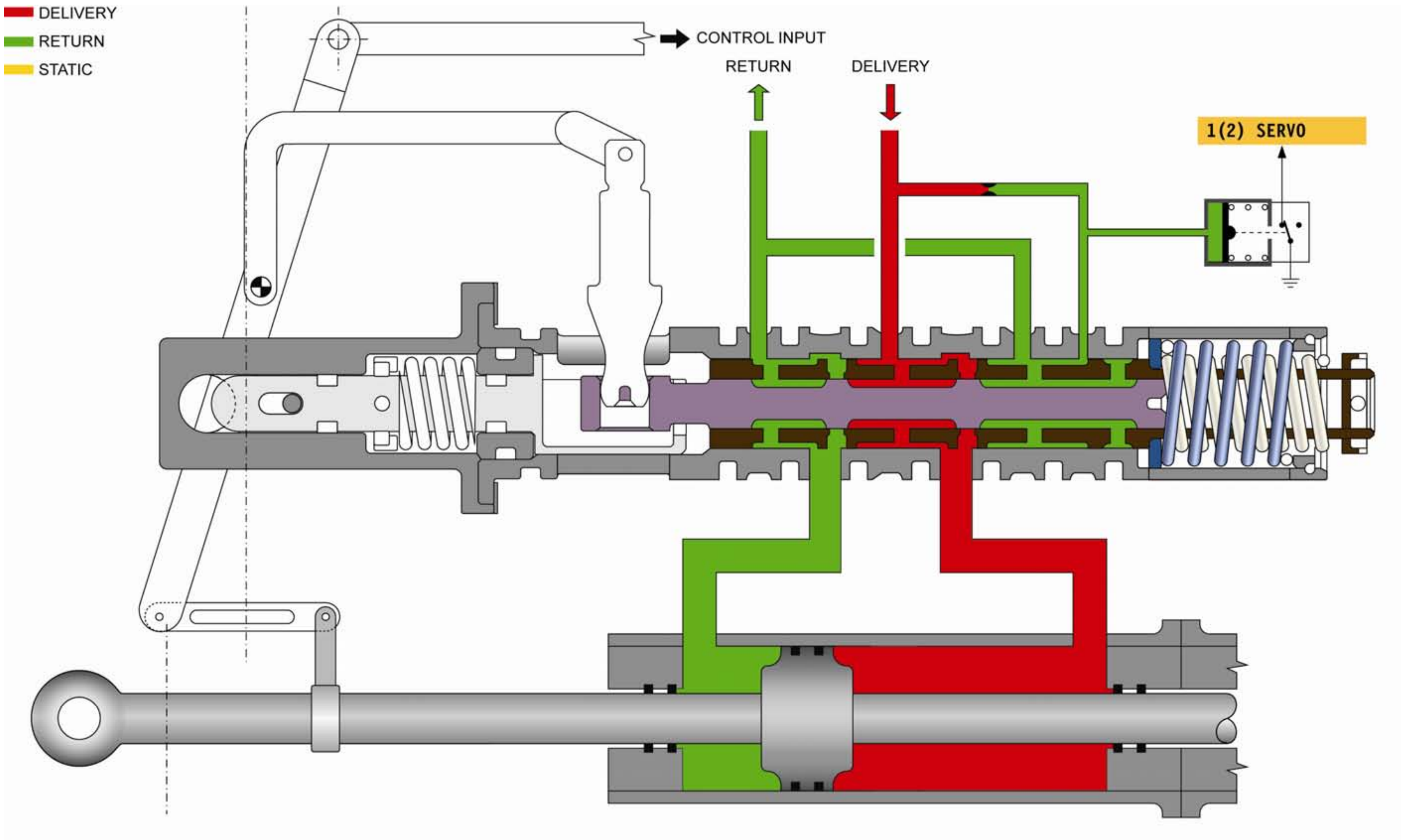
Depending on the main spool valve jammed position, the dual concentric sleeve valve allows two different way of operations

- if the main spool valve is jammed in the center position (ports in the inner sleeve closed), the sleeve moves to distribute the fluid to and from the cylinder in the normal way in response to input commands. This condition is defined as “actuator active”
- if the spool valve is jammed away from the center position, the sleeve moves to connect together both control ports, to and from the cylinder, to the return line. This is a bypass condition and the piston is moved by the remaining active half of the actuator

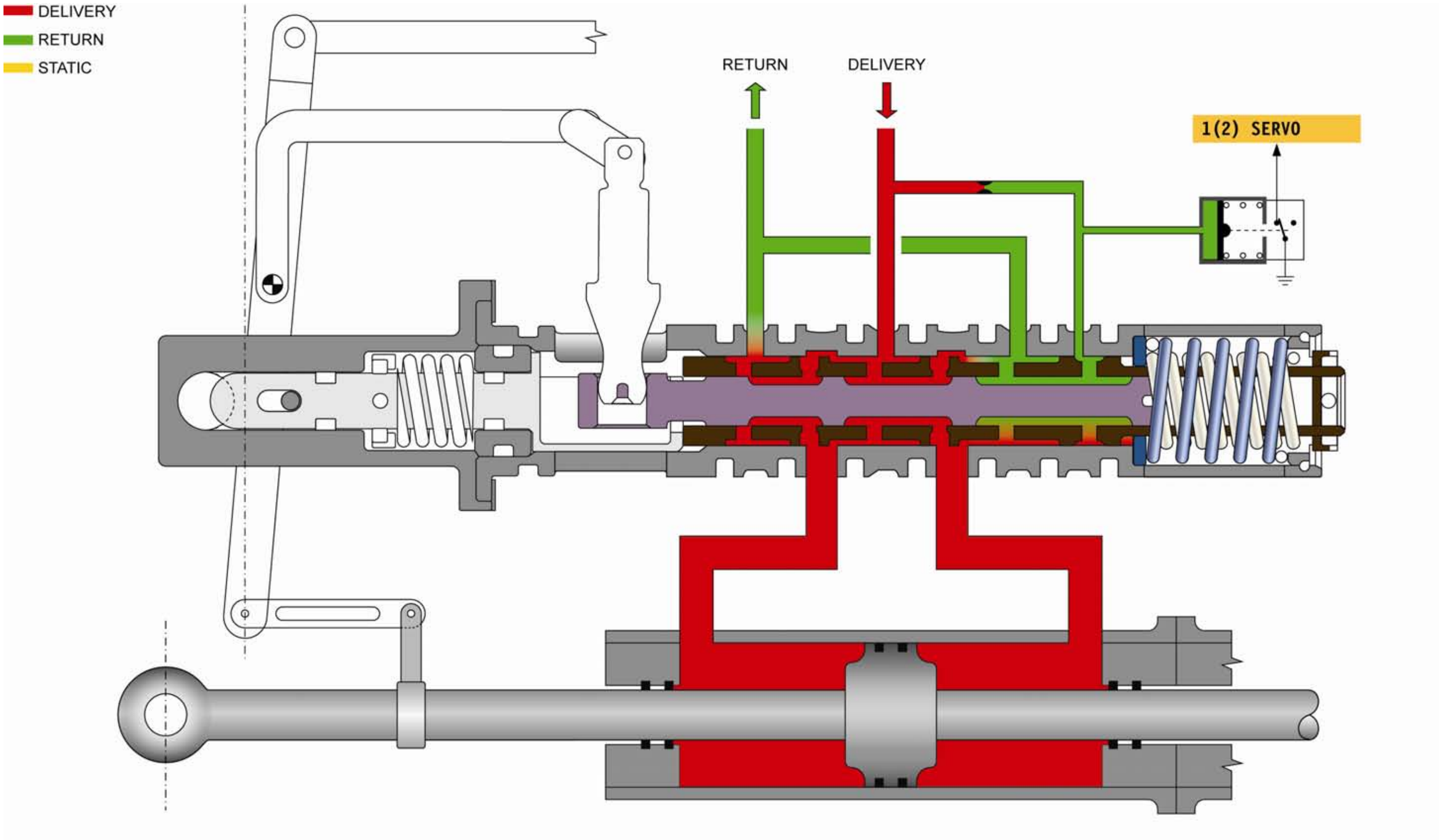
In the first condition the pressure line connected to the microswitch is drained to return only if flight controls are operated so that the caution 1(2) SERVO is provided to the MFD.

In the second condition the pressure line connected to the microswitch is always drained to return so that the caution 1(2) SERVO is always provided to the MFD.

A maintenance check may be performed on the anti-jam device to ensure that the function is working correctly and that no dormant failures exist in the components.



EMERGENCY OPERATION – CONTROL SPOOL VALVE JAMMED IN CENTER



EMERGENCY OPERATION – CONTROL SPOOL VALVE JAMMED OUT OF CENTER

CAS CAUTION MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
1 (2) SERVO	Associated hydraulic servo actuator in bypass NOTE When the flight controls are operated and a main spool valve in any servo actuator is jammed in center or a main spool in any servo actuator is jammed out of the center regardless of flight controls operation (the caution is also provided when the pressure in the hydraulic circuit is below 138 bar). WARNING Do NOT switch SOV to CLOSE on the UNAFFECTED system since this will cause loss of control in the affected servo jack NOTE Loss of hydraulic fluid in system no.2 will automatically close the Tail Rotor Shut Off Valve (TRSOV). This will be indicated by a **2 SERVO** caution on the CAS and a TRSOV closed indication on the hydraulic synoptic page. Once the TRSOV has operated the SOV no.1 is inhibited.	MAIN VALVE SEIZURE IN MAIN OR TAIL ROTOR SERVO	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES HYDRAULIC SYSTEM

CHAPTER
71
POWER PLANT

SECTION 00 – GENERAL

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POWER PLANT – GENERAL

The AW139 is powered by two Pratt & Whitney Canada PT6C-67C turbine engines. The engines are installed in separate fire-proof compartments above the cabin roof and are attached to the helicopter airframe by two links located one inboard and one outboard on the rear side of the engine. The forward mount is attached to the Main Gearbox (MGB) input case.

Each engine is a free turbine turbo-shaft propulsion engine that includes a four stage axial compressor and a one stage centrifugal compressor. The compressors are driven by a single-stage compressor turbine.

The combustion chamber is a reverse flow combustion camber type where flow direction changed 180 degrees as it enters and mixes with the fuel. Metered fuel is sprayed into the combustion chamber through 14 fuel nozzles.

The power turbine turns the output shaft at the front of the engine. The output shaft supplies power to the input gears of the MGB.

A high voltage ignition unit and two spark igniters are used to start combustion. When combustion is self sustaining the ignition is turned off. An Electronic Engine Control (EEC) unit with an hydro-mechanical Fuel Management Module (FMM), ensures automatic control of the engine and fast response changes in power demand.

A manual back-up of the automatic engine control system is operated by the Engine Control Lever (ECL).

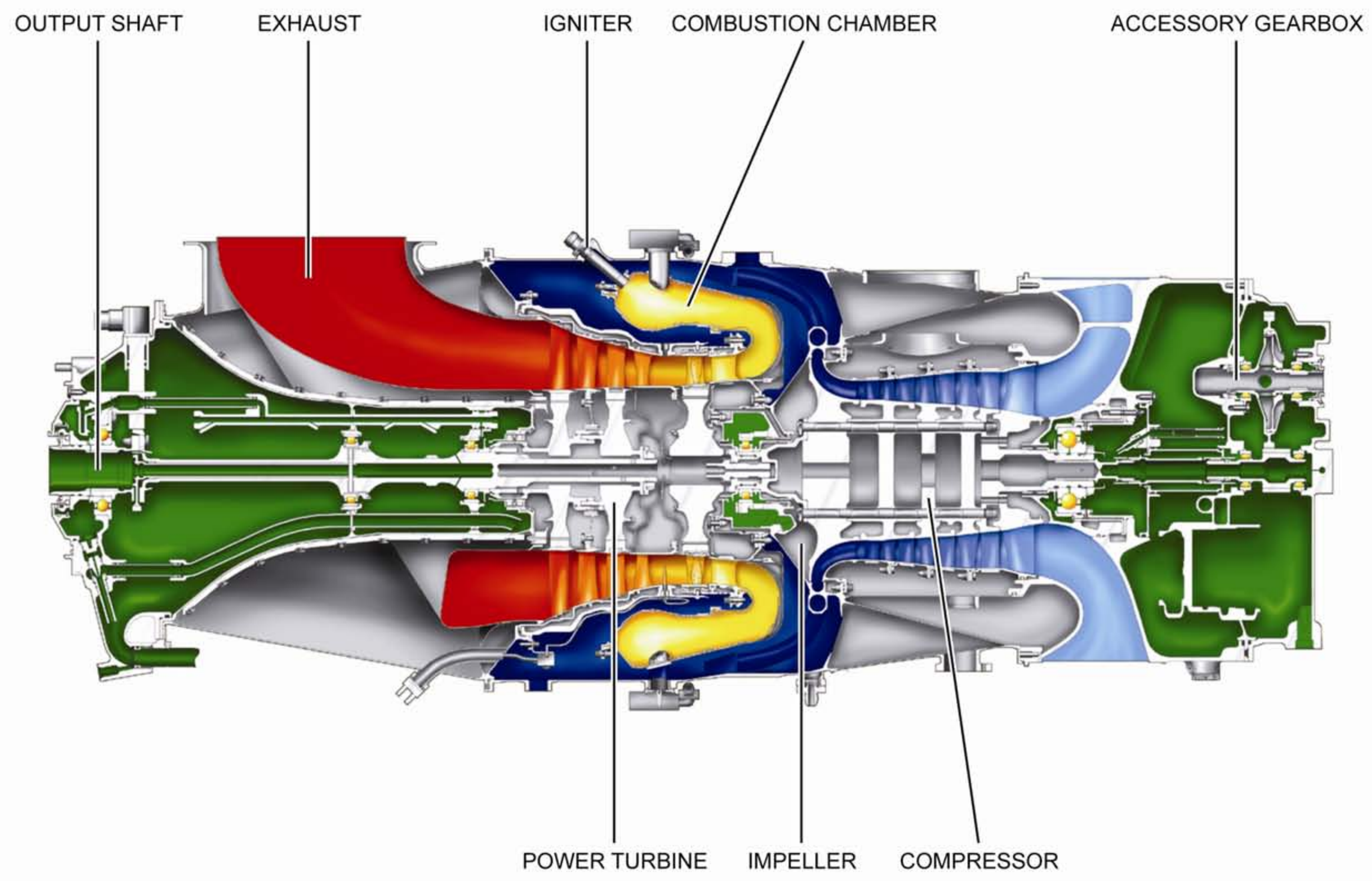
The engine lubrication system supplies a regulated and filtered oil flow to the engine. The oil system includes an oil tank, a pressure and a scavenge system and a breather system.

Fuel to the engines is supplied by the fuel system through the selector manifold, while air is supplied by individual air inlets. The exhaust gases are sent through an exhaust duct from the engine into the atmosphere.

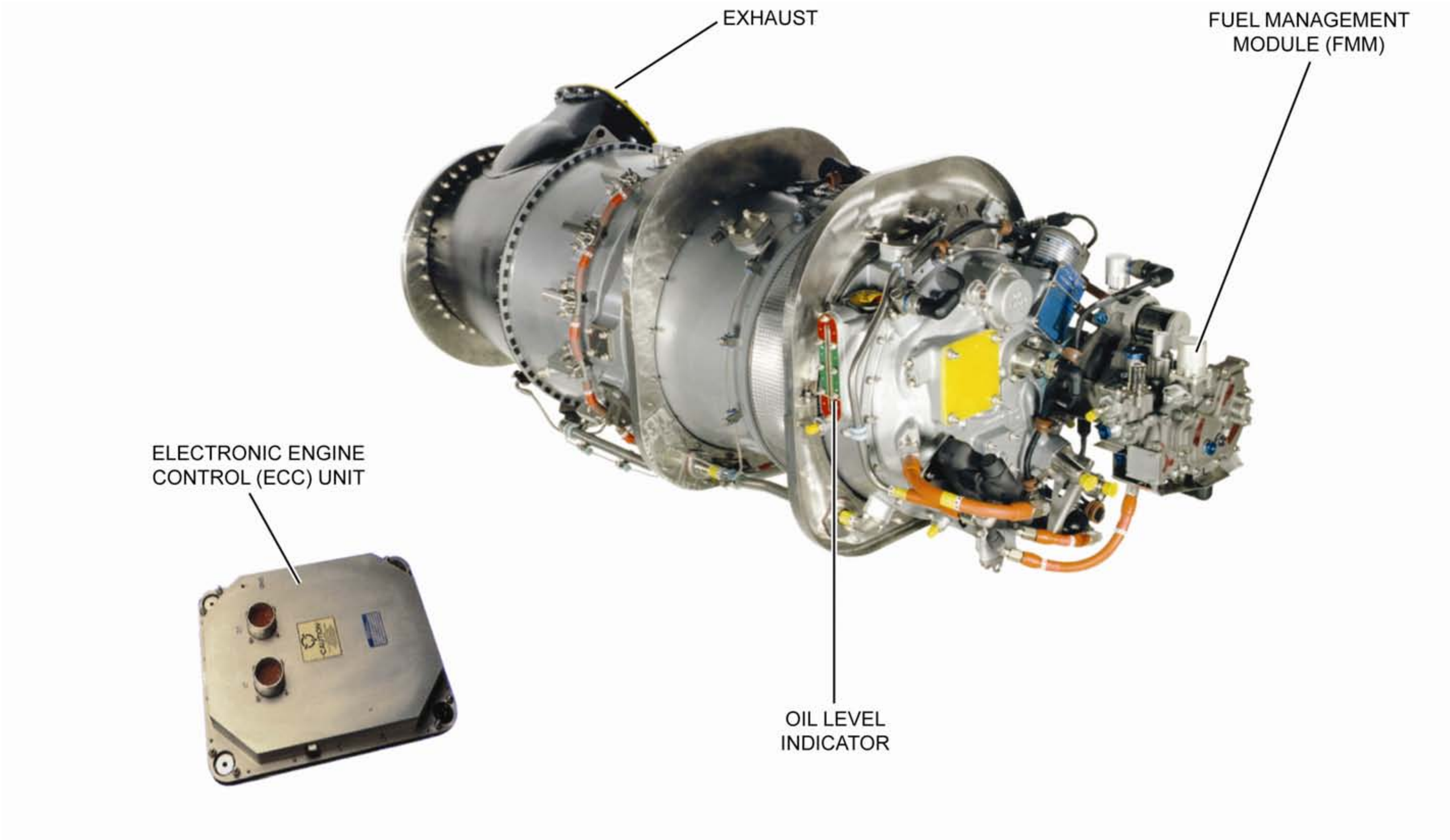
The engine indicating system includes sensors and probes installed on the engine that check the status of the engines. The signals are sent either directly to the cockpit instrument panel or via the EEC unit. A Data Collection Unit (DCU) stores data used for engine maintenance.

An Accessory Gear Box (AGB) supplies drive to the starter-generator, the Fuel Management Module (FMM), the Permanent Magnetic Alternator (PMA), the gear pump of the oil system.

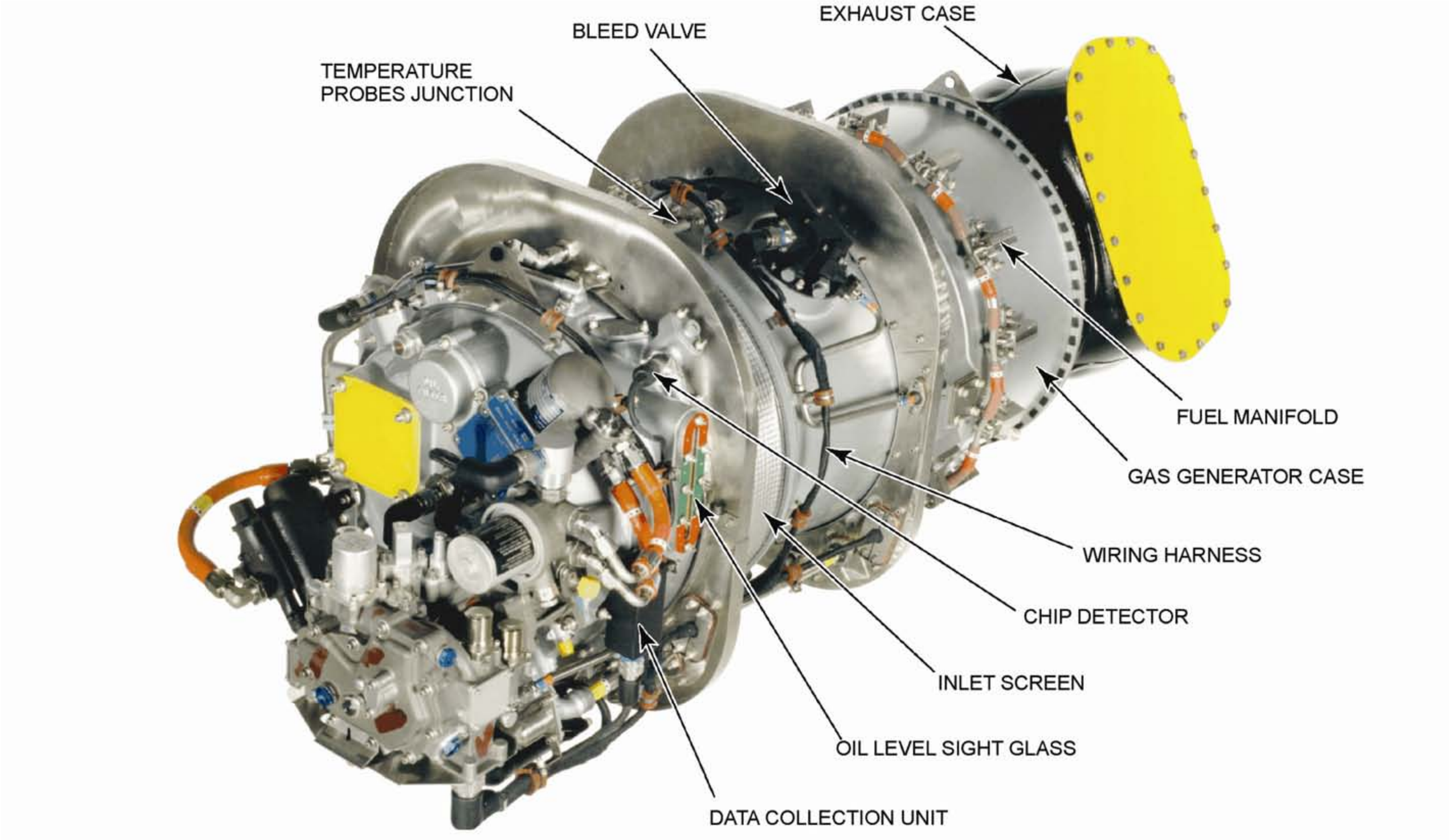
PRATT & WHITNEY PT6C-67C TURBINE ENGINE



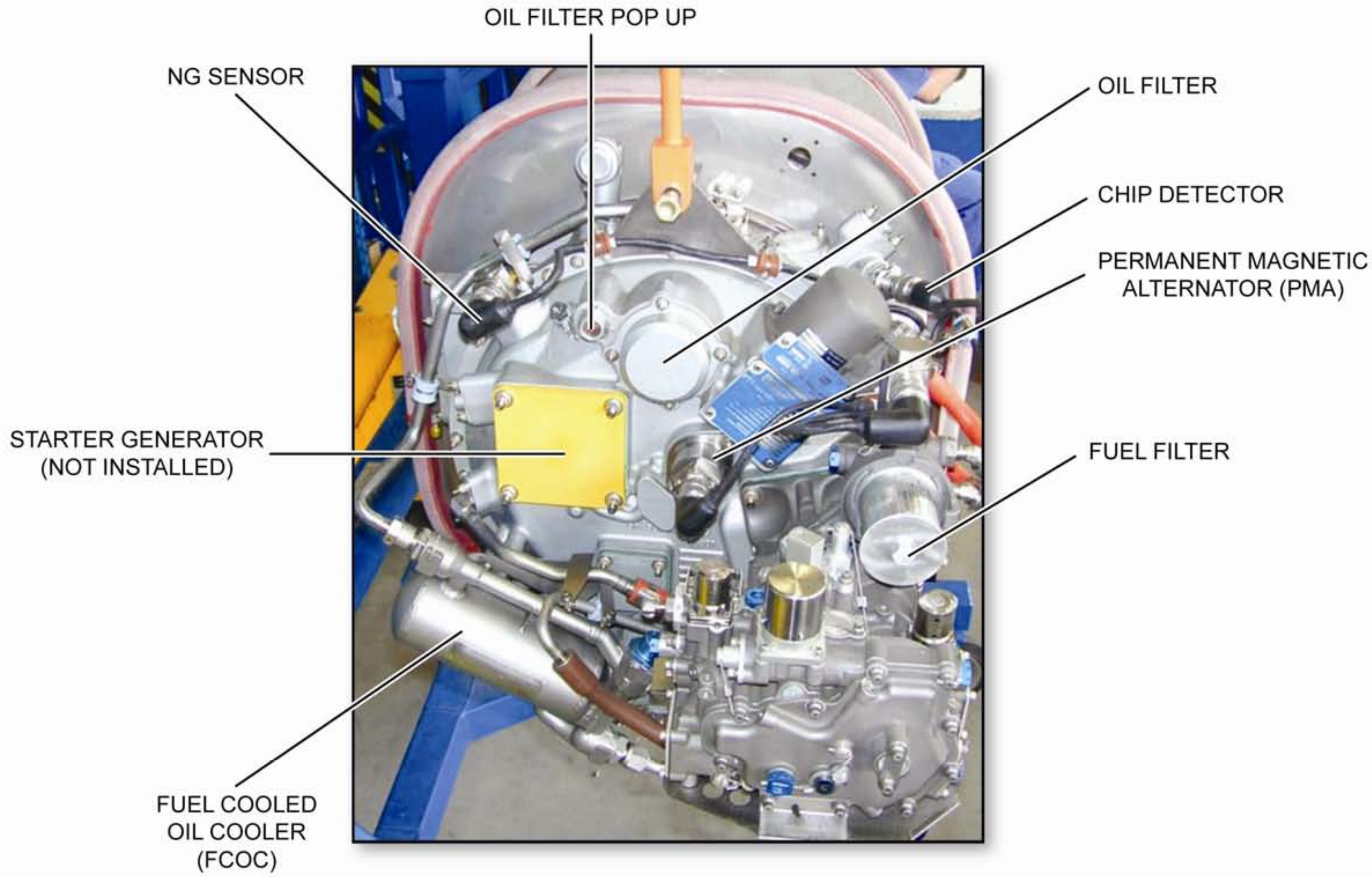
POWER PLANT SCHEMATIC



MAIN COMPONENTS (1 OF 3)



ENGINE COMPONENTS (2 OF 3)



ENGINE COMPONENTS (2 OF 3)

POWER PLANT - MAIN FEATURES

CONSTRUCTION
- Non-modular free-turbine engine

COMPRESSOR
- 4 axial stages plus 1 centrifugal impeller
- Compressor bleed valve (pneumatic and electronically controlled)
- Jet flap inlet configuration

COMBUSTION CHAMBER
- Annular / Reverse flow

COMPRESSOR TURBINE
- Single stage (CCW rotation)

POWER TURBINE
- Free turbine / 2 stage turbines (CW rotation)

EXHAUST
- PT6C-67C: 60° up LEFT or RIGHT configuration

OUTPUT SECTION
- Direct drive to the aircraft transmission

ENGINE OIL SYSTEM
- Integral oil tank
- Regulated oil pressure system
- Engine mounted oil cooling system
- Sight glasses for oil level check
- Chip detection
- Oil pressure and temperature sensors

FUEL & CONTROL SYSTEM
- Electronic Engine Control (EEC)
- Permanent Magnetic Alternator (PMA)
- Fuel Management Module (FMM) with integral fuel pump
- Electronic torque measuring system (torque shaft)
- NG/N1 speed sensor
- NPT/NF/N2 speed and torque sensors
- Interturbine gas temperature system (ITT, MGT, T5)
- Data Collection Unit (DCU)

IGNITION SYSTEM
- High energy
- 1 exciter box
- 2 igniters
- 2 cables

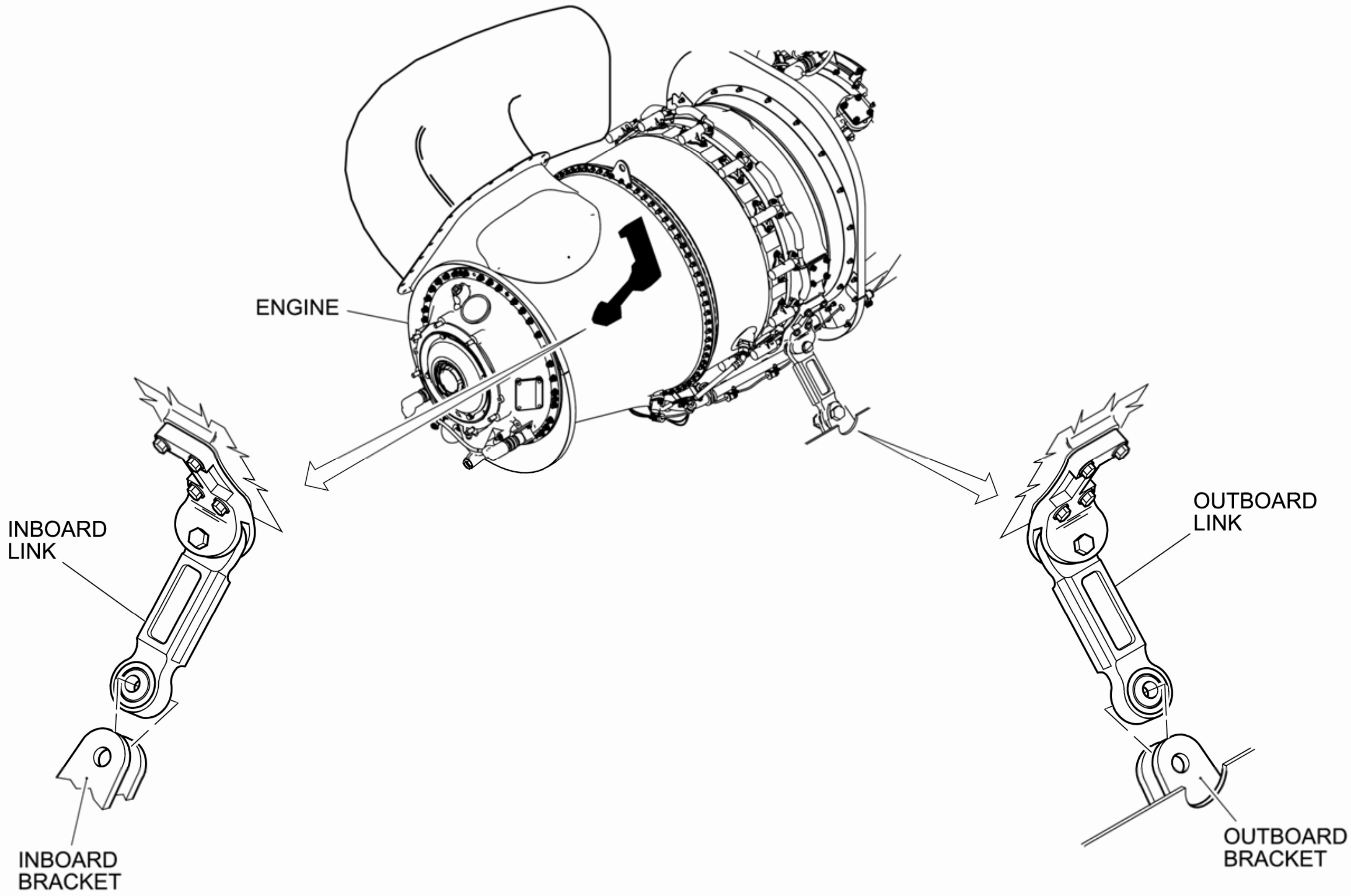
ACCESSORY GEARBOX
- Driven by the compressor
- Provide drives for engine and aircraft accessories

ENGINE MOUNTS

Engine mounts connect each engine to the aircraft structure on three points, one on the forward side and two on the rear side of the engine.

The forward mount is a gimbal installation attached to the Main Gearbox input case and carries vertical, horizontal and torsional loads.

The two rear mounts are composed of a link on each engine side (inboard and outboard) attached to a support on the engine bay floor. The mounts allow engine thermal radial expansion.



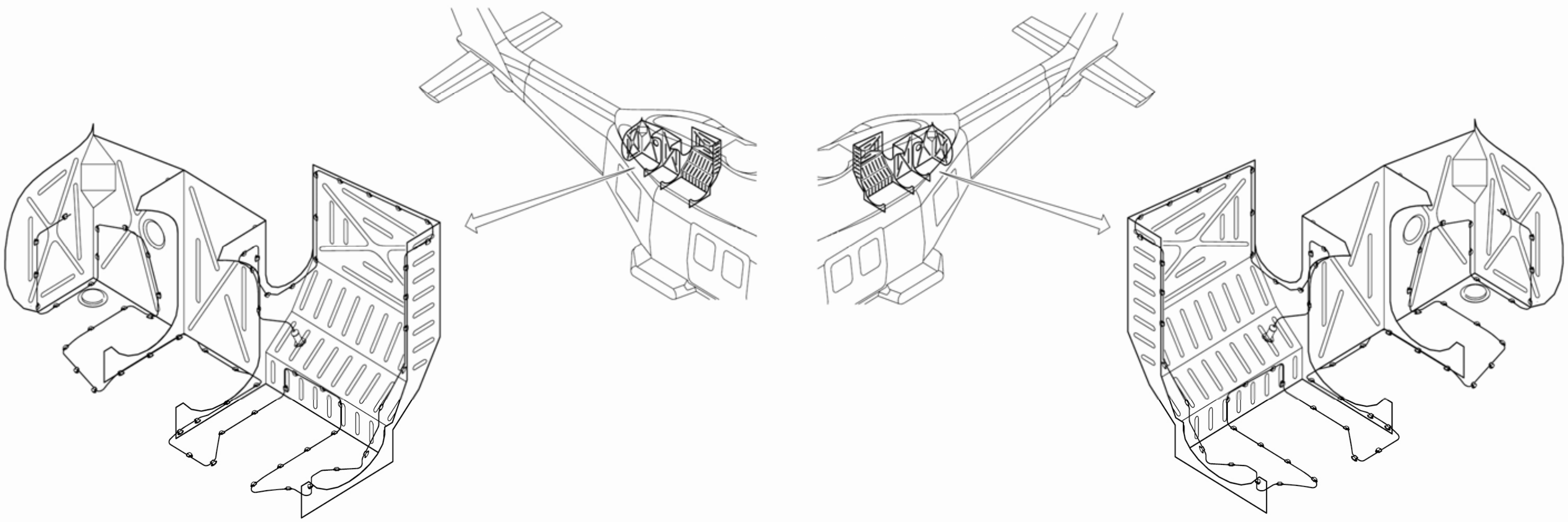
ENGINES MOUNTS

FIREWALL COMPARTMENTS

Engine firewalls are composed of a titanium structure which surrounds the engine on three sides.

The forward side is split to allow engine removal/installation. Firewalls are shaped to house the exhaust collector in the area between the two engines.

On firewalls are installed the electrical fire detection circuits and the outlet pipelines of the fire extinguishing system.



FIREWALL COMPARTMENTS

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CHAPTER 72

ENGINE

SECTION 00 – GENERAL

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ENGINE – PRINCIPLE OF OPERATIONS

Inlet air enters the engine through an annular chamber formed by the compressor inlet case. A wrap-around inlet screen prevents entrance of large particles of dirt or debris from entering the engine.

The inlet air moves rearward to the set of axial compressor and to the centrifugal impeller. The air from the impeller passes through the diffuser tubes, which turn the air 90° in direction and converts air velocity to static pressure. This high-pressure air surrounds the combustion chamber and is used in the combustion process. The combustion chamber has perforations of various sizes that allow entry of compressor delivery air.

The flow of air changes direction of 180° as it enters and mixes with fuel. Metered fuel is sprayed into a reverse flow annular combustion chamber through fourteen (14) fuel nozzles mounted around the gas generator case. The nozzles are supplied with fuel by a fuel manifold and each has a built-in flow divider.

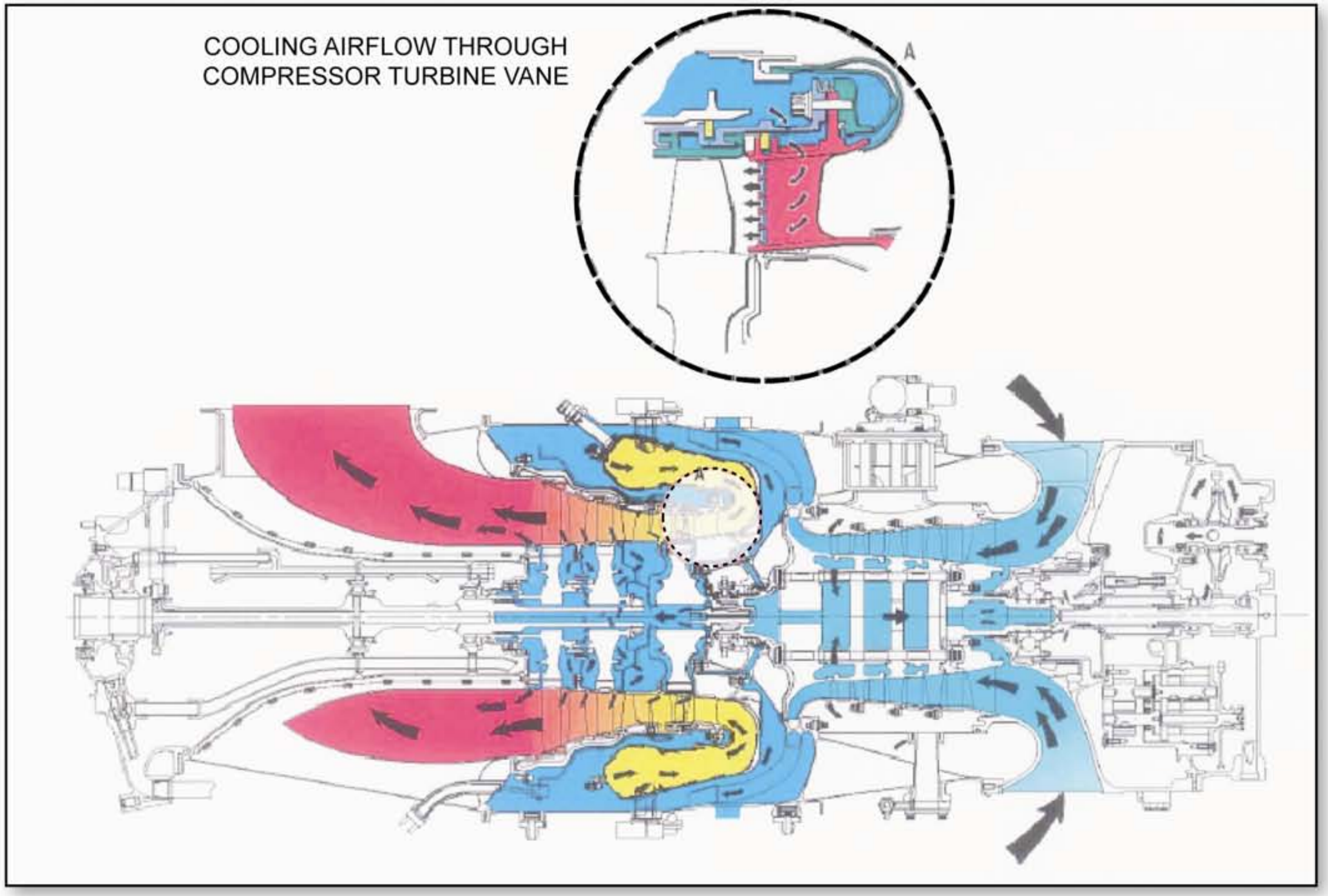
Two spark igniters powered by a high voltage ignition unit ignite the fuel-air mixture. When the combustion is started the ignition is turned OFF.

The resultant expanding hot gases from the combustion chamber reverse direction in the exit duct zone and pass through the compressor turbine vane ring to the single-stage compressor turbine. The stator vanes direct the hot gases towards the power turbine blades at the correct angle, with a minimum loss of energy.

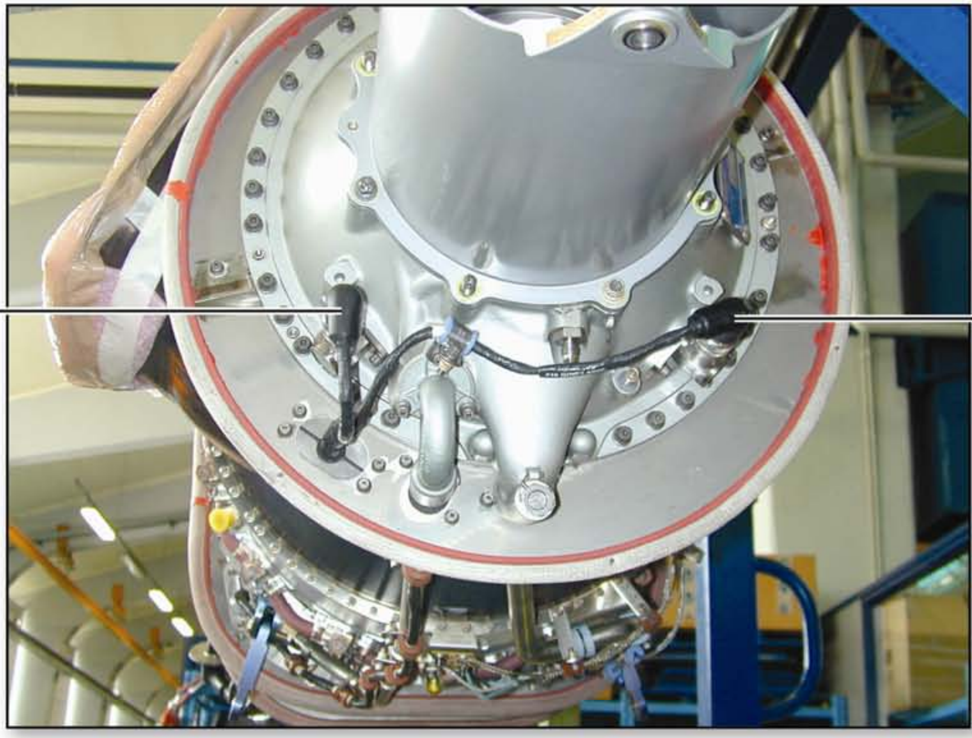
The still expanding gases continue rearward to the first stage power turbine vane ring and turbine, then to the second stage power turbine vane ring and turbine.
The exhaust gas from the power turbine is finally directed through an exhaust duct to the atmosphere.
The power turbine turns the output shaft located at the front of the engine.

The output shaft includes a phase shift torque meter device with sensors that supply input to the Electronic Engine Control (EEC) unit and provides an accurate cockpit indication of engine torque.

The lubricating oil is contained in the integral oil tank located between the air inlet and the accessory gearbox. Pressurized oil is provided to all bearings and various locations for cooling, lubricating and cleaning by pressure pump and jets.



ENGINE SCHEMATIC



TORQUE AND NF SENSORS

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CHAPTER
73
ENGINE FUEL AND CONTROL

SECTION 00 – GENERAL

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ENGINE FUEL AND CONTROL – GENERAL

The engine fuel and control system provides fuel to the engine at the required pressure and flow to avoid main rotor droop (isochronous governing); to reduced pilot workload on engine control; to optimize engine power and to improve engine response.
The engine fuel and control system interfaces the following systems

- Electronic Engine Control (EEC) unit
- Permanent Magnet Alternator (PMA)
- gas generator and power turbine speed sensors
- torque sensor
- gas Temperature (ITT) sensing system
- Data Collection Unit (DCU)

ENGINE FUEL AND CONTROL – MAIN COMPONENTS

The engine fuel and control system comprises the following main components

- Fuel Management Module (FMM)
- Fuel pumps and filter
- Fuel heater
- Fuel Cooled Oil Cooler (FCOC)
- Ecology accumulator
- Fuel manifold
- Fuel nozzles

FUEL MANAGEMENT MODULE (FMM)

The FMM is an hydro-mechanical unit that controls the fuel delivered to the engine using Electronic Engine Control (EEC) signals, Power Lever Angle (PLA) position, engine compressor discharge P3 pressure and N1 / NG speed as input parameters.
The FMM include a Fuel Metering Valve (FMV) which, under the action of the Electronic Engine Control (EEC) unit in automatic mode or the PLA in manual mode, sets the windows width that calibrate the fuel flow over the full range of the engine operations.

FUEL PUMPS AND FILTER

A low pressure pump provides adequate inlet pressure to the high pressure gear pump which, in turn, provides pressurized fuel to the metering section of the FMM. The filter is provided with a bypass valve.

FUEL HEATER

Prevents fuel filter restriction due to ice formation in the filter allowing engine operation at lower outside temperature.

FUEL COOLED OIL COOLER (FCOC)

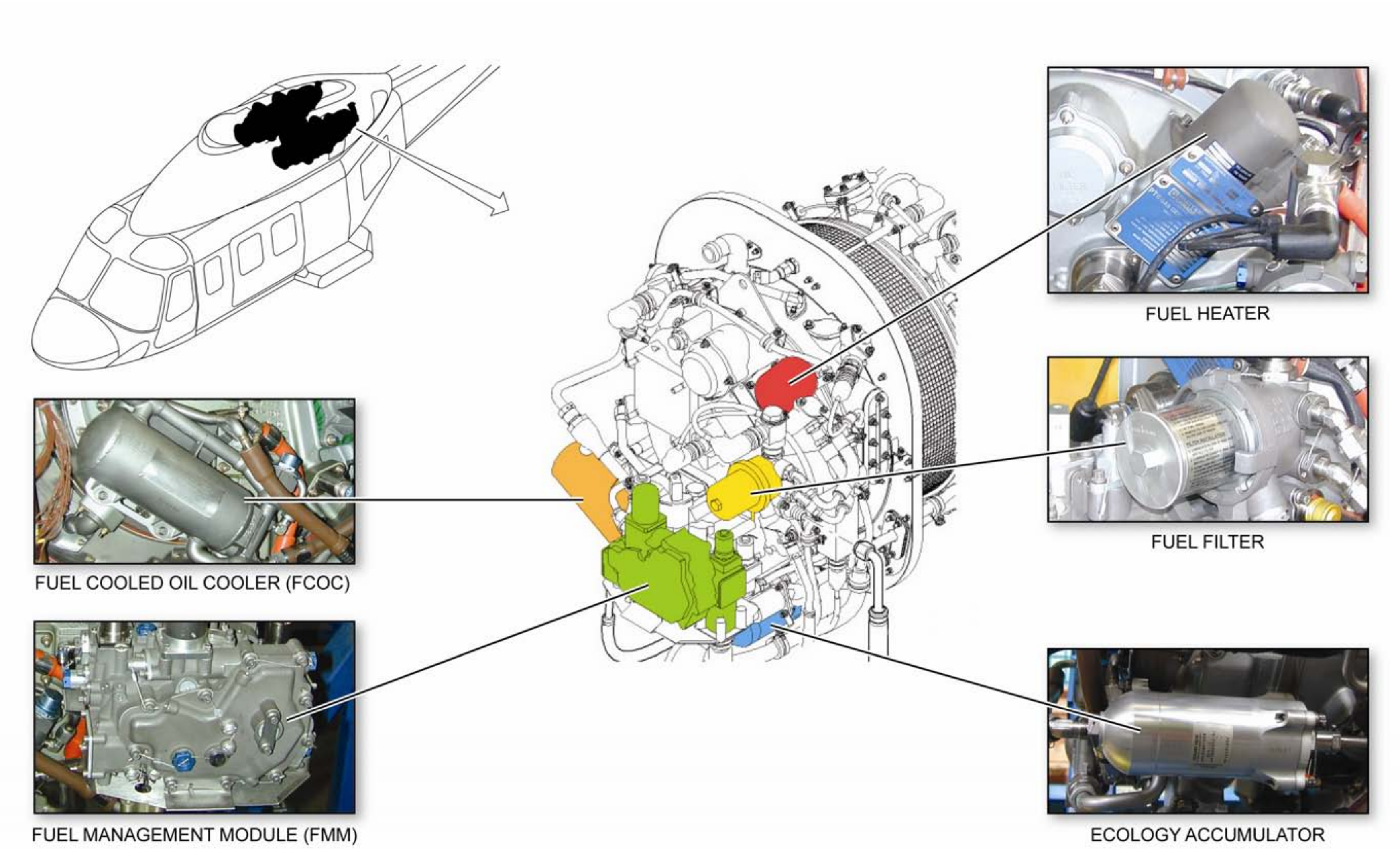
The Fuel Cooled Oil Cooler is a heater exchanger with two flow circuits: engine oil and fuel that provides for engine oil cooling with temperature regulation.

ECOLOGY ACCUMULATOR

Prevents fuel nozzle coking after shutdown and acts as manifold reservoir to collect residual fuel from the manifold on shutdown allowing to re-use this collected fuel for the next engine start.

FUEL NOZZLES

There are 14 fuel nozzles with integral flow divider that deliver and atomize metered fuel into the combustion chamber.



ENGINE FUEL AND CONTROL – MAJOR COMPONENTS

ENGINE FUEL AND CONTROL – OPERATIONS

The FMM operates in two modes

- AUTO mode (normal mode of operation)
- MANUAL mode (backup mode of operation)

The fuel, coming from the tank, is driven by a low pressure pump through a heater and then a filter. After crossing the gear pump, the pressurized fuel passes through a metering unit where a Flow Metering Valve (FMV) keeps a constant value of differential fuel flow pressure.

A by-pass flow line sends the exceeding fuel back to the low pressure pump. The metered fuel flow is sent through the FCOC to the fuel nozzles.

The difference between the AUTO and the MANUAL mode is how the Flow Metering Valve acts to meter the fuel flow:

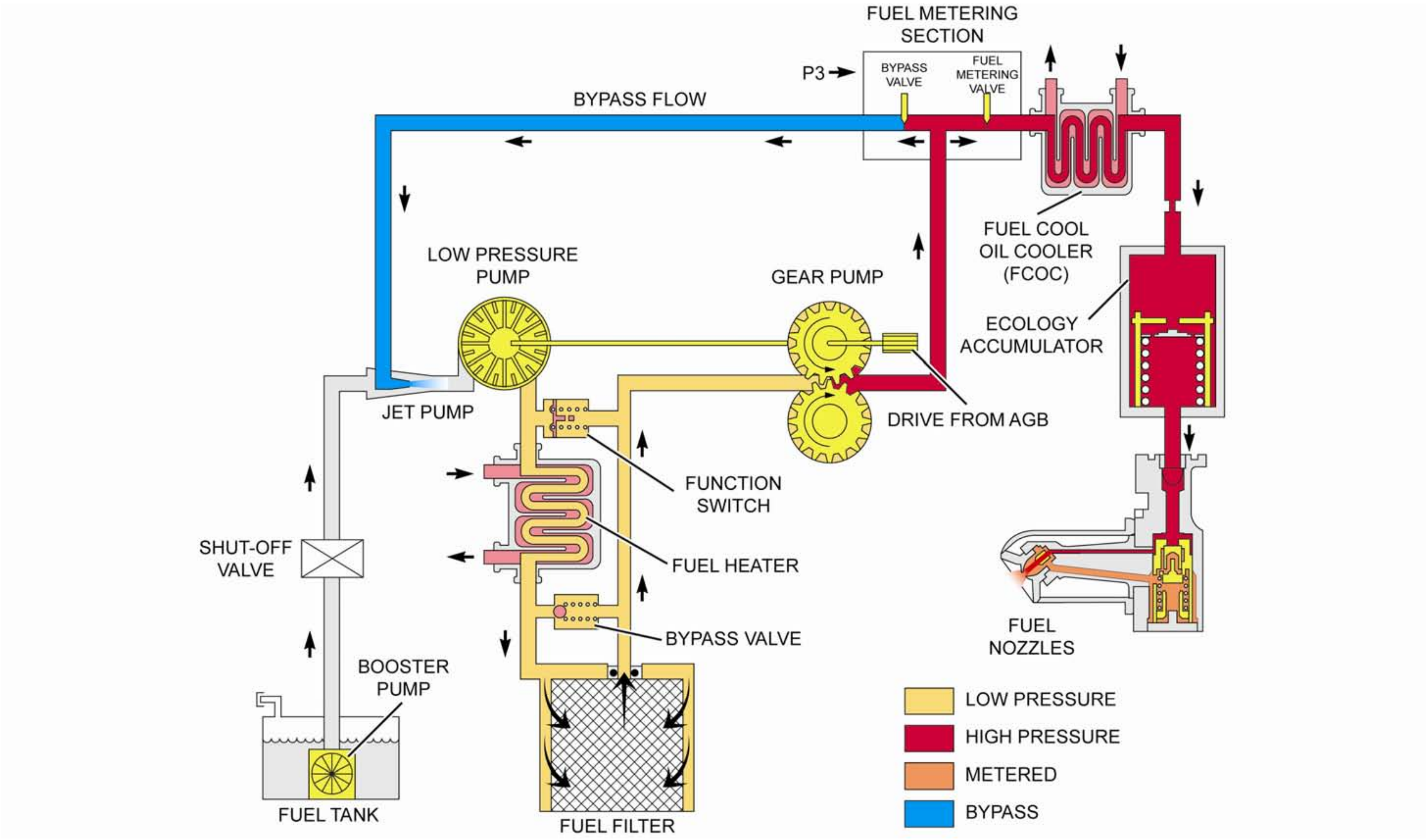
- in automatic mode a torque motor commanded by the EEC in conjunction with the engine compressor discharge P3 pressure is used to meter the fuel flow
- in manual mode a cam actuated by the PLA in conjunction with the engine compressor discharge P3 pressure is used to meter the fuel flow

The transition from AUTO to MANUAL mode and viceversa is possible through the TRANSFER OPERATION MODE that acts so that smooth conditions to be satisfied during the transition.

In case of manual operations, for training or in emergency, the pilot has to adjust the engine power moving the Engine Control Lever (ECL): in this case if the pilot reacts too much quickly then the engine acceleration or deceleration will be automatically adjusted by the FMM in order to prevent surge or flameout.

EEC OPERATIONS

The Electronic Engine Control unit operates in conjunction with the FMM to monitor different engine parameters and adjust the fuel flow delivered to the engine from start to full power within settled upper and lower limits. In case of malfunction, a fuel metering manual backup mode is activated where fuel flow control reverts only to the FMM.



ENGINE FUEL - SCHEMATIC

COMMON MALFUNCTION SYMPTOMS

It is useful for the pilot to set a connection between some common symptoms and the probable cause.

The table reports some examples.

SYMPTOMS	PROBABLE CAUSE
Engine fails to light up	Air in fuel system
	Ignition system
Hung start slow start	P3 leak to FMM
	FMM rigging
	Ignition system
	Ecology fuel accumulator
	Fuel nozzle
	FMM
Insufficient NG/N1 at start	Starter-generator low voltage
	Indicating system
	Starter-generator seizure
	Engine FOD or blockage
Hot start	Insufficient start assist
	Improper starting procedure
	Ignition system
	Ecology fuel accumulator
	Fuel nozzles

CHAPTER
74
IGNITION

SECTION 00 – GENERAL

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IGNITION – GENERAL

Ignition provides the initial spark to ignite the fuel-air mixture into the combustion chamber. Each engine has its proper ignition system that consists of one ignition unit, two high-tension cable assemblies and two spark igniters that transforms the 28 V DC input into a pulsed high voltage output. All the components are installed on the engine except for the electrical supply circuit.

The igniters are not continuously rated but are only operational during the engine starting sequence.

The airframe wiring provides ignition power when commanded by the Electronic Engine Control (EEC). The EEC commands the operation of the ignition system and of the starter when receives a control signal from the Engine Control Panel switch on the central console.

IGNITION – PRINCIPLE OF OPERATION

The ignition can be performed in two ways

- automatically
- manually

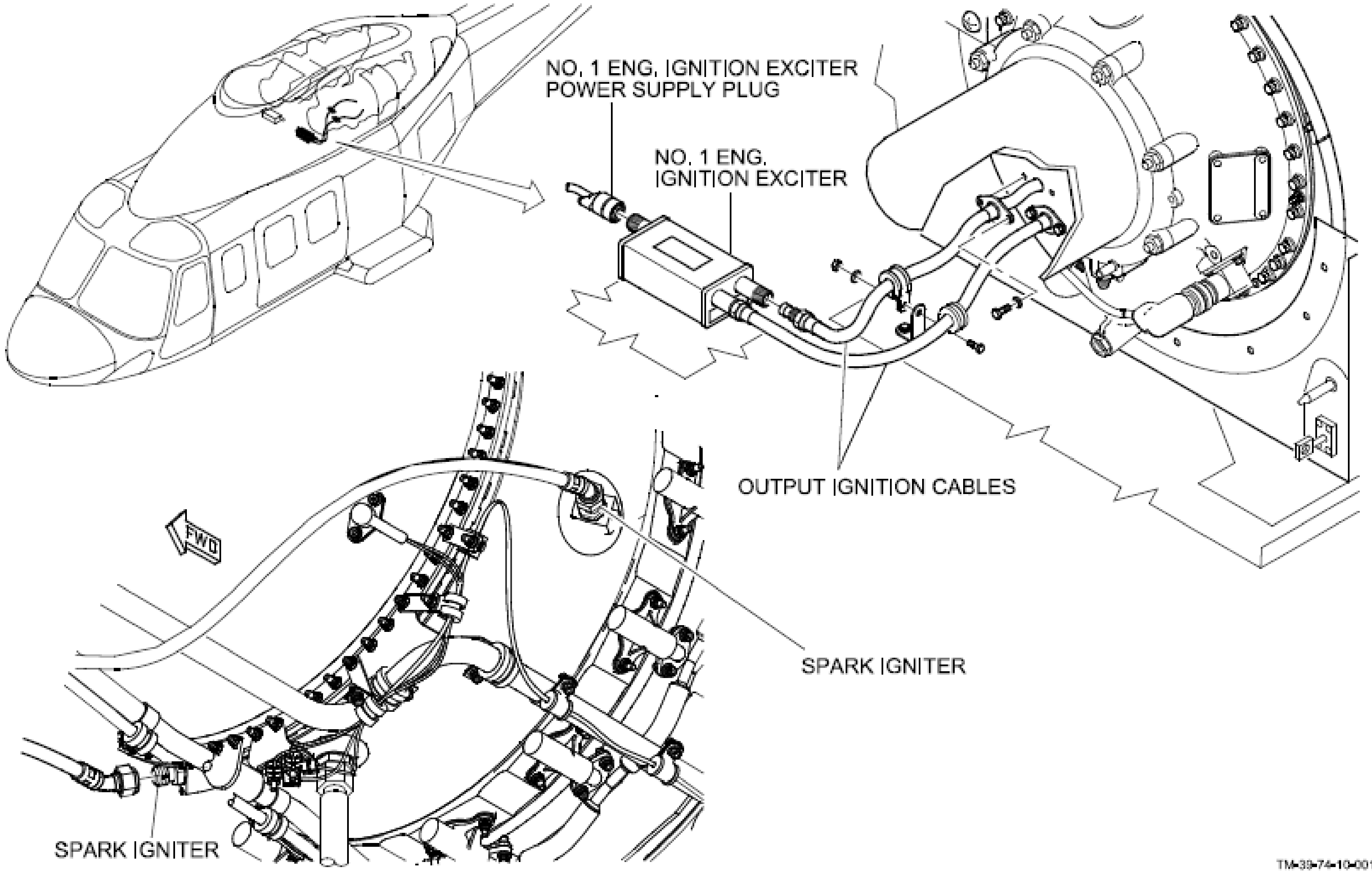
For each engine, the selection is made by the ENG GOV switch on the pilot collective grip setting the switch to AUTO or MANUAL.

AUTOMATIC MODE

The automatic ignition starts by setting the ENG 1 MODE selector switch to IDLE (or the ENG 2 MODE selector switch to IDLE).
During automatic start the green legend IGN will be displayed on the MFD aside the ITT scale (on the left for the engine no.1 and on the right for the engine no.2). The ignition is disengaged when the EEC senses that the engine speed value is at 49 ± 1% NG.

MANUAL MODE

In case of the automatic mode cannot be performed, the manual mode is made by setting the ENG GOV switch to MANUAL. The pilot starts the engine by pressing the START button on the Engine Control Level (ECL). As the engine speed reaches the 49 ± 1% NG, the ignition sequence is stopped automatically by the relative GCU.
During manual start the green legend IGN will be displayed on the MFD aside the ITT scale (on the left for the engine no.1 and on the right for the engine no.2).



IGNITION SYSTEM – COMPONENT LOCATION

IGNITION SYSTEM – CONTROLS AND INDICATORS

1. 1 ENG GOV 2 switch on PLT collective stick

   AUTO …….. when selected enables the automatic start of the relevant engine

   MANUAL …. when selected enables the manual start of the relevant engine

2. ENG 1 (2) MODE switch on Engine Control Panel (ECP)

   OFF ……..… no electrical power is sent to spark igniters

   IDLE …........ when selected electrical power is applied to spark igniters until the 49 ± 1% Ng is reached (AUTO MODE)

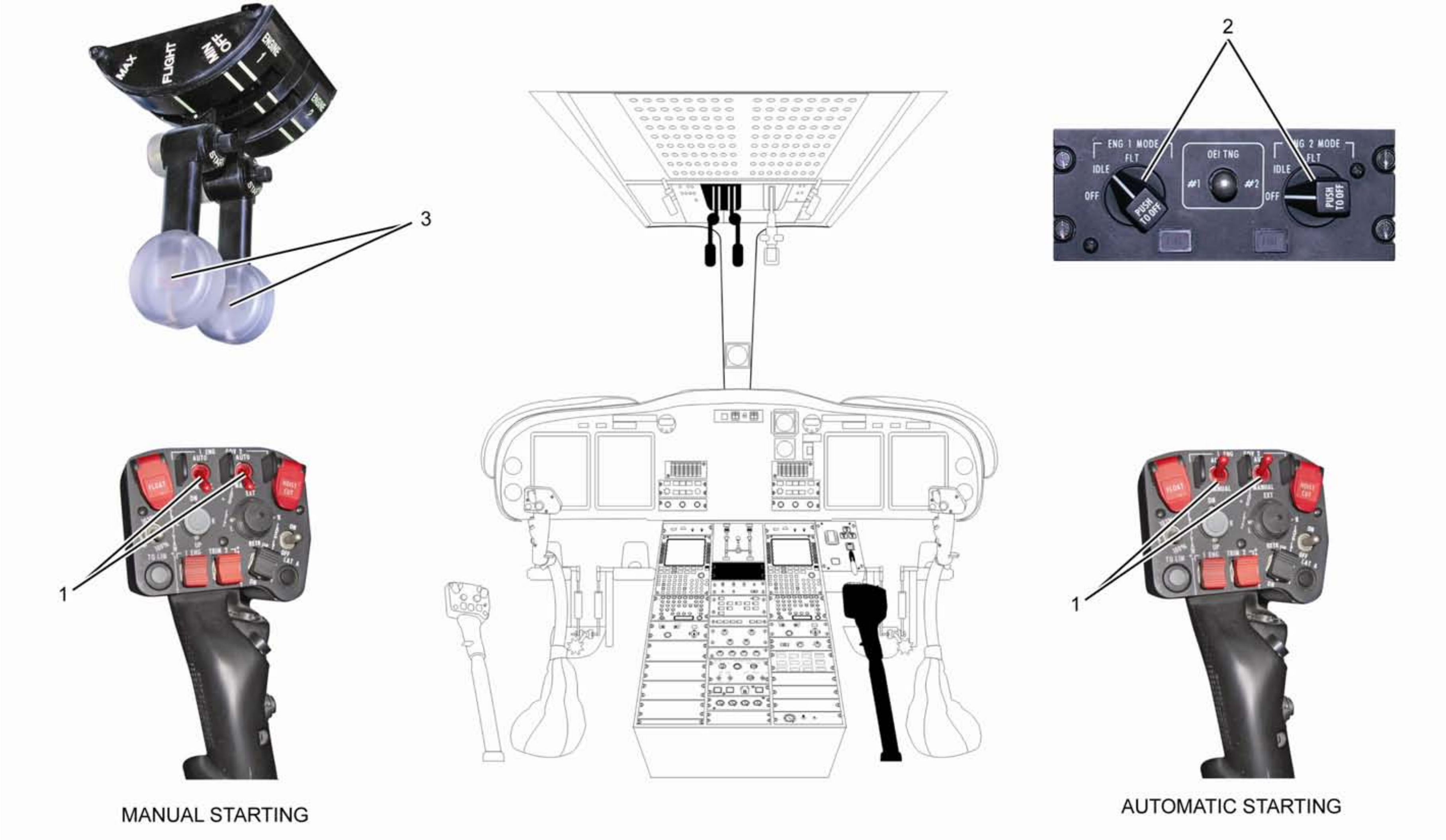
   FLT ..…......... when selected allows the quick start of the engine

   NOTE. During ignition the MFD provides the green legend IGN.

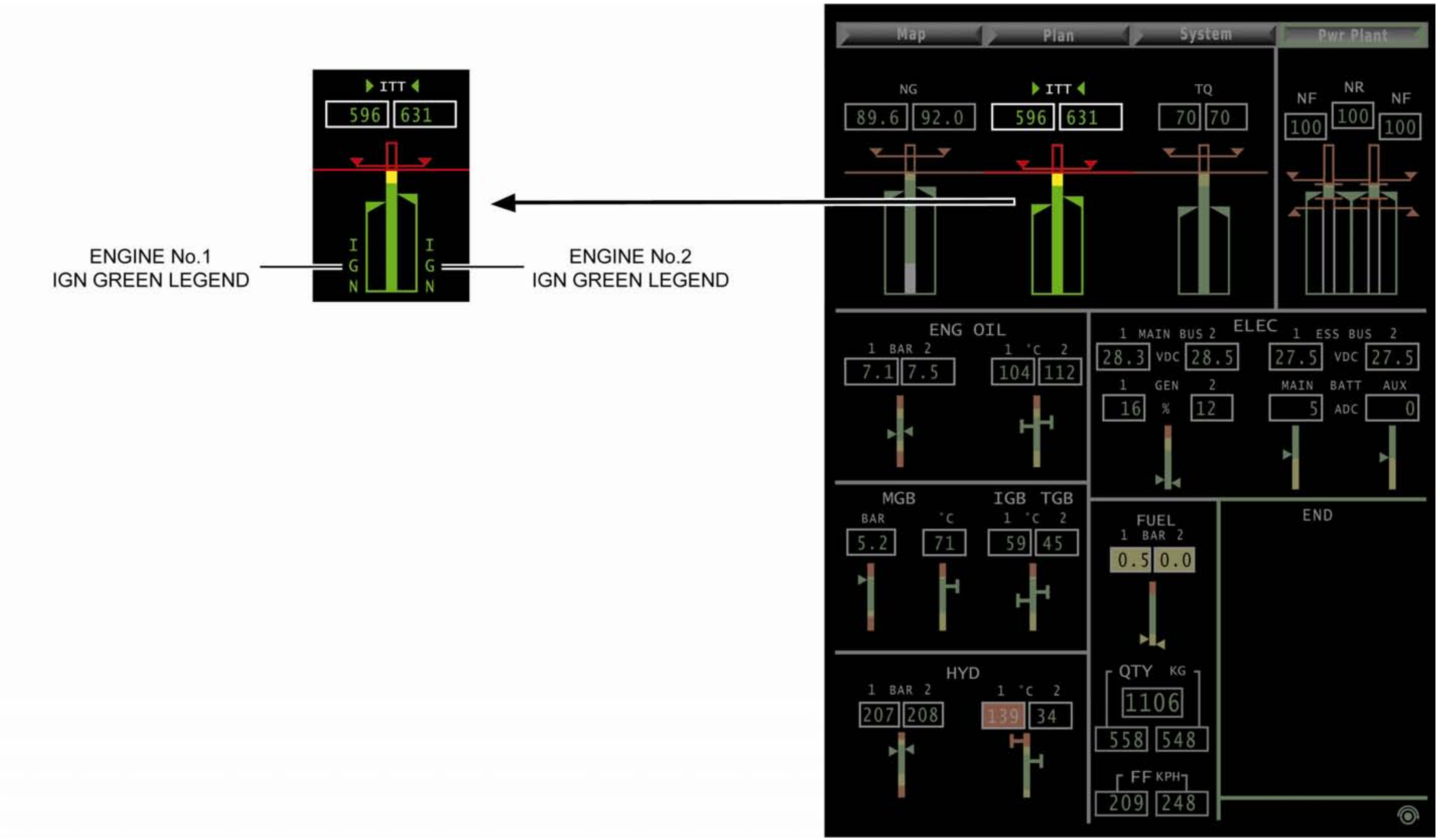
3. ENG 1 (2) start button switch on Engine Control Lever (ECL)

   IDLE …........ when pressed electrical power is applied to spark igniters until the 49 ± 1% Ng is reached (MANUAL MODE)

   NOTE. During ignition the MFD provides the green legend IGN.



IGNITION SYSTEM – STARTING



IGNITION SYSTEM – CONTROLS AND INDICATIONS

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CHAPTER 75

AIR

SECTION 00 – GENERAL

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AIR – GENERAL

Assuming all the air entering the engine as the 100%, than

the primary air system (air used to produce power) uses

- the 25% for combustion process
- the 60% to cool the combustion gasses

the secondary air system uses the remaining 15%, that is

- the 8% for hot section cooling
- the 2% for labyrinth / carbon seals and bleed valve
- the 5% for the airframe

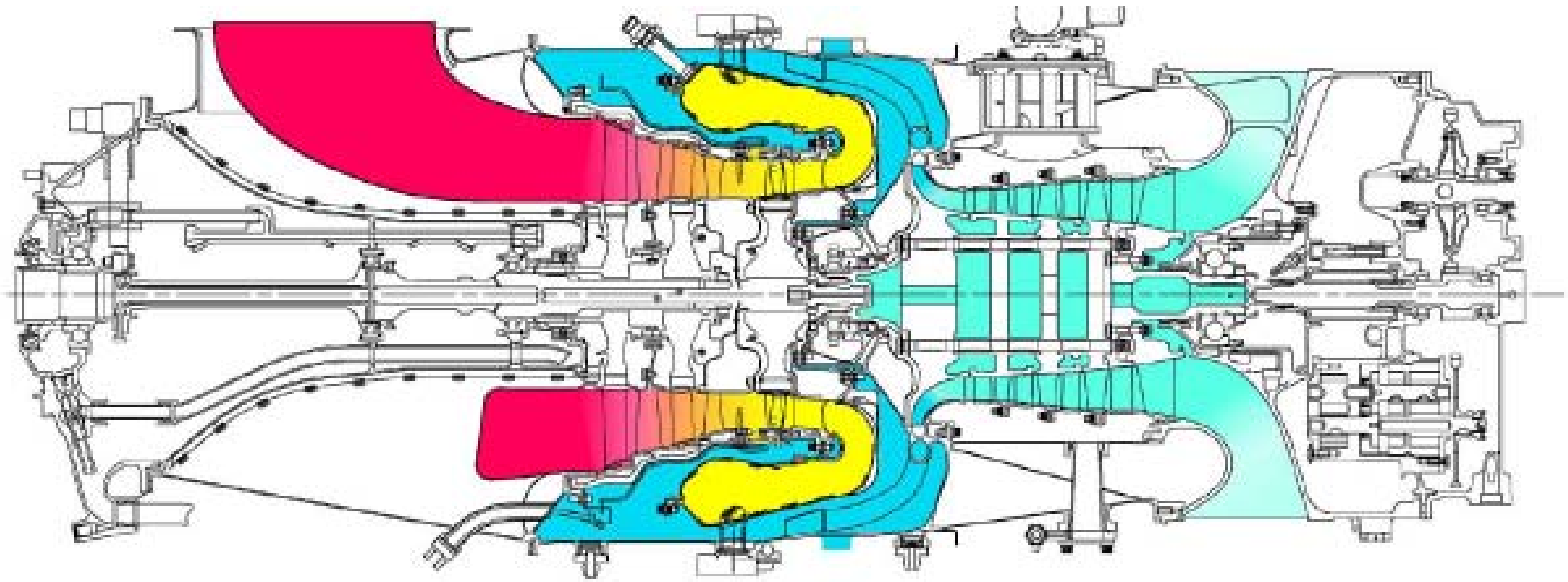
AIR – OPERATIONS

Inlet air enters the engine through an annular plenum chamber made by the compressor inlet case. A wrap-around inlet screen prevents entrance of large particles of dirt or debris from entering the engine. The inlet air moves rearward to the set of axial compressor and to the centrifugal impeller. From the impeller the air passes through the diffuser tubes which turn the air 90° in direction and converts air velocity into static pressure. This high-pressure air surrounds isolating the combustion chamber and is used in the combustion process.

The combustion chamber includes holes of various sizes that allow entry of compressor delivery air. The flow of air changes direction 180° as it enters and mixes with fuel.
Metered fuel is sprayed into a reverse flow annular combustion chamber through fourteen (14) fuel nozzles mounted around the gas generator case. The nozzles are supplied with fuel by a fuel manifold and each has a built-in flow separator.
Two spark igniters powered by a high voltage ignition unit ignite the fuel / air mixture. When the combustion is started the ignition is turned OFF.

The resultant expanding gases are directed to the compressor turbine and power turbine reversing their direction in the exit duct zone and passing through the compressor turbine. The stator vanes direct the hot gases towards the turbine blades with a minimum loss of energy.
The still expanding gases continue rearward to the first stage of the power turbine and then to the second stage of the power turbine. The exhaust gas is at last directed through an exhaust duct to the atmosphere.
The power turbine under the action of the hot gases turns the output shaft located at the front of the engine.

The output shaft includes a phase shift torque meter device with sensors that supply input to the EEC for control and provides an accurate cockpit indication of engine torque.



AIR – SCHEMATIC

SECONDARY AIR SYSTEM - GENERAL

The secondary air system consists of all the pressure air that is not used directly to produce power. There are two sources of secondary air

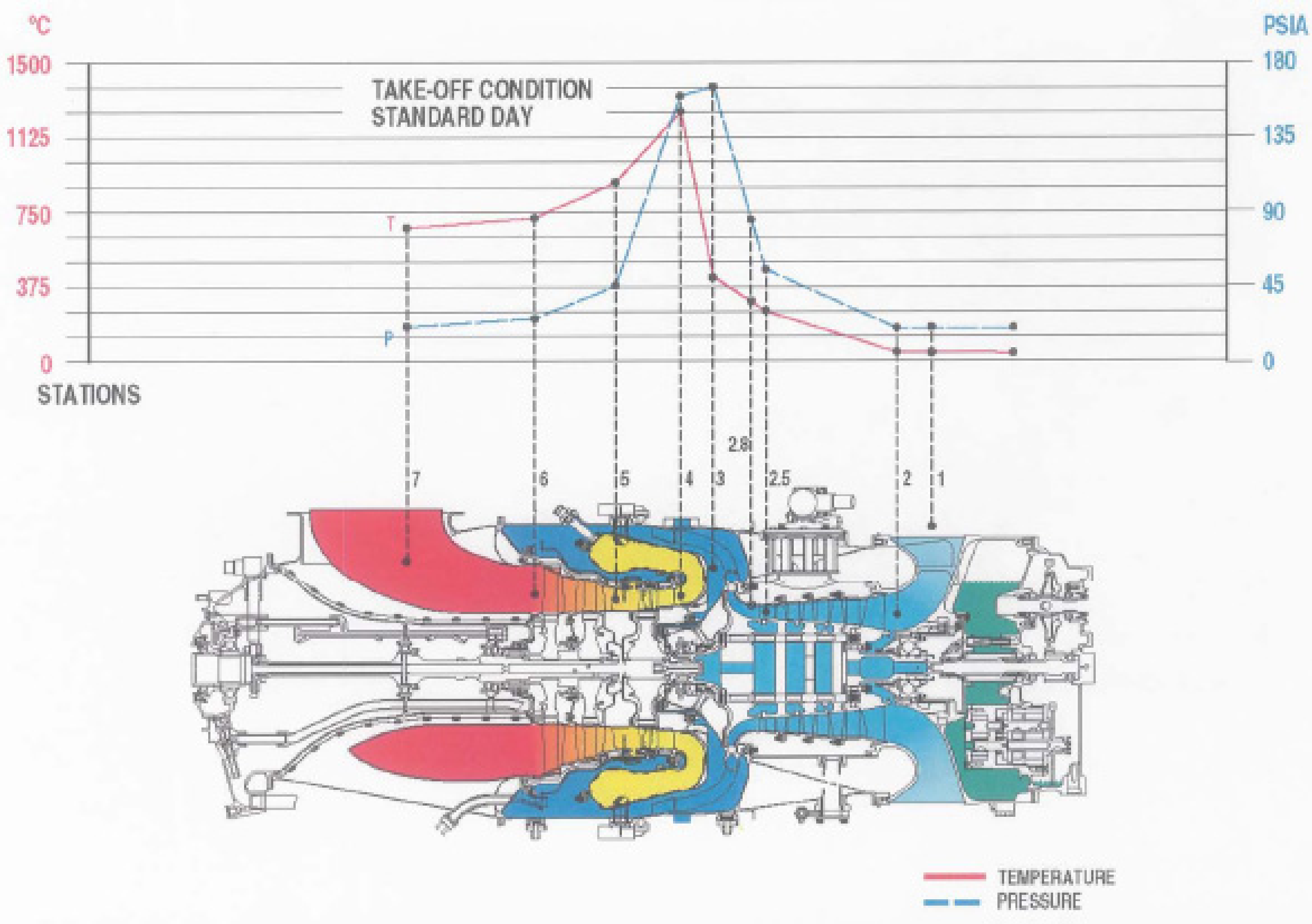
- the P2.5 interstage air source that is used for

  - sealing of the no.1 bearing cavity

- the P3 compressor delivery source that is used for

  - cooling of hot section parts
  - sealing of bearing compartments
  - operation of bleed valve
  - operation of FMM

The total air cabin bleed (heating) is 5% as maximum.



PRESSURE AND TEMPERATURE VALUES VS AIR STATIONS

CHAPTER 76

ENGINE CONTROLS

SECTION 00 – GENERAL

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ENGINE CONTROLS – GENERAL

The following systems control of the engines

- the Fuel Management Module (FMM)
- the Electronic Engine Control (EEC)
- sensors (gas generator turbine sensor (NG), torque sensor (TQ), power turbine speed sensor (Nf))
- the Inter Turbine Temperature system (ITT)

The pilot controls the engines

- in AUTO mode (normal automatic mode)
- in MANUAL mode (manual backup mode)

Engine controls and indications are distributed on

- the MISC panel
- the Engine Control Panel (ECP)
- the Engine Control Lever (ECL)
- the pilot and copilot collective grip
- the MFD Main and Cruise pages
- the PFD default page

ENGINE CONTROLS – OPERATIONS

NORMAL AUTOMATIC MODE

The normal automatic mode (AUTO) is the primary mode of operation achieved with the ENG MODE switch on FLIGHT position and ENG GOV switch on AUTO.

In automatic mode the EEC

- provides automatic controls of the full operating envelope of the engine without exceeding limiting parameters
- computes the desired fuel flow and through a torque motor sets the position of the metering valve in the FMM to achieve a new NG reference point in accordance with variation of pressure altitude, OAT and collective position

In case of failure the torque motor is set to a position which allows the maximum power to be used with the manual backup.

In automatic mode the manual backup constantly tracks the actual NG speed in order to minimize the change in engine power when a transfer to MANUAL mode will occur.
If the pilot selects the MANUAL mode or if a critical engine failure should occur the system reverts to the manual backup mode.

MANUAL BACKUP MODE

The manual backup mode (MANUAL) is the secondary mode of operation achieved with the ENG MODE switch on IDLE or FLIGHT, the ENG GOV switch on MAN and using the ENG TRIM selector to operate the Engine Control Lever (ECL) (in case of failure of the electrical motor the ECL can be operated directly).

In manual mode the ECL

- is used to match the power required for the load on the rotor
- is used to set the power of the engine operating in manual mode about 10% torque below the engine running in AUTO mode during normal flight

The ECL is provided with two latches at MIN and FLIGHT position and two mechanical stop at OFF and MAX. The MIN latch is a mechanical one used to prevent to move unintentionally the ECL to OFF unless first pulled down. The FLIGHT latch is a magnetic stop which prevents to move the ECL beyond the FLIGHT position when the AUTO mode is selected; in case of MANUAL mode selection the latch is electrically disabled in order to allow pilot's free operation over MIN and MAX positions. A small mechanical step is present at FLIGHT position so that pilot can feel it and overcome with a negligible force.

AUTOMATIC STARTING AND SHUT DOWN

The starting is controlled by the EEC.

At all time during start sequence, pilot monitoring of engine limitations is required to prevent engine deterioration due to abnormal starting conditions.
During the starting sequence the ignition system is automatically turned on and the EEC provides the signal to the GCU to terminate the starting cycle at 50 ±1% NG.

The pilot to starts the engines in AUTO mode via the 1 (2) ENG GOV switch on the pilot collective grip and the Engine Control Panel (ECP).

Refer to AW139-RFM-4D for the complete starting procedure.

Engine starting abort or engine shut down can be accomplished at any time by setting the ENG MODE switch to OFF.
Normal engine shut down is carried out following a stabilisation period at idle speed.

It is recommended that engine start be carried out in AUTO mode when possible.

MANUAL STARTING AND SHUT DOWN

The starting is controlled by the GCU.

At all time during start sequence, pilot monitoring of engine limitations is required to prevent engine deterioration due to abnormal starting conditions.
During the starting sequence the ignition system is automatically turned on and the GCU uses an internal automatic starter cut-out feature, such that the GCU will open the starter-generator line contactor to terminate the start cycle at 49 ±1% NG.

The pilot starts the engines in MANUAL mode via the 1 (2) ENG GOV switch on the pilot collective grip, the Engine Control Lever (ECL) and the Engine Control Panel (ECP).

Refer to AW139-RFM-4D for the complete starting procedure.

Engine starting abort or engine shut down can be accomplished at any time by setting the ENG MODE switch to OFF and the Engine Control Lever to OFF.
Normal engine shut down is carried out following a stabilisation period at idle speed.

MAIN ROTOR GOVERNING

NORMAL FLIGHT

The EEC allows to maintain the nominal NF in all flight conditions in accordance with the NF selected on the collective grip (100% RPM or 102% RPM for cat A flight).

Collective stick position is provided to EEC through an LVDT used to anticipate a load change such that rotor droop or overshoot can be minimized during fast collective lever applications.

Main rotor governing is also achieved maintaining power matching between engines in AUTO mode. The principle of the torque/ITT matching is to increase power delivered by the engine which has the lower load until the matching is achieved.

Torque or ITT matching can be selected using the LD SHARE switch on the MISC control panel

- torque matching is the normal mode of operation
- ITT matching should be selected in case of the engines are ITT limited and there is a large ITT mismatch

AUTOROTATION

The EEC recognizes an autorotation condition by comparing main rotor speed Nr with engine power turbine NF speed and engine torque level.

Recovery from an autorotation condition is anticipated by the control logic in order to minimize rotor droop on fast collective load application.

OEI TRAINING MODE

The Training Mode logic uses twin engine power to simulate an initial single engine transient to “maximum torque” and rotor droop. Then both engines are used but they are limited to a maximum total PI of 140%.

When the Training Mode is activated, the PI and NF displays on the PFD are artificially configured to show OEI condition for the engine not selected to OEI TNG.

On the MFD, the actual engine parameters are displayed while on the NR/NF indicator the coloured ranges are modified, from AEO to OEI, to allow NR/NF droop to 90% as required by the cat A procedure. With these presentations the PFD presents the simulated OEI condition while the MFD, for safety reasons, presents the real AEO conditions.

In order to simulate the transient following an engine failure the engine selected to OEI TNG will accelerate to a maximum of 110% torque (MFD display) then reduce to approximately 70% while the inoperative engine will decelerate to a minimum of approximately 25% torque then accelerate to around 70%.

The OEI training mode is disabled if

- either engine is in manual mode
- a critical or non-critical fault exists on either engine
- either engine flames out
- the torque limiter switch is OFF
- the ENG switch is not in FLIGHT position on either engine
- the Nr speed drops below 87%

In case the above conditions apply when the OEI training mode is selected, the switch is reset to the center (OFF) position.

ENGINE CONTROLS AND INDICATORS

The Engine Control Panel (ECP) and the pilot collective stick are used to start the engine in AUTO mode.

1. ENG 1 (2) MODE three selector switch

   OFF ……..…. allows to shut down the engine no.1 (2)

   IDLE ……..…. allows

   - the starting of the engine no.1 (2)
   - the minimum on ground (65% Nf)
   - the no.1 (2) engine stabilization for a period of 60 sec. during shutdown
   - the no.1 (2) engine wet motoring procedure both in AUTO and MANUAL mode

   FLT ……..….. allows to perform the no.1 (2) engine quick starting at flight nominal speed (100% Nf)

2. ENG 1 (2) GOV AUTO/MANUAL selector switch

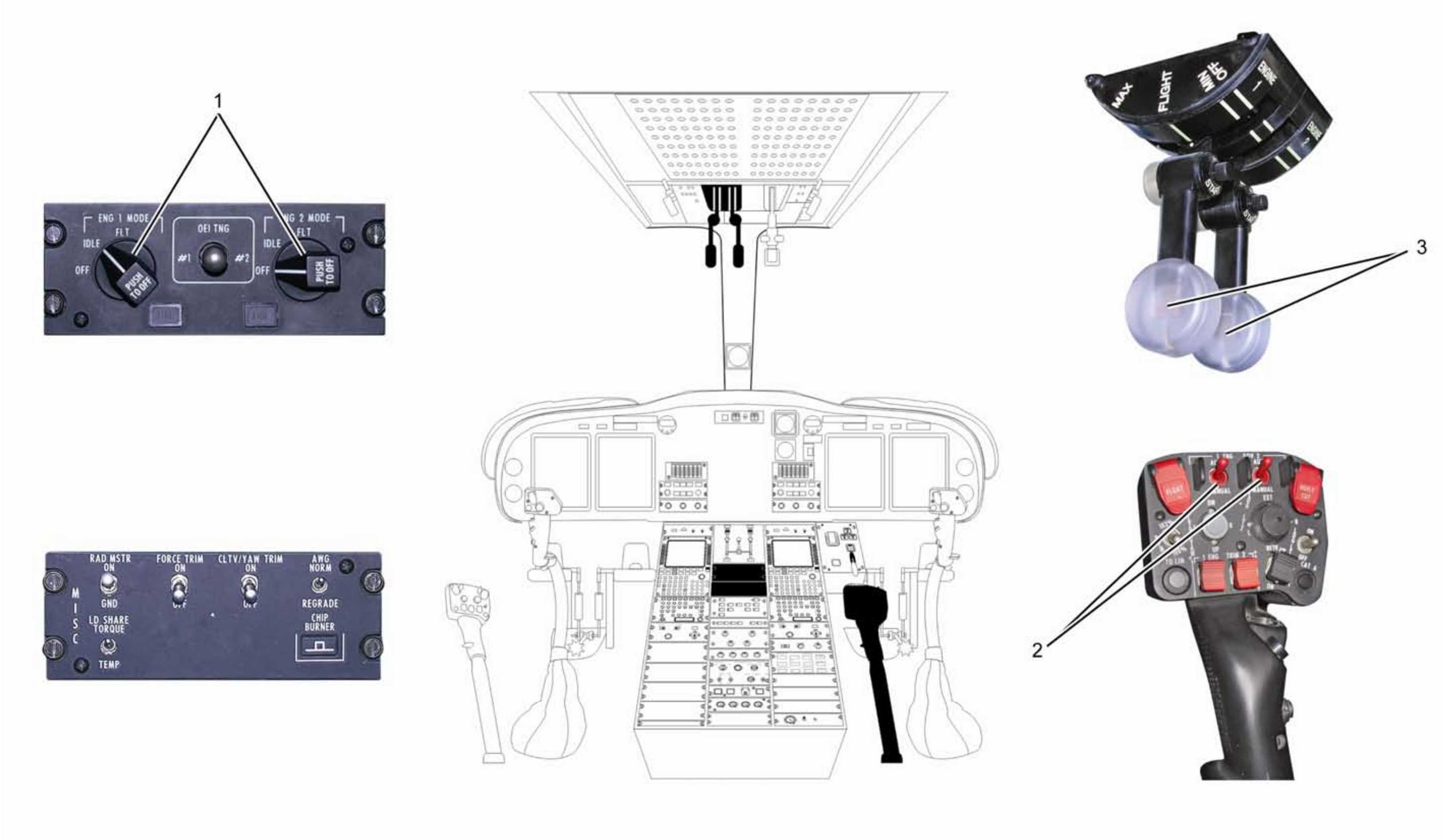
   AUTO …..….. • allows the automatic starting of the engine no.1 (2) and provides the automatic control of the engine

   MANUAL ….... • allows the manual starting / wet motoring / dry motoring of the engine no.1 (2) and provides the manual control of the engine no.1 (2)

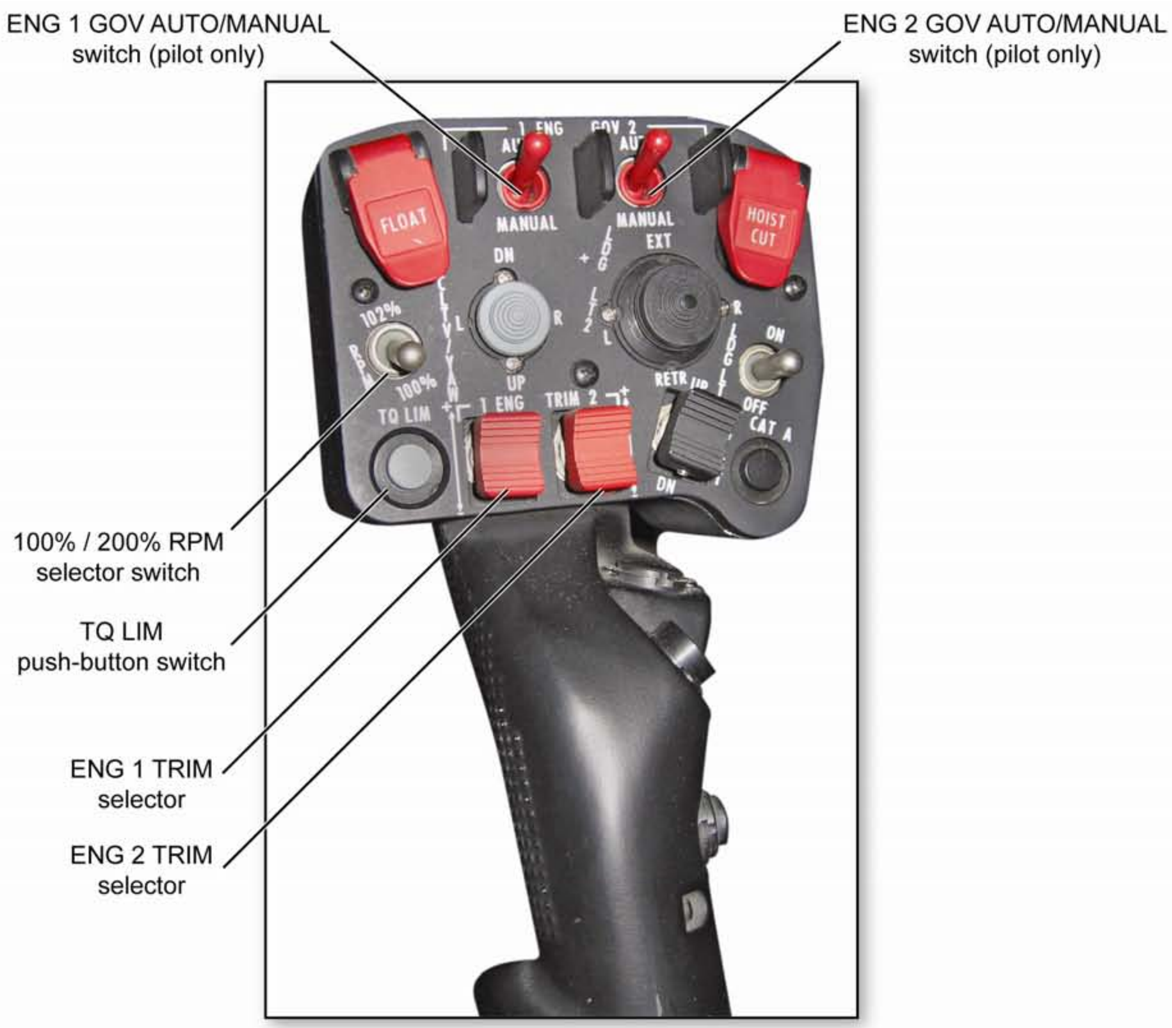
The Engine Control Lever (ECL) on the overhead panel, the ECP and the pilot collective stick are used to start the engine in MANUAL mode.

3. ENG 1 (2) START push-button switch on ECL

   PRESSED …. the manual starting / wet motoring / dry motoring of the engine no.1 (2) is performed in manual mode



ENGINE CONTROLS



PILOT COLLECTIVE GRIP

ENGINE CONTROL LEVER (ECL)

Engine parameters are displayed on the Main and Cruise pages of the MFD.

The NG, ITT, TQ indications are displayed on one single vertical scale with one analog pointer and one digital readout at the top of the scale.

The NF indications are displayed on two vertical scales combined with one NR vertical scale and one analog pointer per engine (triple tachometer scale).

At initial power up

- the MFD displays the All Engine Operative (AEO) mode by default
- the PFD displays the PI (Power Index) scale. The PI provides a single vertical scale on which an overall indication of engine performance is displayed for the three primary engine parameters. To achieve this, the three parameters are re-scaled to obtain a Power Index (PI) which enables a comparison. The PI is expressed in terms of torque equivalent. For ITT and NG, the torque equivalent is obtained by determining the relationship between these parameters and torque when operating near the Maximum Continuous Power (MCP) limit. Having rescaled the parameters, the largest parameter is displayed on the PI scale together with an annunciator of which parameter is currently being displayed.

The engines have been assigned to three different ratings

- All Engines Operating (AEO) rating
- One Engine Inoperative (OEI) rating. This rating is active if one engine fails or is not able to deliver power
- One Engine Inoperative Training (OEI TNG) rating. This rating is activated manually by the pilot through the switch selection on the Engine Control Panel

The AMBER LEGEND displayed on the MFD Main and Cruise pages and the PFD are the following

FAIL	• displayed on the left/right side of TQ/NF sensors failure
OEI	• displayed on the right/left side of PI, NG, ITT and TQ
MAN	• displayed on the right/left side of TQ/PI scales
OEI TNG	• OEI displayed on the right/left side of PI, NG, ITT and TQ scales, TNG in reverse video on the right/left side of PI and TQ scales
2.5 m	• displayed steady amber on the right/left side of PI scale and between the NG and ITT scales when 2.5 m excursion time is detected • displayed blinking red inverse video when the 2.5 m time is within 10 sec from expiration • displayed steady red inverse video when the 2.5 m time has expired associated with the 1 (2) ENG LIM EXPIRE caution

The GREEN LEGEND displayed on the MFD Main and Cruise pages and the PFD are the following

START	• displayed on the left/right side of NG during start • during start the NG digital readout with the associated pointer are white until the 49% NG is reached
IGN	• displayed on the right/left side of ITT scales when spark igniters are powered
IDLE	• displayed on the right/left side of triple tachometer NF scales when the ENG MODE switch is set to IDLE • the IDLE legend matches the colour of corresponding NF pointer
►◄	• displayed aside the TQ or ITT scale legend according to the load sharing switch position

ENGINE INDICATING

- if both the EEC data and the corresponding back-up sensor input are not available or invalid, the associated pointer is removed from the display and amber dashes replace the numerical readouts

- when a parameter being monitored exceeds the normal range of operation (green band), the colour of associated pointer matches the colour of applicable range marking (i.e. amber or red), in order to highlight that particular critical condition

- if the NF sensor fails, the relative pointer is removed and replaced by the amber legend FAIL in reverse video

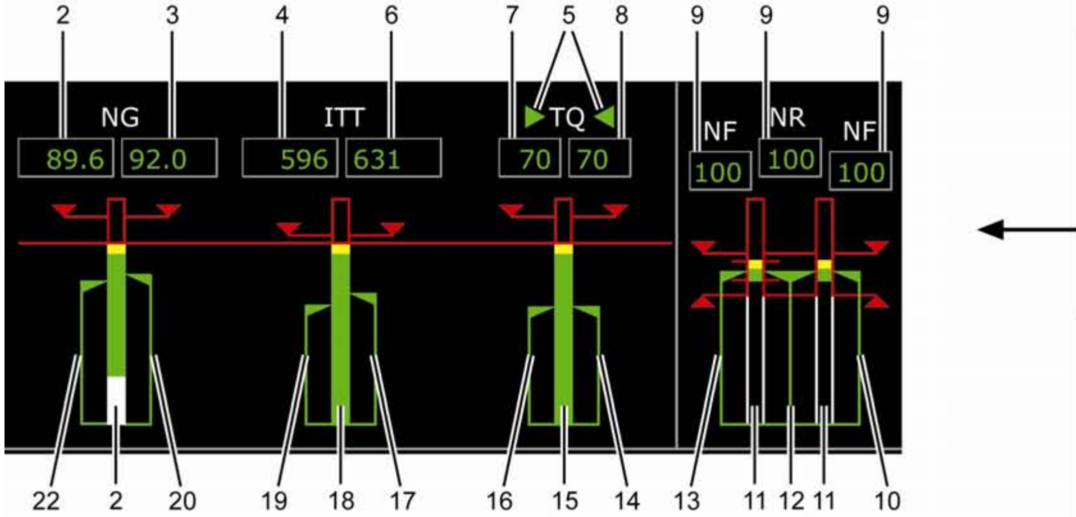
POWER INDEX INDICATING

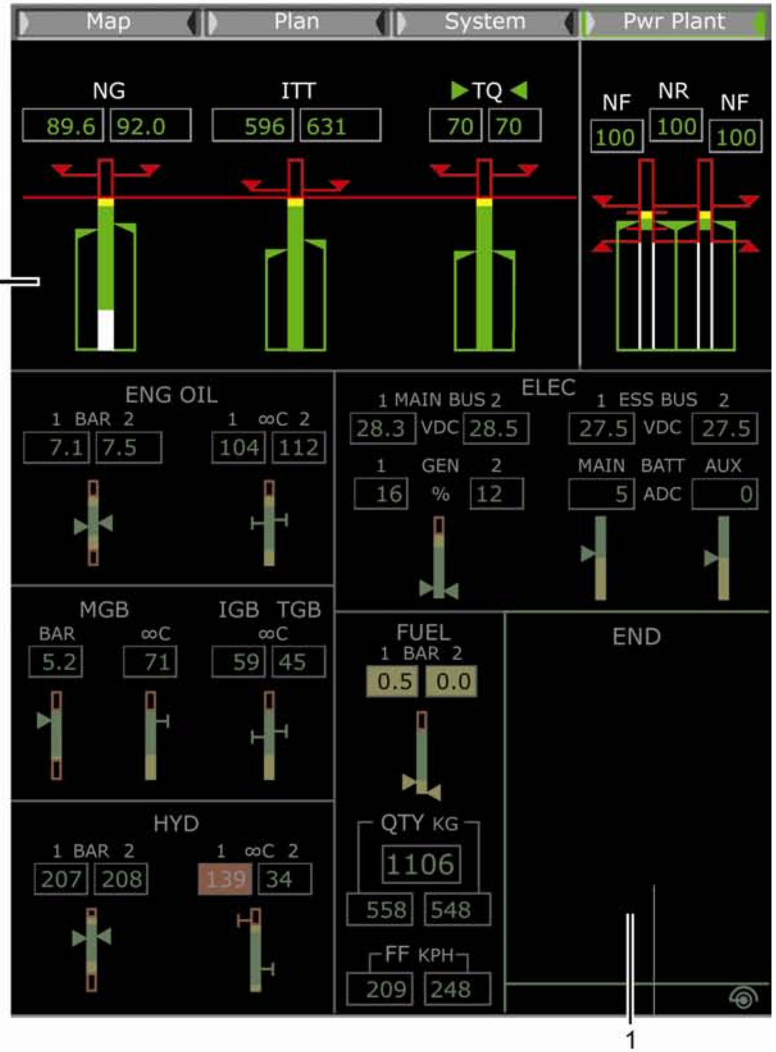
- the amber legend TNG in reverse video is also displayed on the TQ scale (MFD)

- the display of any of the three primary parameters are prioritized as follows, from highest to lowest: TQ, ITT, NG

- the display of the above information on each engine side occurs independently from each other. For instance, TQ may be limiting the engine 1 and ITT limiting the engine 2

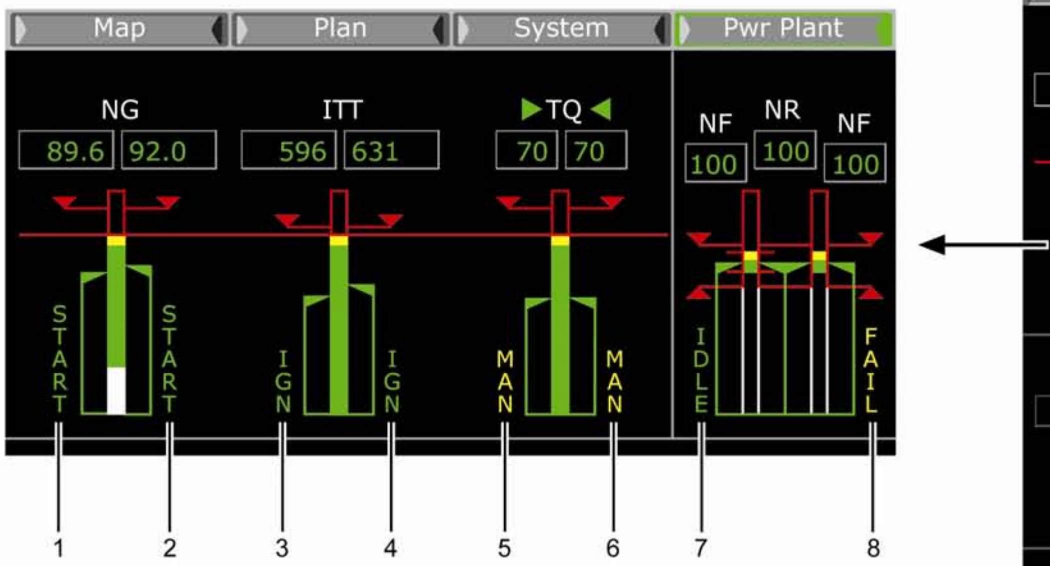
- when a parameter being monitored exceeds the normal range of operation (green band), the colour of associated pointer matches the colour of applicable range making (i.e. amber or red), in order to highlight that particular critical condition

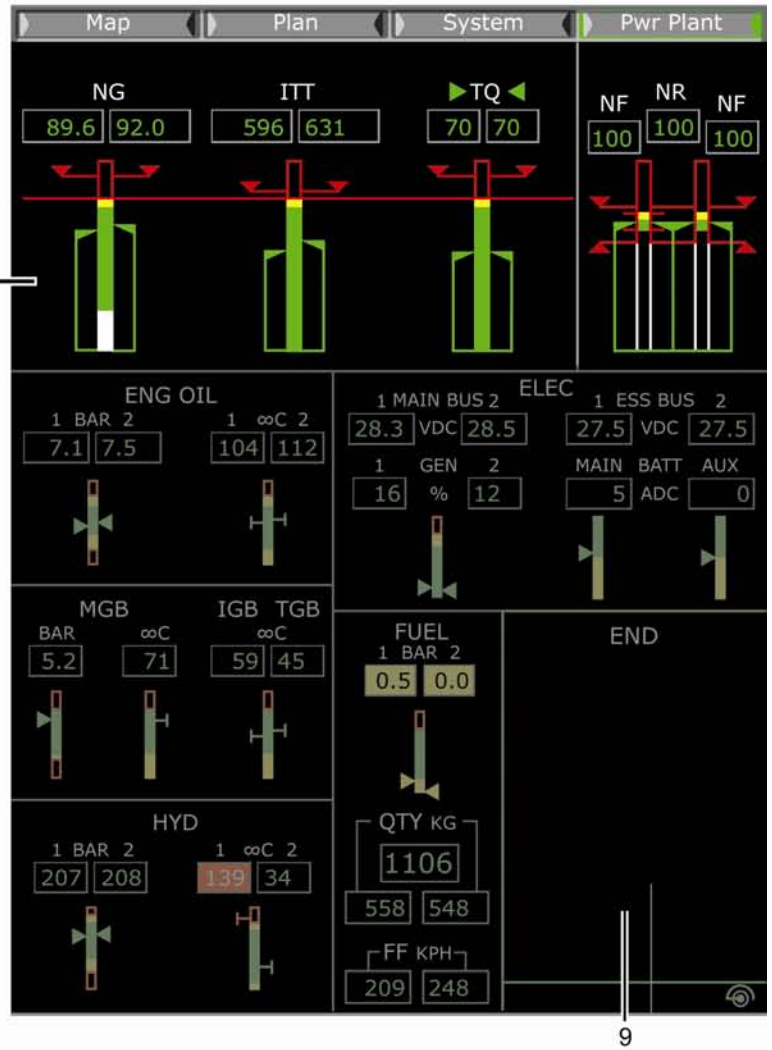




1. CASmessage window No.1
2. ENG.1 NG digital readout
3. ENG.2 NG digital readout
4. ENG.1 ITT digital readout
5. LD SHARE GREEN LEGENDS
6. ENG.2 ITT digital readout
7. ENG.1 TQ digital readout
8. ENG.2 TQ digital readout
9. NR digital readout
10. ENG.2 NF analog scale pointer
11. NR/NF analog scale
12. NR analog scale pointer
13. ENG.1 NF analog scale pointer
14. ENG.2 TQ analog scale pointer
15. ENGINE TQ analog scale
16. ENG.1 TQ analog scale pointer
17. ENG.2 ITT analog scale pointer
18. ENGINE ITT analog scale
19. ENG.1 ITT analog scale pointer
20. ENG.2 NG analog scale pointer
21. ENGINE NG analog scale
22. ENG.1 NG analog scale pointer

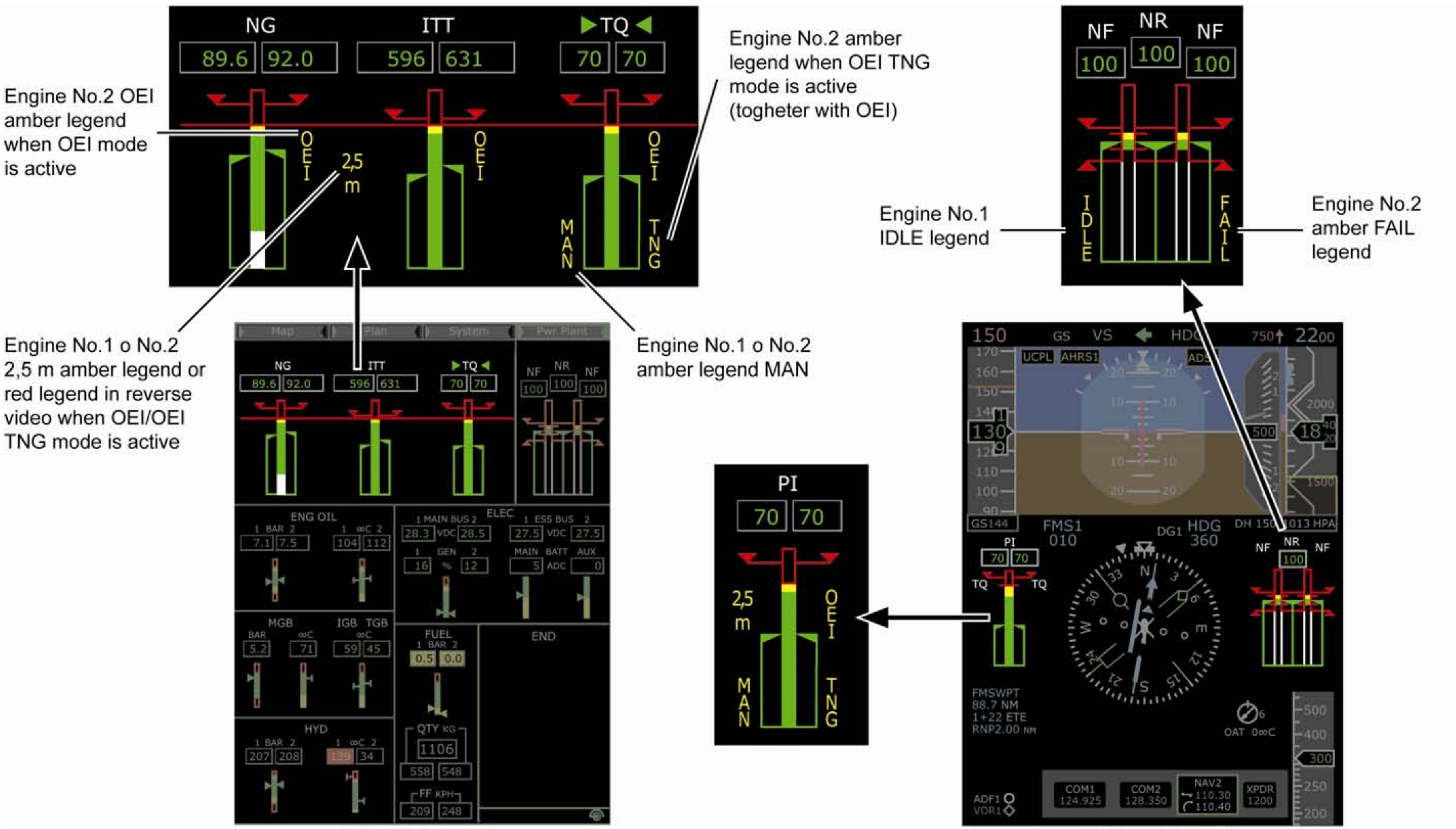
ENGINE INDICATING





1. Engine No.1 START green legend
2. Engine No.2 START green legend
3. Engine No.1 IGN green legend
4. Engine No.2 IGN green legend
5. Engine No.1 MAN amber legend
6. Engine No.2 MAN amber legend
7. Engine No.1 IDLE legend
8. Engine No.1 FAIL amber legenD
9. CAS message window #1

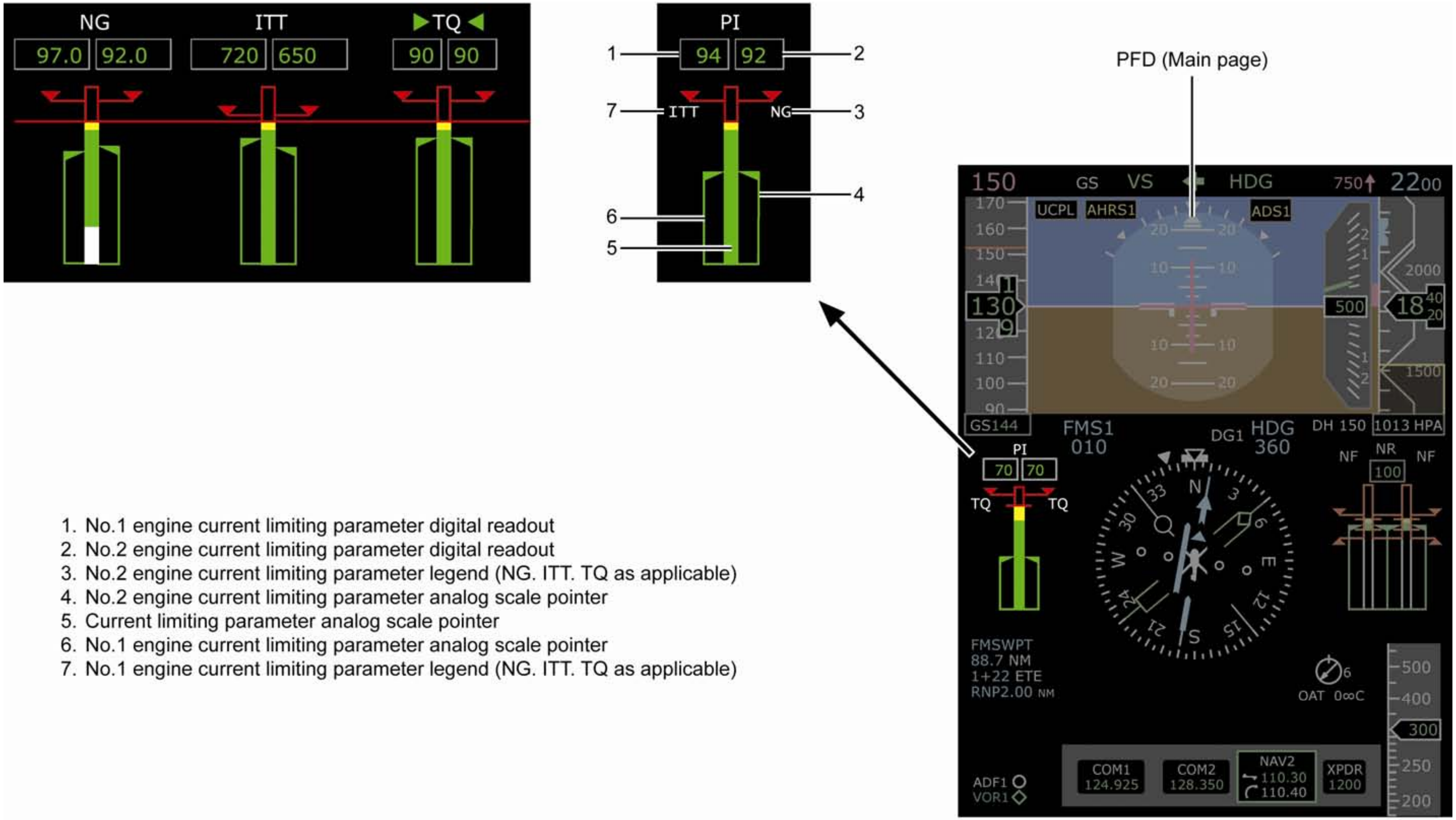
ENGINE STARTING INDICATING



MFD (Main page)

PFD (Main page)

ENGINE INDICATING



1. No.1 engine current limiting parameter digital readout
2. No.2 engine current limiting parameter digital readout
3. No.2 engine current limiting parameter legend (NG. ITT. TQ as applicable)
4. No.2 engine current limiting parameter analog scale pointer
5. Current limiting parameter analog scale pointer
6. No.1 engine current limiting parameter analog scale pointer
7. No.1 engine current limiting parameter legend (NG. ITT. TQ as applicable)

POWER INDEX INDICATING

ENGINE FAILURE

GENERAL

In the event of partial or complete power failure, establishing a safe flight condition is the prime consideration, until the cause of the failure can be analysed.

Care should be taken in confirming the failed engine prior to beginning engine shutdown as given in the ENGINE SHUTDOWN IN AN EMERGENCY procedure.

ENGINE FAILURE RECOGNITION

The following cues will be available to the crew following a single or multiple engine failure

- noticeable right sideslip (helicopter nose swinging to the left)
- illumination of the CAS Warning 1(2) ENG OUT caption
- audio tone plus ENGINE 1(2) OUT voice warning
- the failed engine PI/torque will split significantly from the operational engine
- dependent on collective position at the time of the failure, a drop in rotor speed (NR) may occur

SINGLE ENGINE FAILURE

A single engine failure will result in an increase in PI/torque on the live engine. Depending on collective position and airspeed at the time of the failure, a drop in rotor speed (NR) may occur requiring a collective pitch adjustment in order to maintain rotor speed within the Power On range. If the execution of the ENGINE FAILURE procedure has resulted in shutting down the engine, consider analyzing the cause of the failure with a view toward re-starting the engine. To attempt the re-start use the ENGINE RESTART IN FLIGHT procedure.

DOUBLE ENGINE FAILURE

A sequential or simultaneous failure of both engines will require entry into autorotation.

ENTRY IN AUTOROTATION

Depending on collective pitch and airspeed at the time, a simultaneous engine failure will result in a large and very rapid drop in rotor speed (NR) requiring a large and rapid collective pitch adjustment in order to recover and maintain rotor speed within the Power Off range. It is imperative that these adjustment be made quickly and decisively.

If the failure occurs at considerable height Above Ground Level (AGL), it is possible that sufficient time will be available for attempting an engine re-start (assuming that the cause of the failure can be rapidly analysed). Assuming an average autorotative sink rate of 2500 feet per minute, a minimum AGL height of 3000 to 4000 feet would be required to provide sufficient time to complete the re-start procedures.

- If time and conditions permits and no attempt to restart is made, carry out the ENGINE SHUTDOWN IN AN EMERGENCY procedure while the helicopter is manoeuvred toward the landing area.

- If sufficient additional time is available to make an engine re-start feasible, use the ENGINE RESTART IN FLIGHT procedure.

CAS WARNING MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
1(2) ENG OUT	Associated engine NG less than 34.3% or rate of change outside predetermined limits NOTE • if the ECC 1 (2) is faulty, the MAU detect the engine failure through the NG1 backup signal whose threshold point is active at NG <50% and inactive at NG >55% • if the NG1 (2) coming from the EEC 1 (2) is not valid, the relevant backup analog values is considered • the AWG provides for two pairs of tones + voice warning ENGINE 1 (2) OUT – ENGINE 1 (2) OUT repeated only once • this message has priority number 2	ENGINE OUT	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES ENGINE
1(2) ENG IDLE	Associated engine in IDLE and collective being raised (Triggered on ground only) NOTE • the AWG provides for the voice warning 1(2) ENG IDLE repeated once	ENGINE IDLE	

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
1(2) ENG OIL PRESS	Associated engine oil pressure below limit (less than 4.2 bar; resets above 4.7 bar) NOTE • the AWG provides for the voice warning WARNING – WARNING repeated once	ENGINE OIL PRESSURE LOW	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES ENGINE
1(2) EEC FAIL	the no.1 (2) engine control system is not operating due to a critical fault NOTE • the EEC 1 (2) will revert to MANUAL mode automatically with the amber legend MAN in reverse video displayed on the TQ/PI scales. The critical fault is latching and the EEC 1 (2) will not resume engine control in AUTO mode even through signal becomes valid again, unless the AUTO/MANUAL switch is cycled from AUTO to MANUAL back to AUTO or upon the EEC 1 (2) power reset • the AWG provides for the voice warning WARNING – WARNING repeated once • this message has priority number 5	ENGINE EEC FAIL	

COMPRESSOR STALL

A compressor stall is normally recognized by an abnormal increase of ITT (or abnormal increase in PI indication with ITT as limiting parameter) and may be accompanied by an audible bang or pop and fluctuating NG and TQ (monitored on MFD PWR PLANT page). The compressor stall may be transient or steady.

The degree of compressor stall may be indicated by one or all of the following

- a rapid increase in PI with ITT as limiting parameter.
- fluctuating NG speed coupled with failure to respond to power demand.
- loud banging or popping noises.
- a reduction in torque.
- if compressor stall occurs, carry out the procedure of the RFM

If compressor stall occurs, the pilot has to carry out the procedure described in the AW139-RFM-4D Section 3.

UNUSUAL ENGINE NOISE

Compressor damage as a result of FOD may increase the engine noise level and is detectable by a high-pitched whining sound. The noise level of the high pitched whine should vary with NG (monitored on MFD PWR PLANT page) and should be significantly higher than the usual engine noise.

If an unusual noise is detected and FOD damage is suspected

1. switch ENG MODE to IDLE sequentially to determine the affected engine
2. shutdown the affected engine as soon as practicable to avoid possible secondary compressor damage
3. land as soon as practicable

CAS CAUTION MESSAGES

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
1(2) OVSPD	NF approx 110% and/or PI 0%, drive shaft failure on affected engine	ENGINE DRIVE SHAFT FAILURE	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES ENGINE
1(2) ENG LIM EXPIRE	associated engine has exceeded OEI 2.5 minute time rating	ENGINE LIMIT EXCEEDANCE	
1(2) ENG OIL TEMP	associated engine oil temperature high (greater than 140°C; resets below 137°C)	ENGINE OIL TEMPERATURE	
	if engine oil pressure operates continuously above oil pressure limit	ENGINE OIL PRESSURE HIGH	
1(2) ENG CHIP	chip detected in the associated engine oil lubricating system	ENGINE OIL CHIP DETECTOR	
1(2) FIRE DET	associated engine fire detect system not operational	ENGINE FIRE DETECTOR SYSTEM	
1(2) ECL FAIL	associated ECL not responding to internal Built In Test or link failure between the trim switch and the electrical motor in the ECL	ENGINE CONTROL LEVER	
1(2) ECL POS	associated engine ECL out of FLIGHT position detent (only active when engine control in AUTO mode)	ENGINE CONTROL LEVER POSITION	

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
1(2) ENG MODE SEL	associated engine MODE SELECT switch failure	ENGINE MODE SELECT SWITCH	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES ENGINE
1(2) OVSPD	associated engine NF in overspeed condition	ENGINE POWER TURBINE OVERSPEED	
1(2) OVSPD DET	associated engine NF overspeed detection system not operational	ENGINE POWER TURBINE OVERSPEED DETECT FAILURE	
1(2) EEC DATA	associated EEC link interface between EEC and MAU is lost (non critical fault)	ENGINE ELECTRONIC CONTROL DATA	
1(2) DCU	associated engine control function degraded	DEGRADATION OF ENGINE CONTROL FUNCTIONS	
1(2) HOT START	associated engine ITT limits exceeded during start	ENGINE HOT START	
1(2) TQ LIMITER	associated engine torque limiter system not functioning	TORQUE LIMITER	
1(2) ITT LIMITER	associated engine ITT limiter not functioning	INTER TURBINE TEMPERATURE LIMITER	
RPM SELECT	RPM select switch for engine 1 or 2 malfunction	ROTOR SPEED SELECTOR	

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
1(2) FUEL FILTER	associated fuel filter blockage, impending by-pass condition	FUEL FILTER BY-PASS	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES ENGINE
1(2) FUEL ICING	associated fuel temperature less than 5°C, possible fuel heater malfunction and fuel icing	FUEL ICING	
1(2) FUEL HEATER	associated fuel temperature greater than 74°C or fuel temperature less than 5°C with associated engine oil temperature greater than 82°C. Possible fuel heater malfunction. (Caution only active with associated engine running → NG 1 (2) above 60%)	FUEL HEATER	
AVIONIC FAULT	loss of communications to a single MAU is detected for EEC 1 and EEC 2 NOTE The maintenance message in the CMC is A429/RS422 BUS	AVIONIC	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES AVIONICS

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
1(2) NF MISCOMPARE OR 1(2) NG MISCOMPARE OR 1(2) ITT MISCOMPARE OR 1(2) TQ MISCOMPARE OR NR MISCOMPARE	associated parameter EEC and analogue backup data comparison discrepancy. The MISCOMPARE caution is generated when comparison with backup parameter exceeds the following values: NF 3% NG 3% ITT 50°C TQ 5% NR 3% NOTE the analogue sensors are selected from MFD PWR PLANT page, menu selection using Cursor Control Device (CCD)	ENGINE AND ROTOR PARAMETERS MISCOMPARE	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES ENGINE
ENG ANALOG FAILURE	engine analogue monitoring systems failed NOTE the analogue sensors are selected from MFD PWR PLANT page, menu selection using Cursor Control Device (CCD)	ENGINE ANALOGUE SENSOR FAILURE	

CAS CAPTION	FAILURE DESCRIPTION	PROCEDURE NAME	AW139-RFM-4D
AVIONIC FAULT	Loss of communication to a single MAU is detected for FCU1 or FCU2	AVIONIC FAULT	Section 3 EMERGENCY AND MALFUNCTION PROCEDURES AVIONIC
1 (2) MAU OVHT	Associated MAU overheat <u>If MAU1 fails</u> The secondary engine parameter 1 FUEL PUMP is not valid and the following CAS cautions are not available: • 1 FUEL HEATER • 1 FUEL ICING <u>If MAU2 fails</u> The secondary engine parameter 2 FUEL PUMP is not valid and the following CAS cautions are not available: • 2 FUEL HEATER • 2 FUEL ICING	MODULAR AVIONICS UNIT OVERHEAT / FAIL	

CAS ADVISORY MESSAGES

CAS CAPTION	FAILURE DESCRIPTION		AW139-RFM-4D
TQ LIMITER ON	engine torque limiter activated		Section 2 NORMAL PROCEDURES ADVISORY CAPTIONS DEFINITIONS

ENGINE LIMITATIONS

Refer to AW139-RFM-4D Section 1 for

- engine limitations
- NG limitations
- ITT limitations
- TQ limitations
- NR/NF limitations
- POWER INDEX limitations

PERFORMANCE CHECK

The performance check allows to verify the engine condition over a wide range of ambient temperatures without exceeding any limits. The performance check should be performed

- after engine installation
- at regular interval as per flight manual

All forms of engine deterioration can cause increased Inter Turbine Temperatures (ITT) and fuel consumption at a given power. Compressor deterioration causes increase of Gas generator speed (NG) at given power. Hot section deterioration causes decreases in NG at given power settings.

The physical aspects considerably influence the performance parameters of an installed engine. For this reason, the completed hover power checks procedure is described in Section 4 - Performance Data of the AW139 RFM. Refer to this section for a detailed description.
In general the charts establishes acceptable engine parameter limits for different atmospheric conditions. The check is performed at a given power (normalized torque).

EXAMPLE

A ground power check on the engine no.1 give the following results

- TRQ = 102%
- Pressure Altitude = 5000 ft
- OAT = -10°C
- ITT = 600
- NG = 91

SOLUTION

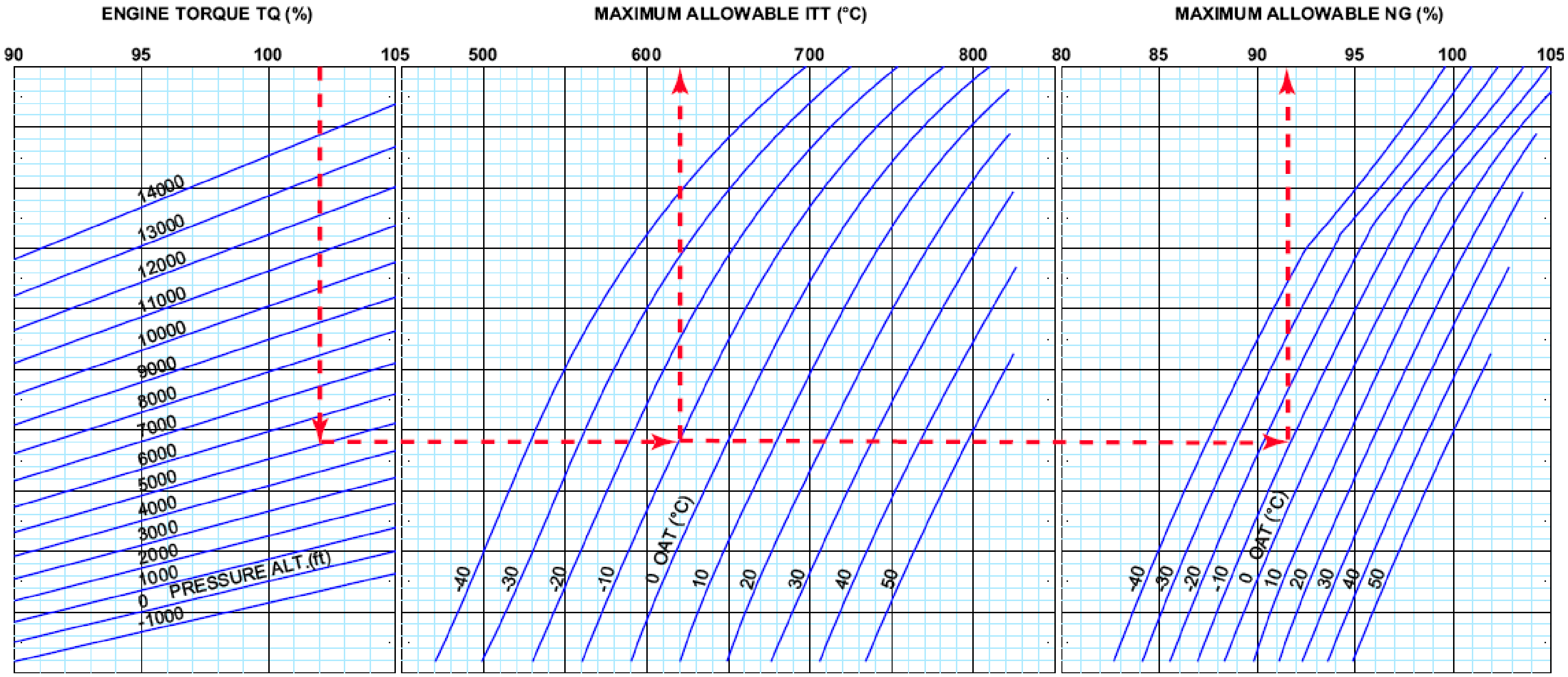
1. Entering the left of the graph at 102% torque.
2. Drop down to the Pressure Altitude curve for 5000 ft.
3. Move right to the -10°C OAT curve for Maximum Allowable ITT and Maximum allowable NG.
4. From the -10°C OAT curves move vertically up to an ITT value of 620°C and NG value or 91.5%.

CONCLUSIONS

The recorded ITT value of 600° C is less than the maximum allowable (620° C) and the NG value of 91% is less than the maximum allowable (91.5%) so the engine is acceptable for flight.

POWER ASSURANCE CHECK in HOVER FLIGHT (NR=100%)

HEATER/COND OFF
GENERATOR LOAD TO MINIMUM (BELOW 17%)
SET NR to 100%
TEST ENGINE MODE SWITCH: FLIGHT
OTHER ENGINE MODE SWITCH: IDLE
INCREASE COLLECTIVE UNTIL LIGHT ON WHEELS OR HOVERING AT 5 FEET, NOSE ON WIND. DO NOT EXCEED 775°C ITT OR 102.4% NG OR 100% TQ
STABILIZE POWER 1 MINUTE, THEN RECORD OAT, PRESSURE ALTITUDE, ENGINE TORQUE, ITT AND NG
ENTER CHART AT INDICATED TQ, MOVE DOWN TO INTERSECT PRESSURE ALTITUDE, PROCEED TO THE RIGHT TO INTERSECT OAT, THEN MOVE UP TO READ VALUES FOR MAXIMUM ALLOWABLE ITT AND NG
IF INDICATED ITT OR NG EXCEEDS MAXIMUM ALLOWABLE, REPEAT CHECK
REPEAT CHECK USING OTHER ENGINE
IF EITHER ENGINE EXCEEDS ALLOWABLE ITT OR NG, PUBLISHED PERFORMANCE MAY NOT BE ACHIEVABLE. REFER TO EMM



PWC PT6C-67C HOVER POWER CHECK CHART

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CHAPTER 79

ENGINE OIL

SECTION 00 – GENERAL

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ENGINE OIL – GENERAL

The engine oil system supply a flow of filtered oil to the engine in order to cool, lubricate and clean different components.

The oil system consists of

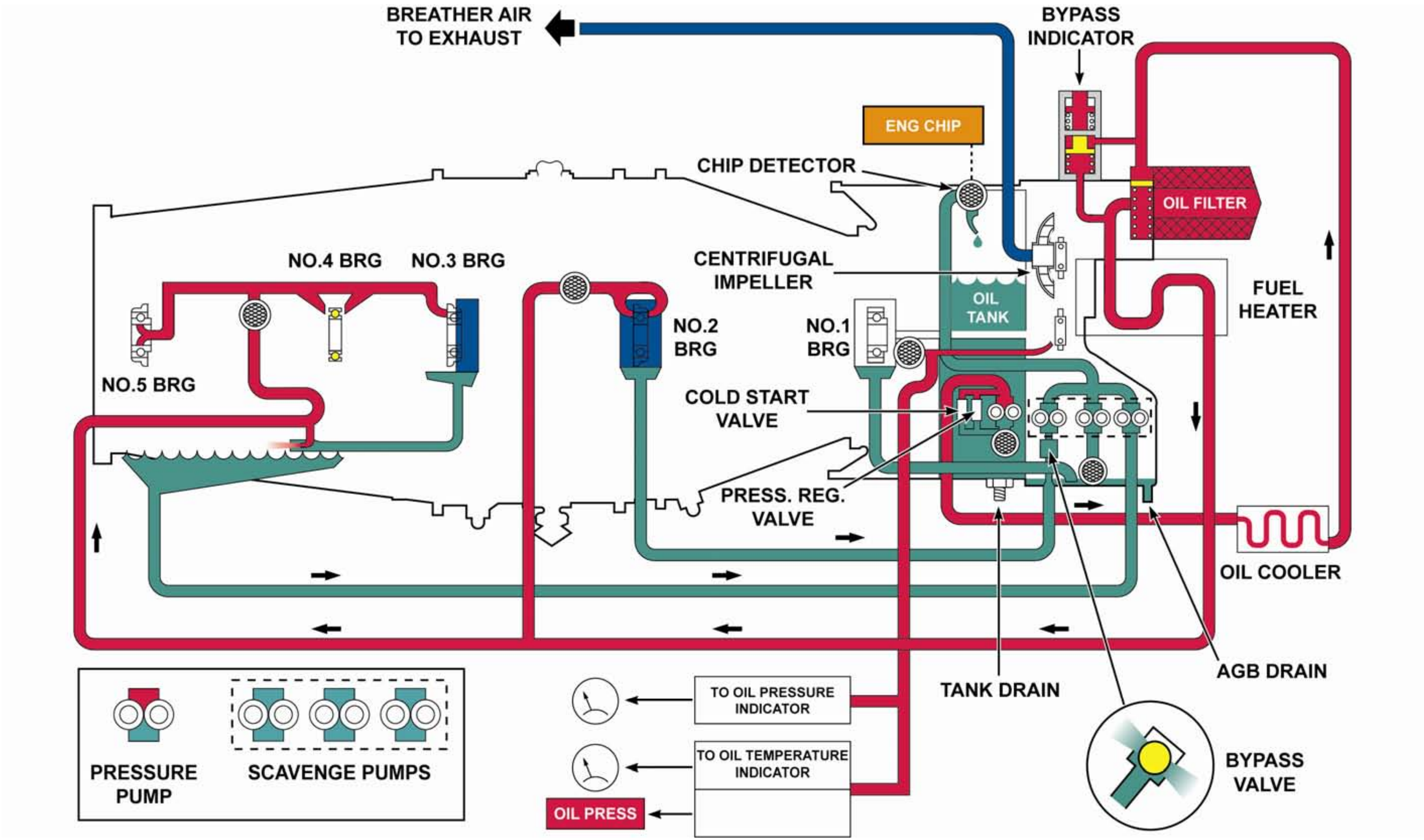
- an integral oil tank
- a pressure system
- a scavenge system
- a breather system

The oil tank is integral with the engine. It is the annular cavity created between the air inlet case and the accessory gearbox rear case. A drain plug located at the bottom of the Accessory Gear Box (AGB) permits drainage of the cavity.

Oil level indication is achieved by means of two oil level sight glasses on each side of the AGB.
A pressure-regulating valve regulates the pressure system.

The scavenge system returns the oil to the AGB by means of gravity and dedicated pumps. All returned oil flow by a magnetic chip detector located at the inlet of the oil tank.

P2.5 and P3 air pressures are used to pressurize various bearing cavities for sealing. The air/oil mixture from the bearing cavities is routed to the AGB via the scavenge system. The air is separated from the oil and vented overboard through the centrifugal impeller.


BREATHER AIR TO EXHAUST
BYPASS INDICATOR
ENG CHIP
CHIP DETECTOR
OIL FILTER
CENTRIFUGAL IMPELLER
FUEL HEATER
NO.4 BRG
NO.3 BRG
OIL TANK
NO.2 BRG
NO.1 BRG
NO.5 BRG
COLD START VALVE
PRESS. REG. VALVE
OIL COOLER
AGB DRAIN
TANK DRAIN
TO OIL PRESSURE INDICATOR
TO OIL TEMPERATURE INDICATOR
OIL PRESS
BYPASS VALVE
PRESSURE PUMP
SCAVENGE PUMPS

OIL – GENERAL

ENGINE OIL – CONTROLS AND INDICATIONS

The engine oil controls and indications are obtained by means of

- a chip detector
- an oil pressure transducer
- a low oil pressure switch
- an oil temperature transducer

If a ferrous particle is detected, a signal is provided to MAU that, in turn, generates the relevant caution message. The pilot can try to burn the particle. The TEST control panel provides for the system test.

1. CHIP DETECTOR ENG 1 push-button switch

   PRESSED …. the eng 1 chip detector test is provided and the caution 1 ENG CHIP appears in inverse video on the MFD

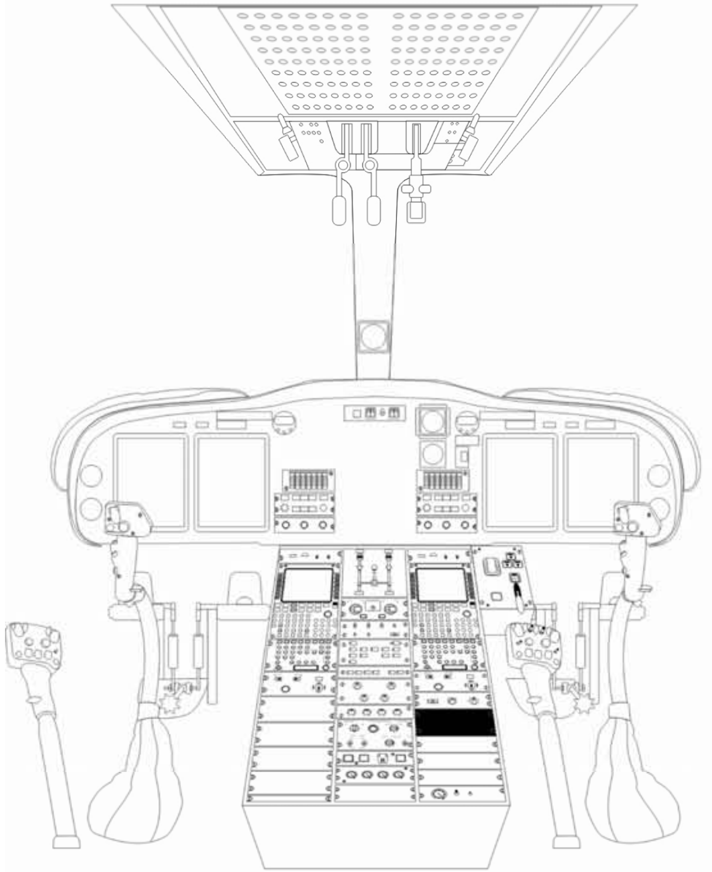
2. CHIP DETECTOR ENG 2 push-button switch

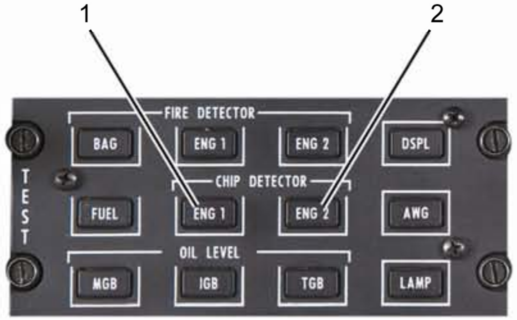
   PRESSED …. the eng 2 chip detector test is provided and the caution 2 ENG CHIP appears in inverse video on the MFD

NOTE.

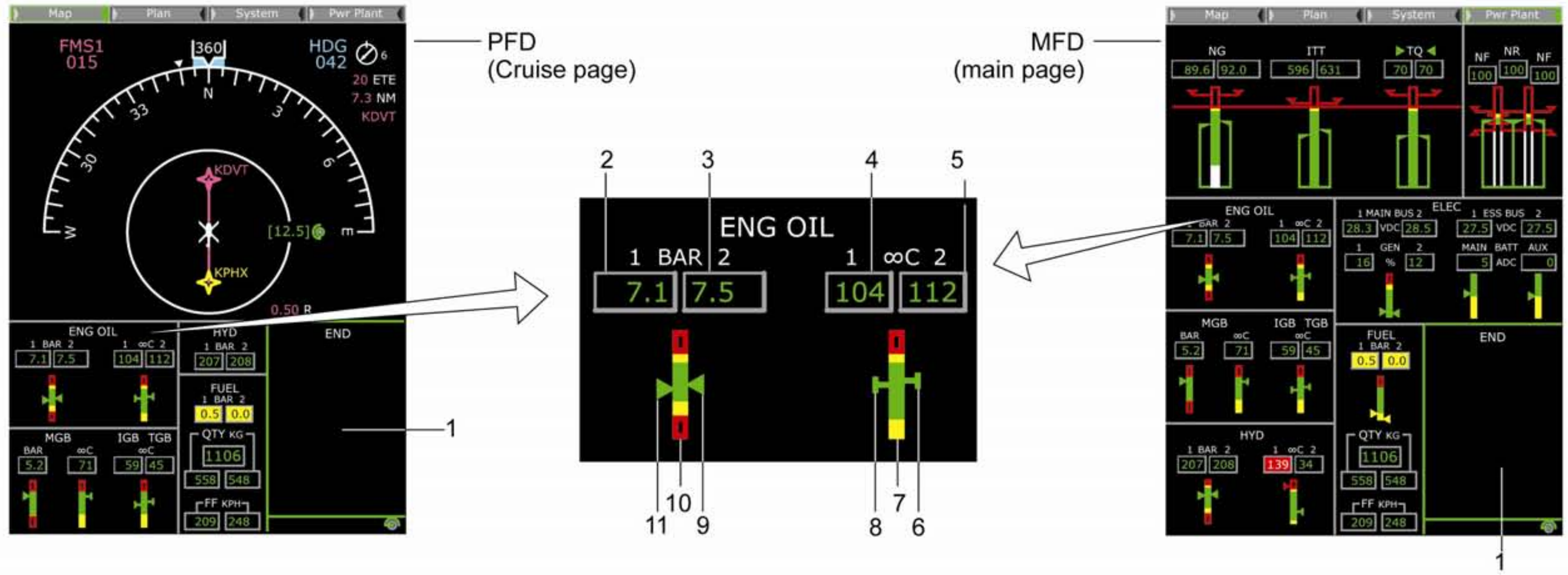
The oil pressure transducer and the oil temperature transducer send a signal to MAU 1 (for the engine 1) or to MAU 2 (for the engine 2) which generate the relevant caution message (see Chapter 76).

If the oil pressure drops below a set value, the low oil pressure switch send a signal to MAU 1 (engine 1) or to MAU 2 (engine 2) which generate the relevant warning message (see Chapter 76).





ENGINE OIL CONTROLS AND INDICATIONS

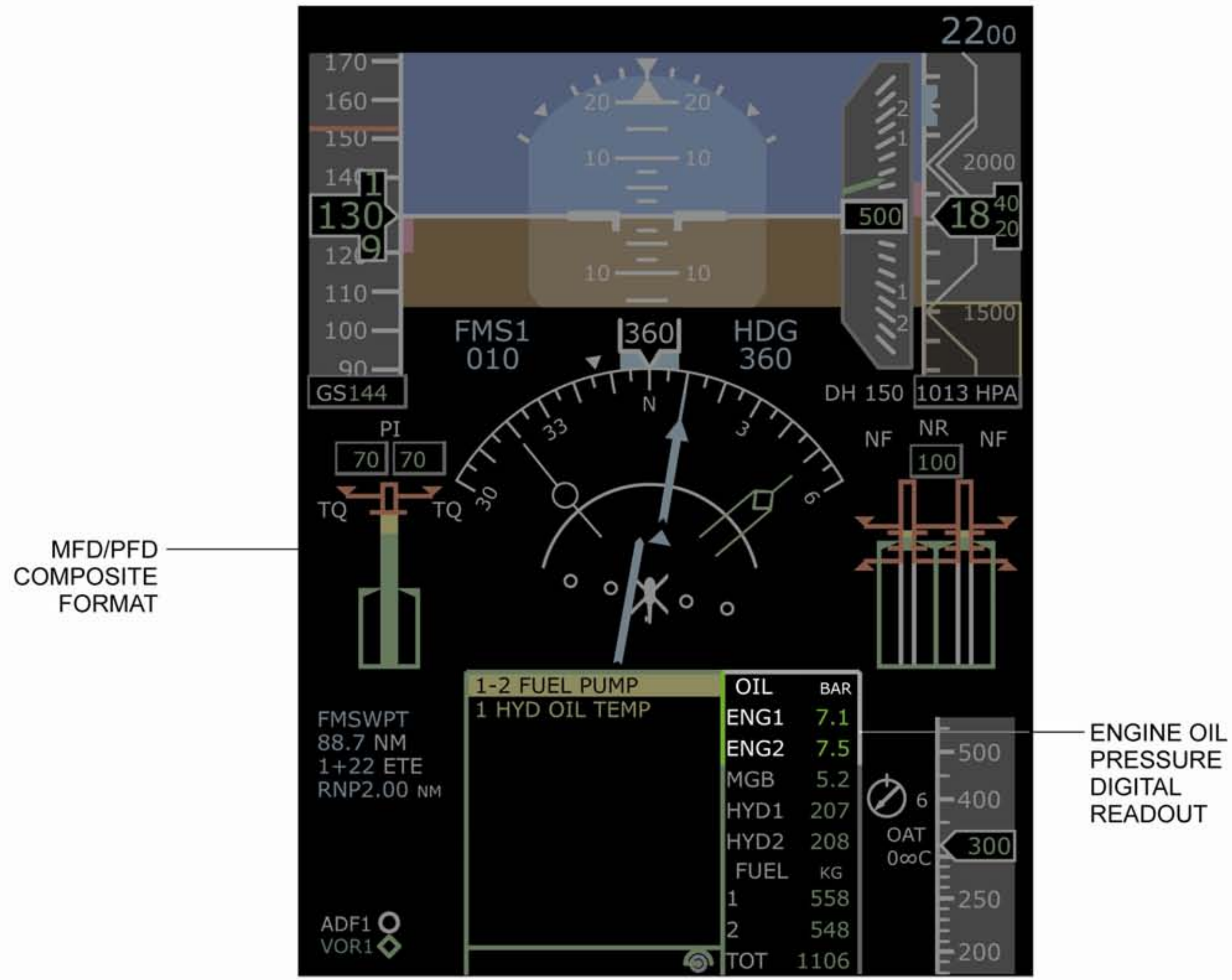


1. CAS message window N.1
2. ENG.1 OIL PRESSURE digital readout
3. ENG.2 OIL PRESSURE digital readout
4. ENG.1 OIL TEMPERATURE digital readout
5. ENG.2 OIL TEMPERATURE digital readout
6. ENG.2 OIL TEMPERATURE analog scale pointer
7. ENGINE OIL TEMPERATURE analog scale
8. ENG.1 OIL TEMPERATURE analog scale pointer
9. ENG.2 OIL PRESSURE analog scale pointer
10. ENGINE OIL PRESSURE analog scale
11. ENG.1 OIL PRESSURE analog scale pointer

NOTE When the loss of an analog input signal or an out of range input condition is detected, the asoociated pointer is removed from the display and amber dashes replace the numerical readouts.

NOTE When a parameter being monitoring exceeds the normal range of operation (green band), the color of associated pointer matches the color of applicable range marketing(i.e amber or red), in order to highlight that particular critical condition.

OIL INDICATIONS (1 OF 2)



OIL INDICATIONS (2 OF 2)